

Exploring the design space for a hybrid-electric regional aircraft with multidisciplinary design optimisation methods

Jérôme Thauvin

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> Directeur(s) de Thèse : M. XAVIER ROBOAM M. MARC BUDINGER

Rapporteurs :

M. MIRKO HORNUNG, TECHNISCHE UNIVERSITAT MUNICH M. STÉPHANE BRISSET, ECOLE CENTRALE DE LILLE

Membre(s) du jury :

Mme DELPHINE RIU, INP DE GRENOBLE, Président M. BRUNO SARENI, INP TOULOUSE, Membre M. MARC BUDINGER, INSA TOULOUSE, Membre M. XAVIER ROBOAM, CNRS TOULOUSE, Membre

Résumé

Envisagée à partir des 15 à 30 années à venir dans l'industrie aéronautique, la propulsion hybride électrique permet d'intégrer de nouvelles briques technologiques offrant des degrés de liberté supplémentaires pour améliorer les performances des aéronefs, limiter l'utilisation de ressources fossiles et réduire l'impact environnemental des avions. Aujourd'hui, la technologie hybride électrique est principalement appliquée aux transports terrestres, aux voitures, aux bus et aux trains, mais aussi aux navires. La faisabilité pour le transport aérien doit encore être établie et l'amélioration des performances des aéronefs reste à démontrer. Cette thèse vise à évaluer les gains énergétiques apportés par l'hybridation électrique d'un avion régional de 70 places.

Tout d'abord, les opportunités d'économie d'énergie sont identifiées à partir de l'analyse des rendements propulsifs et aérodynamiques d'un avion bi-turbopropulsé conventionnel. Les gains potentiels provenant de la variation de la taille des moteurs principaux et de nouvelles gestions de puissance par l'utilisation de batteries sont étudiés. De plus, les possibles améliorations aérodynamiques émanant de nouveaux positionnements des hélices sont considérées. Pour chaque sujet, des analyses simplifiées fournissent une estimation d'économie d'énergie. Ces résultats sont ensuite utilisés pour sélectionner quatre systèmes propulsifs électrifiés qui sont étudiés plus en détail dans la thèse: un hybride parallèle, un turboélectrique avec propulsion distribuée, un turboélectrique partiel à hélices hypersustentatrices, et un tout-électrique.

L'évaluation des avions hybrides électriques sélectionnés est d'autant plus difficile que le dimensionnement des différentes composants, les stratégies de gestion d'énergie et les profils de mission que l'on peut imaginer sont nombreux et variés. En outre, le processus global de conception de l'avion et les outils d'évaluation doivent être adaptés en conséquence. L'outil interne de conception par optimisation multidisciplinaire d'Airbus nommé XMDO, qui inclut la plupart des modifications requises, est finalement sélectionné et développé au cours de la thèse. Par exemple, de nouveaux modèles paramétriques de composants (voilure soufflée, moteur électrique, turbine à gaz, hélice, etc...) sont créés, une formulation générique pour résoudre l'équilibre du système de propulsion est mise en place, et les modèles de simulation de décollage et d'atterrissage sont améliorés.

Afin d'évaluer l'efficacité énergétique des avions hybrides électriques, un avion de référence équipé d'un système propulsif conventionnel est d'abord optimisé avec XMDO. Différents algorithmes d'optimisation sont testés, et la consistance de la nouvelle méthode de conception est vérifiée.

Par la suite, les configurations hybrides électriques sont toutes optimisées selon les mêmes exigences de conception que l'avion de référence. Pour les composants électriques, deux niveaux de technologie sont définis selon la date d'entrée en service de l'aéronef. Les résultats d'optimisation pour le turbo-électrique et le turbo-électrique partiel sont utilisés pour mieux appréhender les gains aérodynamiques potentiels identifiés en première partie de thèse. Les optimisations pour l'hybride parallèle, comprenant différents scénarios de recharge batterie, mettent en évidence les meilleures stratégies de gestion d'énergie lorsque des batteries sont utilisées comme sources d'énergie secondaire. Tous les résultats sont finalement comparés à la référence en termes de consommations de carburant et d'énergie, pour les deux niveaux de technologie électrique.

La dernière partie de la thèse se concentre sur l'avion tout électrique. Elle vise à identifier l'énergie spécifique minimale requise pour les batteries en fonction de la distance maximale à parcourir. Une étude de sensibilité est également réalisée en fonction de la date d'entrée en service pour les autres composants électriques.

Abstract

Envisioned in the next 15 to 30 years in the aviation industry, hybrid-electric propulsion offers the opportunity to integrate new technology bricks providing additional degrees of freedom to improve overall aircraft performance, limit the use of non-renewable fossil resources and reduce the aircraft environmental footprint. Today, hybrid-electric technology has mainly been applied to ground-based transports, cars, buses and trains, but also ships. The feasibility in the air industry has to be established and the improvement in aircraft performance has still to be demonstrated. This thesis aims to evaluate the energy savings enabled by electric power in the case of a 70-seat regional aircraft.

First, energy saving opportunities are identified from the analysis of the propulsion and aerodynamic efficiencies of a conventional twin turboprop aircraft. The potential benefits coming from the variation of the size of prime movers and the new power managements with the use of batteries are studied. Also, possible aerodynamic improvements enabled by new propeller integrations are considered. For each topic, simplified analyses provide estimated potential of energy saving. These results are then used to select four electrified propulsion systems that are studied in more detail in the thesis: a parallel-hybrid, a turboelectric with distributed propulsion, a partialturboelectric with high-lift propellers and an all-electric.

Evaluating the selected hybrid-electric aircraft is even more challenging that the sizing of the different components, the energy management strategies and the mission profiles one can imagine are many and varied. In addition, the overall aircraft design process and the evaluation tools need to be adapted accordingly. The Airbus in-house Multidisciplinary Design Optimisation platform named XMDO, which includes most of the required modifications, is eventually selected and further developed during the thesis. For examples, new parametric component models (blown wing, electrical motor, gas turbine, propeller, etc...) are created, a generic formulation for solving the propulsion system equilibrium is implemented, and simulation models for take-off and landing are improved.

In order to evaluate the energy efficiency of the hybrid-electric aircraft, a reference aircraft equipped with a conventional propulsion system is first optimised with XMDO. Different optimisation algorithms are tested, and the consistency of the new design method is checked.

Then, all the hybrid-electric configurations are optimised under the same aircraft design requirements as the reference. For the electrical components, two levels of technology are defined regarding the service entry date of the aircraft. The optimisation results for the turboelectric and the partial-turboelectric are used to better understand the potential aerodynamic improvements identified in the first part of the thesis. Optimisations for the parallel-hybrid, including different battery recharge scenarios, highlight the best energy management strategies when batteries are used as secondary energy sources. All the results are finally compared to the reference in terms of fuel and energy efficiencies, for the two electrical technology levels.

The last part of the thesis focuses on the all-electric aircraft, and aims at identifying the minimum specific energy required for batteries as a function of the aircraft design range. A trade study is also carried-out in accordance with the service entry date for the other electrical components.

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Acronyms

AoA	Angle of Attack
BLI	Boundary Layer Ingestion
CAS	Calibrated Air speed
CG	Centre of Gravity
COC	Cash Operating Cost
DOC	Direct Operating Cost
EIS	Entry Into Service
FL	Field Length
FPA	Flight Path Angle
HLP	High-Lift Propeller
HTP	Horizontal Tail Plane
IDF	Individual Design Feasible
IM	Induction Machine
ISA	International Standard Atmosphere
LHV	Lower Heating Value
LW	Landing Weight
MAC	Mean Aerodynamic Chord
MCL	Maximum Climb rating
MDF	Multidisciplinary Design Feasible
MDO	Multidisciplinary Design Optimisation
MLW	Maximum Landing Weight
MTO	Maximum Take-Off rating
MTOW	Maximum Take-off Weight
MZFW	Maximum Zero Fuel Weight
NPL	Non-Propulsive Load
OAD	Overall Aircraft Design
OEI	One Engine Inoperative
OEW	Operating Empty Weight

PHE	Potential for Hybridisation in Energy
PHP	Potential for Hybridisation in Power
PMSM	Permanent Magnet Synchronous Machine
SFC	Specific Fuel Consumption
SOC	State Of Charge
STOL	Short Take-Off and Landing
TAS	True Air Speed
TLAR	Top Level Aircraft Requirements
TOFL	Take-Off Field Length
TOW	Take-Off Weight
VTOL	Vertical Take-Off and Landing
VTP	Vertical Tail Plane
WRSM	Wound Rotor Synchronous Machine
ZFW	Zero Fuel Weight

Introduction

Propulsion system innovations have been a key driver of aeronautic evolution. The increase of propulsion performance and efficiency has enabled aircraft to travel at higher speeds over longer ranges while carrying larger payloads. Today the improvement of conventional engine technologies is reaching an asymptote, while future demands on the air transport systems still dictate that aircraft should be less polluting, less noisy and more fuel efficient. In this context, electrified propulsion systems offer the opportunity to transform in the long term the landscape of aircraft propulsion and furthermore enable new aircraft configurations.

In this thesis the term hybrid-electric aircraft is used to define an aircraft that operates more than one type of energy source and/or power flow for propulsion means. Aircraft propulsion is indeed currently limited to kerosene and mechanical transmissions. Hybrid-electric propulsion offers the opportunity to integrate new technology bricks providing additional degrees of freedom to improve overall aircraft performance, limit the use of non-renewable fossil resources and reduce the aircraft environmental footprint. Today, hybrid-electric technology has mainly been applied to ground-based transports, cars, buses and trains, but also ships. However, the feasibility of hybrid-electric propusion in the air industry has to be established and the improvement in aircraft performance has still to be demonstrated. This thesis aims to evaluate the new energy savings enabled by electric power in the case of a 70-seat regional aircraft.

Designing a hybrid-electric aircraft is even more challenging that the sizing of the different components, the energy management strategies and the mission profiles one can imagine are many and varied. This thesis also adresses design methods to deal with the optimisation of the airframe, the propulsion system, the control laws and the trajectories in the same workflow.

Chapter 1 evaluates the new energy saving opportunities with respect to a reference conventional aircraft that was designed for the purpose of this chapter only. These opportunities account for the potential benefits, but do not include the penalties coming from the additional mass or drag imparted by the use of electrical components. The correct evaluation of energy savings requires the implementation of new design methodologies and new design models that are presented in the next chapters.

To begin with, the potential benefits of transient energy storage are estimated. The mission profile of the conventional aircraft is analysed according to energy variations and the results are compared to typical figures for cars, trains, and ships. In addition, the opportunities of recovering energy in descent and during landing are evaluated, and a quick study tends to highlight the best energy management strategies in descent.

The power chain efficiency of the reference aircraft is also studied along its nominal mission in order to identify potential improvements enabled by hybridisation, and a particular attention is given to the propeller and the gas turbine. For the latter, the scale effect on efficiency is discussed.

Then, this chapter addresses potential aerodynamic improvements enabled by new propeller or fan integrations. Concepts such as differential thrust, blown wing, boundary layer ingestion and wing-tip propellers are analysed. In particular, a first order study regarding the implementation of boundary layer ingestion on the reference aircraft is presented.

Lastly, the opportunities of replacing kerosene by a greener source of energy are commented. A special focus is given on batteries and fuel cells as means of providing this energy on-board.

Chapter 2 introduces the different aircraft configurations that are studied in more detail in the last chapters of the thesis.

The general airframe configuration is described, and the conventional propulsion system that is used to optimise the reference aircraft is presented.

Then, the chapter focuses on electrified propulsion system architectures, and starts by addressing their classification. Since different technologies of electrical components can be used for the purpose of transmitting power from generators to propellers, a comparison of serial arrangements of these components is provided. Finally, the selection of electrified propulsion systems for this thesis is commented in accordance with the analyses of Chapter 1, and the associated aircraft configurations are presented: a parallel-hybrid, a turboelectric with distributed propulsion, a partial-turboelectric with high-lift propellers and an all-electric.

Chapter 3 deals with the aircraft design process addressed by the Overall Aircraft Design discipline.

After presenting the constrained multidisciplinary optimisation problem that must be solved when designing an aircraft, a brief overview of the most suitable multidisciplinary optimisation methods is provided.

The second part focuses on the case of conventional aircraft drawing energy from a unique source, namely kerosene. A special attention is given to the mission performance evaluation during which the fuel burn is calculated, and to the iterative Mass-Performance Loop sub-process required to define the total structural weight. Typical constraints and optimisation criteria are also commented, and existing software for carrying out overall aircraft design studies are presented.

The last part addresses the case of hybrid-electric aircraft that can be designed with more than one type of energy storage device. Using energy from multiple sources involves additional variables in solving the design optimisation problem, requiring modifications of the classical design process, and in particular of the performance evaluation modules and of the Mass-Performance Loop. Modifications are proposed, including a new mathematical formulation of the latter. Finally, the most promising software for conducting the studies of this thesis are discussed.

Chapter 4 is dedicated to the Multidisciplinary Design Optimisation (MDO) platform that was eventually selected.

First, the origins and history of this tool, called XMDO, are introduced. After providing an overview of the tool framework, this chapter addresses the XMDO top level notions and methods that are used to model a vehicle and to evaluate its performance along a mission. The MDO architecture implemented in the tool is also discussed, and the tool optimisation features are presented.

The other sections focus on detailed implementations in XMDO: trajectory and control law modelling, aircraft equations of motion and solving, and generic propulsion system modelling. For each topic, the section begins with the presentation of the initial implementation in XMDO and follows with the main developments brought by the author of this manuscript.

In the last section, the different constraints that are defined throughout the chapter for solving the global MDO problem are summarised.

Chapter 5 focuses on the optimisation of the reference aircraft with XMDO.

The design models that are used at aircraft level but also at component levels are first presented. A great deal of work was dedicated to the development of these models: turboshaft, gearbox, propeller, blown wing, etc. Then, the optimisation cases are defined: the different missions, the constraints and the objective function, and the optimisation algorithms that were selected.

Lastly, two optimisation cases deferring from each other by the set of missions included in the optimisation process are compared. In the first case, the objective function is calculated according to a single mission while the second case uses a weighted objective function. The main outputs of these two optimisations are provided and discussed.

Chapter 6 deals with the optimisation of the different hybrid-electric aircraft configurations presented in Chapter 2. In the description of each optimisation case, a particular attention is given to the assumptions relating to the propulsion system modelling and the constraints considered in the optimisation problem. Throughout this chapter, results are compared to the conventional aircraft optimised in Chapter 5.

First, the models and assumptions applying to all the aircraft configurations are exposed, and the different missions considered in the optimisation cases are presented.

Then, the component models that were developed and implemented for the purpose of electrified propulsion system studies are addressed. In order to include the effect of technology improvements with time, two technology levels are defined according to the service entry date of the aircraft: 2025 and 2030+.

The third section presents and analyses the optimisation results for the turboelectric and the partial turboelectric, the two architectures that were selected to better understand the potential aerodynamic improvements identified in Chapter 1.

The fourth section deals with the optimisation of the parallel-hybrid that was defined for the study of fuel savings brought by new energy management strategies. Different battery recharge scenarios are considered, including in-flight recharging.

Afterwards, the results of the previous sections are all compared in terms of fuel and energy consumptions versus the reference aircraft, and conclusions are drawn.

The last section is dedicated to the all-electric aircraft, and aims at identifying the minimum specific energy required for batteries as a function of the aircraft design range. A trade study is also carried-out in accordance with the service entry date for the other electrical components.

Chapter 1

New potentials enabled by electric power for regional aircraft

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This chapter aims to review the new energy saving opportunities enabled by electric power with respect to a reference conventional aircraft that was designed for the purpose of this chapter. These opportunities account for the potential benefits, but do not include the penalties coming from the additional mass or drag imparted by the use of electrical components. The correct evaluation of energy savings requires the implementation of new design methodologies and new design models that are presented in the next chapters.

To begin with, the potential benefits of transient energy storage are estimated. The mission profile of the conventional aircraft is analysed according to energy variations and the results are compared to typical figures for cars, trains, and ships. In addition, the opportunities of recovering energy in descent and during landing are evaluated, and a quick study tends to highlight the best energy management strategies in descent.

The power chain efficiency of the reference aircraft is also studied along its nominal mission in order to identify potential improvements enabled by hybridisation, and a particular attention is given to the propeller and the gas turbine. For the latter, the scale effect on efficiency is discussed.

Then, this chapter addresses potential aerodynamic improvements enabled by new propeller or fan integrations. Concepts such as differential thrust, blown wing, boundary layer ingestion and wing-tip propellers are analysed. In particular, a first order study regarding the implementation of boundary layer ingestion on the reference aircraft is presented.

Lastly, the opportunities of replacing kerosene by a greener source of energy are commented. A special focus is given on batteries and fuel cells as means of providing this energy on-board.

1.1 Aircraft of interest

The Regional Aircraft class usually covers both Regional Jets and Turboprops categories. Regional Jets are jet aircraft with a maximum range capability from 1,000 nautical miles¹ to 2,000 nautical miles, a typical seat capacity between 70 and 100 passengers and flying at a speed below Mach 0.80. For example, Boeing 717, Bombardier CRJ700/900/1000, Embraer EMB170/175/190/195, Comac ARJ-21, Sukhoi Superjet or Mitsubishi MRJ fall into this category. Turboprops are propeller aircraft with a seat capacity ranging from 50 to 75 passengers and a cruise speed generally between Mach 0.4 and Mach 0.65. ATR42/72, Bombardier Q300/400 and Saab 2000 are members of the Turboprops family. They evidently fly over shorter ranges even if some of them, in particular the Saab 2000, are able to reach 1000 nm.

¹Nautical miles is also abbreviated nm in the manuscript. 1 nm = 1,852 m

This thesis focuses on a 70-seat turboprop aircraft. The design of this aircraft, powered by a hybrid or a conventional propulsion system, depends heavily on the range capability requirement. In addition, setting the range capability too high will make the design of a hybrid aircraft even more difficult (see Section 1.2.1). In order to set the right target for the maximum range capability of our aircraft, it is always useful to look at the flight distributions based on current market or predictions. Antcliff and Capristan [9] provide a 2030 prediction of the distance distribution for regional trips between airports in the United States (Figure 1.1). It shows that setting the maximum range capability to 400 nm (741 km) allows capturing 90% of the regional market. Even if these data are related to United States regional market, they are fairly representative of the global regional market. For this reason the thesis deals with the design of a turboprop aircraft capable of flying up to 400 nm.



Fig. 1.1 Regional Jet and Turboprop cumulative trip distribution versus mission range [9]

In order to produce non-confidential but representative data for this first chapter, a conventional twin-turbo propeller (Figure 1.2) was designed with an Airbus internal tool named OCCAM². This tool results from more than ten years of research and has been used in several public scientific works [12] [14] [102]. A description of the tool, formerly named SiMCAD (Simple Models for Conceptual Aircraft Design), is available in [14]. OCCAM is a fast aircraft design tool based on analytical models that can be calibrated thanks to a database comprising around 60 aircraft. The effect of the modification of one or several design parameters on aircraft performance can be evaluated in a few seconds. Throughout this chapter, the conventional twin-turbo

²In reference to Occam's razor problem solving principle

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Fig. 1.2 Project aircraft TP1

Phase	Time (min)	Distance (nm)	Fuel (kg)	% Total Fuel Burn
Taxi out	3.0	0.0	9	2.0
Take-off	1.3	0.0	21	4.8
Climb	16.1	54.6	177	40.0
Cruise	22.0	101.0	186	42.1
Descent	9.9	44.4	32	7.2
Landing	2.0	0.0	14	3.2
Taxi in	1.0	0.0	3	0.7
Total	55.3	200.0	442	100

Table 1.1 TP1 mission overview

propeller aircraft designed with OCCAM is named **TP1**. The engine size and the wing area are constrained by the time to climb (17 min) and the approach speed (113 kts) requirements, respectively, which is generally the case for turboprop aircraft of this category. More details about the TP1 aircraft design requirements are provided in Table 3.1 of Chapter 2. Despite the range capability of 400 nm, the aircraft performance is evaluated on a 200 nm mission (Table 1.1) as this aircraft is expected to operate most of its life on such range (57th percentile of the range distribution of Figure 1.1). The TP1 aircraft is equipped with two turbopropeller engines, each of them providing a maximum power of 2,500 horsepower³ (1.870 MW) at take-off. Figure 1.3 provides the power profile of one engine along the 200 nm mission. Note that it is common to express operating speeds in terms of Mach number (M) or Calibrated Air Speed (CAS⁴). At first order, a given Calibrated Air Speed describes a given dynamic pressure acting on the aircraft. The red curve shows the maximum power than can be provided continuously by the engine along the trajectory without overworking the engine, and corresponding to the Maximum Climb rating (MCL) [138]. During take-off the Maximum Take-Off rating (MTO) of the engine is used providing augmented power versus MCL rating, enabled by higher combustion temperatures that are permitted for up to 5 minutes of operation. Finally, the maximum continuously available power of the engine inevitably falls with altitude because of the decreasing air density.

³1 horse power \approx 745.7 watts

⁴kCAS: Calibrated Air Speed expressed in knots



Fig. 1.3 Power profile of the TP1 aircraft

1.2 Transient energy storage

In this section, the power profile of the TP1 aircraft is analysed according to energy variations. Also, the opportunities of recovering energy in descent and during landing are evaluated.

1.2.1 Energy sharing

Generally, for hybrid architectures using energy storage devices, the main source of power provides the average load of the mission while energy storage devices provide peak loads. This allows the main source of power to be downsized and to increase the overall efficiency of the power generating system. Akli [5] (French written) defined two criteria to assess the relevance of a propulsion system to be hybridised with energy storage devices: the **potential for hybridisation in power** and the **potential for hybridisation in energy**. These indicators are also addressed in english in [134]. The evaluations of these indicators are based on the analysis of the power profile of the non-hybrid power generating system. The definitions proposed by Akli [5] are presented hereafter:

The potential for hybridisation in power PHP_A expresses the potential reduction in size of the main source of power enabled by the use of an energy storage



Fig. 1.4 Two power profiles and associated values for PHP_A [5]

device. It is defined by Equation 1.1, where P_{moy} is the timed-average value of the required power during the mission and P_{max} is the maximum value of the power profile. Figure 1.4 better illustrates the calculation of PHP_A .

$$PHP_A = 1 - \frac{P_{moy}}{P_{max}} \tag{1.1}$$

This definition assumes that the downsized power source has a maximum output power equal to P_{moy} and is able to continuously provide P_{moy} over the mission. Also the power that must be provided by the downsized power source is equal to $(1 - PHP_A)$ times the maximum power that the non-hybrid power source was able to produce (i.e. P_{max}).

For reasons linked to power density and rarefied air operation, the main power sources of hybrid aircraft will likely remain gas turbines. Because the maximum available output power of a gas turbine evolves with altitude, the proposed formula is thus incompatible with our case study requiring a new definition presented later in this section.

The potential for hybridisation in energy PHE_A is defined to characterise the variation of the stored energy throughout the mission if the main power source is downsized as described in the definition of PHP_A . Since the downsized power source continuously provides P_{moy} in such case, the stored energy E_s at time t is then defined by Equation 1.2.

$$E_s(t) = \int_{t_i}^t [P_{moy}(\tau) - P_{mission}(\tau)] d\tau$$
(1.2)



Fig. 1.5 Two power profiles with same PHP_A but different values for E_u [5]

The useful energy E_u that is representative of the energy storage capacity is defined by Equation 1.3.

$$E_u = \max_{t_i \le t \le t_f} E_s(t) - \min_{t_i \le t \le t_f} E_s(t)$$

$$(1.3)$$

The definitions of E_s and E_u are illustrated in Figure 1.5 which compares two different missions having the same PHP_A .

The author defined the PHE_A as the ratio of the maximum power of the mission profile P_{max} to the useful energy E_u of the storage device (Equation 1.4).

$$PHE_A = \frac{P_{max}}{E_u} \tag{1.4}$$

 PHE_A is thus homogeneous to a frequency characterizing the variations of the stored energy. Also for a given P_{max} , it can be stated that the smaller the PHE_A , the larger the required capacity of the energy storage. The definition of the potential for hybridisation in energy is slightly modified in this thesis for a better characterisation of the storage unit.

The new definitions of potential for hybridisation in power and potential for hybridisation in energy proposed for analysing the power profile of the TP1 aircraft shown in Figure 1.3 are described next.

The definition of PHP_A assumes that the energy provided by the storage unit equals the energy it received over the mission. This assumption is used as a starting point for defining the new potential for hybridisation in power PHP_T . This energy management strategy is surely not best suited for hybrid aircraft operation but is convenient for the purpose of mission profile comparison. Let $P_{mission}$ be the required power for flying the mission (i.e. blue curve of Figure 1.3). Also, let $P_{max_{NH}}$ be the maximum power that can be continuously provided by the non-hybrid gas turbine (i.e. red curve of Figure 1.3). Finally, let k_{down} be the ratio of the maximum continuous available power of the downsized gas turbine to $P_{max_{NH}}$ under given flight conditions. k_{down} is assumed constant over the mission, the engine power lapse rates with altitude and speed being independent of the engine size to the first order. The stored energy E_s at time t is then defined by Equation 1.5.

$$E_s(t) = \int_{t_i}^t k_{down} P_{max_{NH}}(\tau) - P_{mission}(\tau) \,\mathrm{d}\tau \tag{1.5}$$

Based on the first statement of this paragraph, k_{down} must satisfy $E_s(t_f) = 0$ (i.e. the energy storage unit is recharged to its initial state), thus defining k_{down} such as written in Equation 1.6 where the mathematical operator $\langle . \rangle$ returns the timed-average value of the input function.

$$k_{down} = \frac{\langle P_{mission} \rangle}{\langle P_{max_{NH}} \rangle} \tag{1.6}$$

The potential for hybridisation in power PHP_T expressing the potential reduction in size of the non-hybrid gas turbine enabled by the use of an energy storage device is therefore taken equal to $1 - k_{down}$ as written in Equation 1.7.

$$PHP_T = 1 - \frac{\langle P_{mission} \rangle}{\langle P_{max_{NH}} \rangle} \tag{1.7}$$

Similarly to the definition of PHP_A , $PHP_T = 0$ expresses a mission profile that is strictly unsuitable for the downsizing of the main power source through hybridisation while $PHP_T = 1$ expresses a mission profile that firmly supports it. Also, if the maximum available output power $P_{max_{NH}}$ of the non-hybrid source of power is independent of time – typically the case for ground-based application – and is equal to the maximum value of $P_{mission}$, then Equation 1.7 is strictly identical to the definition of PHP_A .

The potential for hybridisation in energy PHE_T is defined as the ratio of the maximum power delivered or received by the energy storage unit during the mission to the useful energy E_u , such as written in Equation 1.8. The expression of the useful energy E_u in Equation 1.3 still holds but the stored energy $E_s(t)$ is to be calculated from Equation 1.5. PHE_T is still homogeneous to a frequency, which relates in this case to the inverse of a time constant that characterises the storage unit in terms of dynamic performance and can be placed in the Ragone chart accordingly (Figure 1.6).





Fig. 1.6 Ragone Plot: Energy Storage Characterisation

The gas turbine size reduction and the energy management strategy based on the definitions of PHP_T and PHE_T are illustrated in Figure 1.7. Note that in this application case, the maximum power sent or received by the energy storage unit is during a recharge phase (i.e. green arrows) while usual design practices use the discharge power for sizing the energy storage unit. Nonetheless, modifying the definitions of PHP_T and PHE_T accordingly would not change the outcomes of the mission profile comparison

presented below.



Fig. 1.7 Illustration of power flows based on PHP_T and PHE_T definitions

The two indicators are now used to characterize the power profile of different vehicles including the TP1 aircraft (Table 1.2). For cars, calculations are based on Common Artemis Driving Cycles [8] assuming a vehicle mass of 1400 kg, a reference area of 2.7 m^2 (frontal area), a drag coefficient of 0.25 and a 0.028 rolling friction coefficient. The motorway profile is shown on Figure 1.8. Results for trains are extracted from the study carried out in [6]. The power profile for the local service aplication, which refers to the transportation of freight between two cities that are within 40 km from each other, is shown in Figure 1.9. Mission profiles for ships are taken from [24] and the power required is calculated assuming a proportional increase with the cube of the ship speed (Figure 1.10).

The PHP_Ts of the regional aircraft and container ship are very low compared to the other applications. This can be explained by the fact that their conventional propulsion systems operate at a relatively high level throughout the mission versus its maximum capability. Therefore the potential reduction in size of the main power source is small. Obviously, the longer the cruise phase, the smaller the PHP_T . The time constant τ_{sto} of the regional aircraft falls into the battery category. However, the small PHE_T calculated for the regional aircraft also expresses a huge capacity need that might be critical for this application. Thus, based on this set of indicators, the



Fig. 1.8 ARTEMIS motorway speed profile (left) and resulting power profile (right) [8]



Fig. 1.9 Power profile for train – local service [6]



Fig. 1.10 Speed profile for Ro/Ro ferry [24] (left) and resulting power profile (right)

implementation of hybridisation to aircraft appears as a challenging task from the energy point of view.

	Car			Train			Shij	Aircraft	
	Urban	Rural	Motorway	Local	Switching	Urban	Containor	Ro/Ro	Regional
	Orban	road	$150 \mathrm{km/h}$	service	Switching	transport	Container	ferry	200 nm
PHP_T (%)	94	85	74	65	83	91	43	63	33
$PHE_T (mHz)$	66	30	12	3	29	20	n/a	n/a	0.22
$ au_{sto}$	15s	34s	$1.4 \mathrm{min}$	$5.8 \mathrm{min}$	35s	50s	n/a	n/a	$77 \mathrm{min}$

Table	1.2	Mission	profile	comparison

1.2.2 Energy recovering

While hybrid cars are now entering the market in increasing numbers, their interest is mainly limited to urban driving cycles characterized by its numerous stops and starts. Main savings come from the ability of the hybrid architecture to recover kinetic energy during deceleration. This section analyses potential energy savings of the TP1 aircraft through energy recovering at landing and during the descent phase.

Braking energy

In order to stop at landing an aircraft dissipates its kinetic energy through several braking systems: disc brakes, air brakes and thrust reversers. Calculating the maximum kinetic energy at landing of the TP1 aircraft using the maximum landing weight and the reference approach speed yields to 36 MJ or 0.19% of the total energy consumed during the nominal mission. Assuming a Lower Heating Value of 42.8 MJ/kg for kerosene and a gas turbine efficiency of 40%, 36 MJ is the energy that can be obtained by burning 2 kg of fuel (to be compared with the total fuel burn of Table 1.1). Also, as the landing phase lasts only few seconds, the energy recovering system should be able to withstand very high power flows leading to a heavy and complex device. Considering the small portion of energy that can be recovered and the additional complexity brought by an energy recovering system, the potential benefit of braking energy recovering will not be investigated further in this thesis.

Gravitational potential energy

With its cruise altitude of 20,000 ft the TP1 aircraft benefits from a 1,183 MJ (329 kWh) potential energy at the end of cruise (19,800 kg). That is, 6.3% of the energy consumed over the mission. Hybridisation could enable aircraft to convert gravitational potential energy during descent through windmilling propellers, store it in electric batteries, and use it during a later phase. The following study addresses this point

and tends to highlight the best energy management in descent.

Let D_{ac} be the drag force applied to the aircraft without its propeller blades, but including nacelles and spinners. In constant-speed cruise, the thrust generated by the propulsion system exactly compensates D_{ac} . For the purpose of this study, four different operating modes of the propellers in descent are defined:

- *Folded*: propeller blades are fully folded along or in the nacelles to reduce as far as possible the propeller drag in descent. The aircraft drag penalty associated to this propeller mode is assumed to be zero.
- Feathered: the blade pitch angle is increased to the point that the chord line of the blade is approximately parallel to the on-coming airflow reducing the propeller drag in comparison with uncontrolled windmilling. Still, a feathered propeller generates drag that is taken into consideration by scaling the drag coefficient of [15] with respect to the propeller diameter and the number of blades. In this propeller mode, both propellers of the TP1 aircraft are assumed to be feathered, and a drag penalty of 60 drag counts in the aircraft reference area⁵ is imparted to D_{ac} .
- Transparency: the propeller is rotating but produces neither drag nor thrust. The small amount of power required to operate this mode is taken into consideration using the propeller efficiency map (Figure 1.16) and optimizing the propeller speed as a function of flight conditions to minimize the required amount of shaft power along the curve $C_t = 0$ (thrust coefficient C_t defined in Equation E.6). No drag component is added to D_{ac} as for the folded mode.
- Wind turbine: the aircraft recovers some energy during the descent. The additional drag is defined proportionally to D_{ac} through the energy recovering coefficient k. The total aircraft drag is therefore $(1+k)D_{ac}$. The shaft power extracted from the windmilling propellers is calculated using the actuator disk theory [111], hence neglecting friction losses and blade tip losses.

Figure 1.11 illustrates the effect of the different propeller modes on the aircraft flight path. The distance covered in folded mode was chosen as the reference distance to destination from cruise altitude. Note that under the assumptions of this study, the transparency mode involves the same flight path as the folded mode. The feathered mode and the wind turbine propeller mode require a steeper descent slope due to the additional drag. As a consequence the distance covered in descent is less than

⁵i.e. the drag force $1/2\rho S_{ref}V^2 0.006$ is added to D_{ac}

the reference and the cruise flight must be extended to reach the destination. The comparison between propeller operating modes is performed at iso-time-to-descent which explains why the descent speeds are different. Nevertheless, the cruise part is flown at the design speed.



Fig. 1.11 Trajectory comparison for 10 min descent time

This study aims to compare the onboard energy consumption of the aircraft in descent depending on the propeller operating mode and the descent time. In this analysis the following assumptions are considered:

- The aircraft weight is supposed to be constant and equal to 19,800 kg during the descent.
- Five different descent times are considered ranging from 16 min to 5 min.
- The distance covered in folded mode for the 16 min descent time is chosen as the reference range to destination as the descent speed for this case approaches the speed of best lift-to-drag ratio.
- Descent speeds are adjusted to match the required descent time while the cruise speed is fixed to Mach 0.45.
- For the wind turbine mode, coefficient k was varied from 0.1 to 3.0 with a 0.1 step for each descent time as far as the stall speed was not reached. In the results (Figures 1.13, 1.14 and 1.15) only the case providing the minimum overall energy consumption is depicted.
- The energy consumption of non-propulsive systems is taken into account throughout the range to be covered by assuming a constant power consumption P_{sys} of 140 kW.
- The propeller efficiency for the cruise segment is set to 90%.

To go even further, additional assumptions are made on the efficiency of the power generation system depending on the flight segment:

- If the additional cruise segment is performed with a gas turbine type system, its efficiency is set to 40% which applies between the energy source and the propeller shaft but also between the energy source and the non-propulsive systems. For the descent segment, an efficiency of 10% is considered for this type of system to account for the low efficiency of gas turbines in idle (Section 1.3.2).
- If an electrical power generation system is used in place of the thermal system for the additional cruise and/or the descent a 90% efficiency is considered.

The power flows and possible efficiency combinations are illustrated in Figure 1.12.



Fig. 1.12 Possible power flows for cruise segment (left) and descent (right)

A short Scilab [35] script was developed for the purpose of solving the equations of flight mechanics in descent and evaluating the energy consumption. The aerodynamic forces acting on the aircraft featuring no propeller blades have been calculated from the drag polar of the TP1 aircraft provided by OCCAM (Section 1.1) and corrected according to the different propeller operating modes. Results of this trade study are depicted on Figures 1.13, 1.14 and 1.15. The onboard energy consumption is represented as a percentage of the gravitational potential energy between cruise altitude and 1,500ft (i.e. 1,093 MJ).

The energy management strategy of the TP1 aircraft can be analysed through the 10 minutes descent and the transparency mode of Figure 1.13. The propeller does not operate in a transparency mode in reality but the average thrust along the descent approaches zero. It can be shown that going to the feather mode would reduce the energy consumption by 20% for the same descent time. Of course this may entail a complex modification of the engine architecture as the mechanical off-takes have to be provided even if the propeller gearbox output shaft is stopped.


Fig. 1.13 Comparison of onboard energy consumption for full thermal power generation system – Cruise: Thermal / Descent: Thermal



Fig. 1.14 Comparison of onboard energy consumption for hybrid power generation system – Cruise: Thermal / Descent: Electric



Fig. 1.15 Comparison of onboard energy consumption for full electrical generation system – Cruise: Electric / Descent: Electric

The hybrid power generation system (Figure 1.14) brings significant reduction in overall energy consumption versus the full thermal system thanks to its high efficiency in descent. In particular, an 8.8 ratio in energy consumption can be shown between most energy efficient cases (i.e 123% for full thermal versus 14% for hybrid). Contrary to the full thermal power generation it is better to perform the descent in transparency mode rather that in feather mode: the energy cost for keeping the propellers rotating is much less than the energy required to fly the additional cruise segment when feathered propellers are used. As energy recovering is now assumed possible with this system, it can be noted that the lowest overall energy consumption with the wind turbine mode is performed with a 4% recovery of the gravitational potential energy only. Still, the transparency and folded modes are always more energy efficient than the wind turbine mode for descent time longer than 10 minutes. Note that the descent time is generally constrained by cabin repressurization limits and will likely never be shorter than 7 minutes in normal operation. If the folded mode cannot be implemented because of the complexity of designing a folding variable-pitch propeller, then the most energy efficient operation of this hybrid configuration is reached in transparency mode with a 16 min descent time (20% of reference energy consumed). This result is to be compared with the most energy efficient operation of the conventional aircraft (Figure 1.13) with unfolded propellers, consuming 150% of the reference energy in feathered mode for a 13 min descent time.

For the full electrical power generation system, the overall energy consumption with the wind turbine mode is generally better than the transparency mode as the energy cost for additional cruise is much less than with the previous hybrid system. The recovered energy also participates in lowering this cost even further. Nonetheless, the most efficient case in transparency mode (16 min) to cover the reference distance is still better than the most efficient case in wind turbine mode (16 min also).

This first analysis based on efficiency considerations shows that if possible, propellers should be folded during the descent whatever the power generation system. Also, the use of a hybrid power generation system could largely help reducing the energy consumption in descent (-60% for 10 min descent time, up to -130% for 16 min descent time versus full thermal power generation). With the high efficiency of such system in descent, the propeller (if not folded) should be operated in transparency mode rather than being feathered or being used as wind turbines for descent in normal operation. Finally, energy recovering through windmilling propellers is definitely not the most energy efficient way to use the gravitational potential energy of the aircraft if there is no constraint on the descent time: the aircraft should descend at the speed of best lift over drag with the propellers folded or operated in transparency mode.

Let us recall that this study assumed a constant aircraft weight, whatever the power generation system considered. Weight penalties imparted by new systems, in particular for the full electric one, will highly influence these results. This pointed out the importance of considering the overall aircraft design in energy management trade studies. Moreover, this study highlights the strong influence of the trajectory on the aircraft energy consumption.

1.3 Power management and efficiency

Tackling the problem of low efficiency in certain phases of the mission should lead to immediate fuel savings. This section focuses on propeller and gas turbine efficiency of the TP1 aircraft.

1.3.1 Propeller efficiency

Current turboprop aircraft are fitted with constant-speed propellers meaning that the propeller rotates at a specific speed during each phase of flight. Three different propeller speeds are usually defined, each of them being associated to take-off, cruise and idle operations, respectively. A hydraulic propeller pitch changing mechanism adjusts the blade pitch to keep the propeller rpm to the required value. The idle speed is generally never under 60% of the maximum propeller speed N_p due to the minimum frequency required by non-propulsive systems and other hydraulic power generation equipment connected to the gearbox. The propeller efficiency map of the TP1 aircraft shown in Figure 1.16 is derived from a typical efficiency map of [125] and rescaled according to the number of blades. The advance ratio J, power coefficient C_p , thrust coefficient C_t and propeller efficiency η are defined in Equations E.6, E.7, E.8 and 1.12, respectively. In these equations, V is the aircraft speed, ρ the air density, D the propeller diameter, n the number of rotation per seconde of the propeller, Pthe propeller shaft power and T the propeller thrust. More details about propeller modelling can also be found out in Appendix E.

$$J = \frac{V}{nD} \tag{1.9}$$

$$C_p = \frac{P}{\rho n^3 D^5} \tag{1.10}$$

$$C_t = \frac{T}{\rho n^2 D^4} \tag{1.11}$$

$$\eta = J \frac{C_t}{C_p} \tag{1.12}$$

The propeller efficiency of the TP1 aircraft is analysed using the operating points indicated in Table 1.3. The red dots of Figure 1.16 locate the propeller operating points in these conditions. To place them, an optimisation was performed in order to maximize the propeller efficiency by varying the propeller speed between $60\% N_p$ and $100\% N_p$ and the blade pitch angle to reach the given propeller thrust. The optimisation results are summarised in Table 1.4.

One can notice that three speed levels could be defined for the conventional use of the propeller (i.e. red dots): 100% N_p for TO and CL, 83% N_p for CR and 60% N_p for idle phases. Also, the propeller efficiencies in climb and in cruise are very good while the efficiency in taxi is almost three times less. TX-2p refers to taxiing with two propellers while TX-1p refers to the single propeller taxi case. Performing the



Fig. 1.16 Propeller performance map of the TP1 aircraft derived from [125] - Solidlines are constant blade pitch angle lines

Phase	Code	Altitude (ft)	Mach	Thrust (N)
Taxi	ΤХ	0	0.02	4,021
${\it Take-off}$	TO	1,500	0.18	46,000
Climb	CL	10,000	0.31	$25,\!300$
Cruise	CR	20,000	0.45	$13,\!900$
Descent	DSC	10,000	0.39	-300

Table 1.3 TP1 total thrust on nominal mission

taxi using one propeller instead of two slightly increases the efficiency from 31% to 35%. The main advantage of single propeller taxi on current aircraft is mainly due to the poor efficiency of the gas turbine in idle and only incidentally to the increase of propeller efficiency as shown in the next section. For both taxi cases the propeller rpm was driven by the optimizer to the lower bound 60% N_p . Using hybrid-electric systems during taxi should enable to release this minimum speed constraint.

A new set of calculations have been done for the taxi cases by removing the lower bound on propeller rpm. The new propeller efficiencies are located by the green dots in Figure 1.16. Enabling the propeller to rotate slower than $60\% N_p$ increases the

Code	Speed $(\% N_p)$	Pitch (°)	Efficiency (%)
ТО	100	26	78
CL	100	31	90
CR	83	26	91
TX-2p (red)	60	3	31
TX-1p (red)	60	7	35
TX-2p (green)	33	14	52
TX-1p (green)	46	13	41

Table 1.4 Propeller speed and blade pitch angle for optimum operation

propeller efficiency by at least 7% for single propeller taxiing (41% efficiency). It is also noticeable that the propeller efficiency is now higher when the thrust for taxi is equally shared between the propellers as it reaches 52%.

Changing the blade pitch angle of the TP1 aircraft propeller allows varying the propeller rotation speed so as to meet the highest possible efficiency under given flight conditions. Since the propeller speed is tied to that of the output shaft of the gas turbine through a fixed ratio gearbox, the propulsion chain efficiency is a compromise between propeller and gas turbine efficiency that is analysed in the next section. Nonetheless, the use of hybrid-electric systems to lower the minimum propeller speed should help improving the propeller efficiency in taxi.

1.3.2 Engine operating point

The single spool engine (Figure 1.17) is the simplest gas turbine that can be used for driving a load. In this configuration both the engine compressor and the load are driven by the turbine. Since the engine cannot operate if the load speed is zero, the output torque at zero output speed, also referred to as stall torque, is zero. Also, very little torque is available at low output speed. For this reason, this architecture is almost solely employed for power generation where the shaft must rotate at synchronous speed irrespective of power level.

For vehicle propulsion, the free power turbine engine (Figure 1.18) is preferred. Here the load is driven by a free power turbine separate from that driving the engine compressor. The compressor and turbine combination that provides the hot, high pressure gas that enters the free power turbine is generally referred to as gas generator. Unlike the single spool engine the stall torque is around 2 times that at full power and 100% speed. At part load, the compressor efficiency remains higher than for a single



Fig. 1.17 Single spool shaft power engine [138] – shown with cold end drive

spool turboshaft since the gas generator speed is not tied to that of the load. For a given power or gas generator speed the power turbine speed may vary over a wide range, depending on the absorption characteristics of the driven load. Figure 1.19a illustrates the variation of Specific Fuel Consumption⁶ (SFC) versus the shaft output power and power turbine speed. Note that this efficiency map has been built by using referred parameters [138] which allows us to use it independently of flight conditions. There also exist free power turbine engines with multi spool gas generators [55] resulting in different compressor designs. However, the variation of SFC is basically similar to that for a single spool free power turbine engine.



Fig. 1.18 Free power turbine engine [138] – shown with hot end drive

⁶ratio of fuel flow to shaft ouput power



Fig. 1.19 Typical performance charts of a free power turbine engine

The turboshaft efficiency of the TP1 aircraft can be analysed with Figure 1.19a. For a given turboshaft output power, there is an optimum free power turbine speed which minimises the SFC. As shown, a cube law operation for the output shaft speed coincides roughly with this ideal running line of best SFC. The evolution of SFC with output power along the cube law is plotted on Figure 1.19b. In order to get the best efficiency of the gas turbine, the output speed must be kept close to the ideal running line. The propeller speed being tied to that of the output shaft of the gas turbine, the optimum propeller operating points from Table 1.4 studied in Section 1.3.1 have been reported in Figure 1.19a accordingly. Despite the generic nature of these performance maps, thus approximating the performance characteristics of the TP1 aircraft engines, running the turboprop engine according to the optimum operating points of the propeller alone yields to gas turbine operating points approaching its ideal operating line, thus pointing out the good overall performance of the turbine engine and variable-pitch propeller combination but also the difficulty to drastically improve it.

Along the ideal running line, the gas turbine efficiency increases with the output power and reaches its maximum value at the design power (Figure 1.19). From there, the engine should be operated at the design power in cruise to benefit from the lowest SFC. However, the gas turbine size can be driven by take-off or climb capability requirements. For example, as the TP1 aircraft engines are sized by the time-to-climb constraint, the cruise power is approximately 14% less than the design power at the selected cruise level resulting in part load operation of the engines. This power gap is noticeable between the climb segment and the cruise of Figure 1.3. Climbing to a higher cruise altitude will reduce this gap and help operating the engine closer to its highest efficiency point. This highlights once again the importance of considering the trajectory during the aircraft design optimisation. An alternative would be to downsize the gas turbine for the selected cruise altitude and provide power boost to meet the limiting design requirements through an additional power generation system. Of course the effect of the additional weight of this secondary power generation is to be taken into account when evaluating overall cruise performance. Nonetheless, a quick first calculation can be made assuming a constant aircraft weight and combining the data of Figures 1.19 and 1.21. As presented more into details in Section 1.3.3, the gas turbine efficiency decreases when selecting a smaller gas turbine. By being 14%oversized for the cruise segment, the TP1 engine is 1% less efficient than at the design power according to Figure 1.19. A 14% smaller engine (2,150 take-off horsepower versus 2,500 take-off horsepower for the TP1 aircraft) has a 2.8% worst efficiency than

the TP1 engine at design power based on the PWC data regression of Figure 1.21. Considering the 1% penalty for the TP1 engine due the part load operation in cruise, the efficiency penalty for the downsized gas turbine at the selected cruise conditions falls to 1.7% versus the TP1 engine. Of course this result is based on approximated data and assumes that both maximum take-off and maximum climb ratings can be scaled in the same way. However it shows that the interest of engine downsizing for the TP1 aircraft is not straight forward in term of engine efficiency, not to mention that weight penalties have not been considered. More detail engine models will help concluding about the interest of engine downsizing in this thesis.

If the gas turbine efficiency is good during high power demand phases, the picture changes when it comes to idle operations. While the gas turbine power in idle is between 5% to 10% of the available power under same conditions, its efficiency can be 4 times poorer. Let us come back to the taxi case introduced in Section 1.3.1. For a 60% output speed (value linked to propeller rpm in taxi through fixed reduction gearbox), increasing the output power from 5% – the power required for two propeller taxi – to 10% – the power required for single propeller taxi – reduces the specific fuel consumption by a factor of 2, which confirms that the single engine taxi is preferred to reduce fuel burn.

This leads us to consider two options to decrease the energy consumption in low efficiency phases of a gas turbine: modify the number of prime movers and/or add secondary energy sources.

1.3.3 Number of prime movers and size effect

Some hybrid ship propulsion systems [77] are characterised by their high number of diesel generators (Figure 1.20). The number of running generators depends on the required power, ensuring that diesel generators operate at an optimum efficiency throughout the cycle. Optimizing the number of running prime movers is also investigated for twin-engine helicopters. Twin engines provide a higher level of safety versus single-engine helicopters but the propulsion system is therefore oversized and operates at part loads, resulting in higher fuel consumption. In order to improve the overall efficiency of twin-engine helicopters, turboshaft manufacturers are studying and developing more electric turboshafts [70] to enable the rotorcraft to operate on a single engine mode, the other turbine being kept in stand-by during cruise flight. This system should help reducing the fuel burn by 15%.



Fig. 1.20 Rolls Royce Hybrid Shaft Generator (HSG) System

While twin-engine helicopters are able to fly their nominal mission on one engine (mainly because hovering requires more power than the forward flight), the TP1 aircraft cannot achieve this. Indeed, flying at the one engine inoperative ceiling (10,000 ft) results in lower speed but above all in worse aerodynamic efficiency.

The chart of Figure 1.21 plots the SFC versus take-off power for different turboshaft engines from the database of Meier [83]. This database includes civil and military engines for different applications (airplanes, helicopters and drones). Even if the design requirements highly change from one application to another, the general trend shows a decrease of SFC with turbine power. This is even more obvious when focusing on Pratt & Whitney Canada engines equipping many turboprop aircraft.

Based on PWC data regression formula, using a single gas turbine of twice the power of the TP1 aircraft engines (5,000 shp versus 2,500 shp each for the TP1 engine) would therefore increase the engine efficiency by 12%. However, having a single turbine in nominal operation requires a backup system sized to cope with the failure of this turbine and capable of providing roughly half the nominal aircraft power to ensure same flight capabilities as the TP1 aircraft. Since gas turbines have much higher specific power (5 hp/lb ≈ 8.2 kW/kg [114]) than batteries or fuel cells (Section 1.5), the secondary power generation system will likely remain a gas turbine. Due to the additional weight of this emergency gas turbines of different size, such solutions have not been investigated in this thesis. Of course, using more than two gas turbines in nominal operation would reduce the overal efficiency of the prime movers versus the TP1 aircraft. For these reasons, the hybrid architectures presented in Section 2.2 feature two gas turbines of the same size.



Fig. 1.21 Effect of turboshaft size on Specific Fuel Consumption

1.3.4 Secondary energy source: start and stop

The start and stop function is now being implemented in most cars. In particular, to allow the internal combustion engine to be switched off at a stop, the air conditioning system had to be redesigned and is now powered by batteries. Batteries are also used in some hybrid ship architectures. The additional battery power is then available for slow speed transits in harbour (diesel engine is then turned-off) or for peak power load smoothing. As mentionned in the previous section, helicopter engine manufacturers are studying hybrid systems to enable twin-engine helicopters to operate on a single engine during cruise [70]. The other engine is kept under standby mode to ensure lowest idle fuel consumption and rapid re-activation. Assuming that a start and stop function can be mature and reliable enough for aircraft application, the descent could be flown with the gas turbines off as far as non-propulsive systems are provided by a secondary power generation system. In the same way, taxi phases could be performed on a full electric mode. Batteries or fuel cells are usual candidates as secondary power sources. Section 1.2.2 pointed out a potential benefit of 60% on energy consumption for descent phase. Under the same efficiency considerations (10%) for a gas turbine in idle, 90% for an electrical chain), the energy consumption in taxi could be decreased by 90% to which further improvements could be added with a low propeller speed in taxi (Section 1.3.1).

1.4 Improved aerodynamics

Driving wheels, propellers or fans electrically instead of mechanically provides great flexibility as far as physical installation is concerned. Electrical power is thus a key enabler for distributed propulsion. This section focuses on aerodynamic improvements for the TP1 aircraft enabled by new propeller or fan integrations and the associated concepts such as differential thrust, blown wing, boundary layer ingestion and wingtip propellers.

1.4.1 Differential thrust

The vertical tail plane of a conventional tube and wing aircraft is sized to give yaw stability and controllability. In particular the rudder must be able to counter the yaw moment generated by the asymmetric thrust in case of engine failure. The sizing conditions are generally at take-off, when the air speed is low and the remaining engines are at full power. But gliders also feature a vertical tail plane. This is because the vertical surface also must be able to provide enough Dutch Roll stability – a complex coupling between Yaw and Roll motions – and crosswind capability in order to align the aircraft with the runway prior to landing in such conditions.

The engine failure scenario usually requires a much bigger vertical tail plane than the Dutch Roll and crosswind cases. Since electrical distributed propulsion would limit the asymmetric thrust in case of a main component failure, possibly by reconfiguring the power distribution so as to ensure zero yaw moment, the vertical surface could be downsized resulting in less friction drag. In order to fully remove the aircraft fin the distributed propulsion system should be able to provide active yaw control, which appears possible with the fast dynamic response time of electrical components. Several notable distributed propulsion configurations, such as the Aurora LightningStrike (Figure 1.22), the Lilium Jet and the ONERA AMPERE [54], use multiple ducted fans to provide vehicle control.

In order to assess the maximum aerodynamic benefit of this surface reduction, the vertical tail plane of the TP1 aircraft was totally removed and the drag polar updated accordingly. The performance of this aircraft was analysed without accounting for any weight saving nor engine resizing versus the TP1 aircraft. The calculation with the OCCAM tool (Section 1.1) shows around 4.5% reduction in fuel burn. If such aircraft would exist, the controllability and stability of the yaw moment would entirely rely on the propulsion system and its capability of providing a minimum amount of differential thrust in case of component failure. Even if the potential energy saving is attractive,



Fig. 1.22 Aurora LightningStrike VTOL X-Plane

the right balance between vertical tail plane shrinkage and amount of differential thrust used is to be studied into further details against safety requirements. For these reasons the next chapters consider only the potential vertical surface reduction enabled by the relieve of the engine failure sizing scenario, Dutch roll stability and crosswind capability being ensured by the remaining fin area.

1.4.2 Blown wing

The wing area of an aircraft is generally sized by low speed performance requirements such as take-off field length and approach speed and depends on the choice of high lift devices. In particular for the TP1 aircraft, the resulting wing area is not optimized for cruise performance: the lift coefficient in cruise is smaller than the lift coefficient of best lift-to-drag ratio. The cruise is therefore not flown at best aerodynamic efficiency.

Turbofan aircraft can generally achieve their cruise at best lift-to-drag ratio by flying at higher altitudes. Increasing the TP1 cruise altitude would increase its aerodynamic efficiency, pointing out here again the influence of the trajectory on the overall aircraft performance. But cruising to much higher altitudes also requires more powerful and expensive engines. In light of the short range missions covered by the TP1 aircraft, the choice of the cruise altitude is a compromise between aircraft performance, engine cost but also passengers comfort since the cruise duration must be sufficiently long versus climb and descent phases.

At the selected TP1 cruise altitude (20,000 ft), having a smaller wing would therefore increase its aerodynamic efficiency. To do so, one can upgrade the high-lift systems by more complex devices. With the most complex unpowered high-lift devices, such as slats plus multiple slotted fowler flaps, the maximum lift coefficient reaches 3. But higher level of maximum lift capacity can be achieved with powered high-lift devices. These systems are many and varied [98] but one of them is already used to some extent on the TP1 aircraft: high lift devices blown by propeller slipstream. By blowing the wing with the two 3.93 m propellers, the dynamic pressure over the blown part increases resulting in a lift increment. Of course the larger the blown area, the higher the increase in lift coefficient. This technology was fully implemented on the Breguet 940 STOL^7 aircraft in the 1960s: four turboshaft engines drove a common shaft which in turn, drove four oversize propellers so as to blow the large slotted flaps. At first sight, electrical cables provides higher reliability and installation flexibility than shafts and gearboxes for the purpose of transferring power. The blown wing concept is therefore back on study with the NASA LEAPTech (Leading Edge Asynchronous Propeller Technology) project which tests a composite wing fitted with eighteen electric-powered propellers mounted along the leading edge. NASA first studies [127] show a maximum lift coefficient in the order of 5.

Neglecting any weight penalty imparted by distributed electric propulsion, the implementation of this technology on the reference aircraft should lead to the improvement of the aerodynamic efficiency in cruise coming from the better positioning of the cruise point along the drag polar but also from the reduction of the wetted area and finally from the increase of the aspect ratio if the wing is resized at iso-span. However, using multiple propellers along the leading edge in cruise also increases the local Reynolds number on the blown surface which, in turn, increases the friction drag. Also, the lift distribution over the wing is disturbed by the multiple propeller slipstreams which is to be taken into account during the wing shape optimisation. These effects may not fully counterbalance the benefits mentioned previously but are to be considered in the trade study.

One alternative solution, implemented on the NASA X-57 Maxwell demonstrator (Figure 1.23) [92][19][32], consists in folding the small leading edge-mounted propellers not required for cruise propulsion. Indeed, the twelve small electric motors are used during take-off and landing in order to enhance high-lift capacity while the two other bigger motors located at the wing tips operates during high speed phases. The design of this tight propulsion-wing integration along the leading edge is fed by the different

⁷Short Take-Off and Landing

LEAPTech test results. Also, several studies (Section 1.4.4) show that placing the cruise motors at the wing tips help reducing the aircraft induced drag.



Fig. 1.23 NASA X-57 Maxwell

As the lift gain of a blown wing increases with the thrust, there is no doubt that the blown wing will benefit the take-off field length target. However, concerning approach speed requirement of STOL aircraft (which is the design driver of the wing area of the TP1 aircraft), approaching the landing field with all propellers at full thrust would require very high and unrealistic angle of attack to ensure the lowest allowed approach speed. Still, STOL aircraft fitted with distributed electric propulsion systems such as the NASA X-57 can fully benefit from the blown-wing effect in approach and landing by using a particular propeller power management. As suggested by Stoll and Veble Mikic [128], the power sent to the small leading edge propellers could be that required to ensure the maximum lift coefficient with appropriate stall margin considerations and the two other bigger propellers could be used as needed to keep the -3° approach slope, be it by generating thrust or drag through an active control of the lift-induced drag (Section 1.4.4).

In order to assess the effect of a 10% increase in lift capability during approach on the overall energy consumption, the TP1 model in OCCAM (Section 1.1) was modified. The wing area was reduced by 10% at iso-span and the tail surfaces were redesigned at iso-volume coefficients. Then, the drag polar was updated by OCCAM accordingly but neither the characteristic weights nor the propulsion system were modified. As a result, the block fuel was reduced by 2.5% versus the reference TP1 aircraft. Of course, further fuel burn reduction may be achieved with a higher increase in lift capabilities.

1.4.3 Boundary layer ingestion

The theoretical benefit of boundary layer ingestion on propulsive efficiency has been known for several decades and is already implemented in marine propulsion. In the past, various research works have been carried out on the application of the concept to aircraft propulsion and it is now being studied even further as it could be a key concept to make commercial aircraft more energy efficient. The Aurora D8 [135] (Figure 1.24) or NASA STARC-ABL [140] (Figure 1.25) concepts are some recent examples.



Fig. 1.24 Aurora D8 commercial aircraft concept with Boudary Layer Ingestion

There are many different ways of explaining the boundary layer ingestion benefits. The classical explanation is that ingesting a flow with reduced velocity requires less power from the propulsor to create the same amount of thrust. Another view [135], is that it reduces the power dissipation in the overall flowfield through the reduction of wasted kinetic energy left by the aircraft by filling the aircraft wake with the propulsor outflow.

A preliminary assessment of boundary layer ingestion for the reference aircraft is presented hereafter. The method used is derived from [106] and is based on the actuator disk theory. This method may seem simplistic but provides the order of magnitude of potential benefit from a limited number of parameters.



Fig. 1.25 NASA STARC-ABL – Single-aisle Turboelectric Aircraft with an Aft Boundary-Layer propulsor



Fig. 1.26 Rear propeller fully ingesting the fuselage wake

In this study, a propeller is placed behind the fuselage and fully ingests the fuselage wake as in Figure 1.26. Let consider that the outlet fuselage control volume is sufficiently far downstream the fuselage for the pressure to be equal to that of the free-stream. Also let assume that iso-momentum mixing occurs within the fuselage wake to deal with a fuselage wake of uniform velocity v_{BL} . Then from momentum considerations, the fuselage drag D_F is given by:

$$D_F = m_{BL}^{\cdot} (v_{\infty} - v_{BL}) \tag{1.13}$$

In Equation 1.13, m_{BL} is the mass flow rate passing through section CC (Figure 1.26) in the boundary layer. The boundary layer thickness as well as the velocity profile along this section have to be known for the calculation of m_{BL} . As these data were not directly available, the boundary layer thickness was calculated thanks to the momentum theory within the boundary layer assuming no pressure gradient, which can be formulated as:

$$D_F = \rho 2\pi \int_0^\delta (R_{CC} + r) u(r) (v_\infty - u(r)) dr$$
 (1.14)

A usual 1/7 power law [137] velocity profile u(r) for turbulent boundary layer was chosen. Of course the assumption of zero-pressure-gradient boundary layer is not satisfied in reality and pressure distribution should be taken into account in the application of the momentum theory. Also, the 1/7 power law velocity profile assumes that the boundary layer at section CC is fully attached while flow separation probably occurs at the fuselage rear end. However these rough approximations provide a simple way to obtain a first estimation of the boundary layer thickness. Afterward, m_{BL} can be calculated by integrating the mass flow rate within the boundary layer at section CC.

For simplicity the fuselage wake is assumed to mix ideally at the propeller streamtube inlet providing a uniform flow of velocity v_m that is less than the freestream velocity but at least greater than v_{BL} as the propulsor fully ingests the fuselage wake. Applying conservation of momentum in the mixing region yields:

$$\dot{m}_{BL}(v_{\infty} - v_{BL}) = \dot{m}_p(v_{\infty} - v_m) = D_F$$
 (1.15)

Then, using the actuator disk theory inside the propeller control volume gives the following relations between propeller disk area S_p , propeller thrust T_p and flow properties:

$$T_p = \dot{m_p}(v_e - v_m) \tag{1.16}$$

$$\dot{m_p} = \rho S_p \frac{v_e + v_m}{2} \tag{1.17}$$

Finally, the kinetic power imparted to the flow by the propulsor is:

$$P_k = \dot{m_p} \frac{v_e^2 - v_m^2}{2} \tag{1.18}$$

Therefore, for a given fuselage drag and rear propeller thrust the power provided by the propeller to the flow can be calculated by solving Equations 1.15, 1.16 and 1.17. The power defined by Equation 1.18 assumes ideal efficiency of the propeller and does not account for friction losses and blade tip losses.

Coming back to the reference aircraft, the potential benefit of boundary layer ingestion was evaluated in cruise. In that phase the fuselage drag represents approximately 17% of the total aircraft drag. In the case when a rear propeller provides 17% of the total thrust, based on the outer control volume of Figure 1.26 it can be stated that v_E equals the free stream velocity and the fuselage wake is said fully filled. In order to assess the full benefit of boundary layer ingestion, the thrust sharing between nacelle-mounted propellers and the rear propeller was varied as well as the rear propeller disk loading. The power consumption of the nacelle-mounted propellers was calculated using actuator disk theory with ideal efficiency and their disk loading was kept constant and equal to the initial propeller loading.



Thrust split (% nacelle-mounted propulsors/% rear propulsor in BLI)

Fig. 1.27 BLI benefits as a function of thrust split ratios

The power consumption for the different thrust splits and rear propeller loading were calculated and compared to the power consumption of the non BLI configuration 100/0 (Figure 1.27). With the reference disk loading for the rear propeller, all BLI configurations show a benefit of roughly -0.75%. This benefit can be slightly improved by increasing the rear propeller loading for certain thrust combinations only. Analyzing the system of equations solved during this study, it can be seen that for any given rear propeller thrust that is smaller than two times the fuselage drag, the kinetic power provided by the rear propeller decreases as disk loading increases. But for any given rear propeller thrust that is higher that two times the fuselage drag, the inverse

is observed. This explains why it is interesting to reach higher rear propeller disk loading for thrust split configurations 83/17 and 75/25. However, for a given rear propeller thrust, increasing the propeller loading decreases the mass flow rate $\dot{m_p}$ thus approaching the lower bound m_{BL} that assures that Equation 1.15 is satisfied. This limit was reached with the rear propeller loading increased by 50%. As $\dot{m_{BL}}$ were calculated from rough assumptions, it may be possible to increase the propeller loading even more but the benefit of boundary layer ingestion for these two configurations would probably never exceed 1%. As the four other thrust split configurations require less power when the rear propeller loading decreases, the disk loading was reduced by 25% versus the reference. If configuration 0/100 shows best propulsive efficiency gain (100% of the thrust is provided by the rear propeller operating in BLI configuration), the associated propeller diameter is 6.5 m. As this propeller diameter raises integration issues, it can be concluded that the benefit of boundary layer ingestion for the reference aircraft would probably never exceed 1% whatever the thrust split ratio. In addition, it can be reminded that the calculation of the power consumption does not account for friction losses, blade tip losses, or unfavourable effect of the distorted propulsor inflow on the propeller performance.

As propellers already have relatively good propulsive efficiency in freestream, the benefit of boundary layer ingestion is negligible. The boundary layer ingestion concept is likely more suited to turbofan aircraft whose ducted fans operate with higher pressure ratios and have a lower propulsive efficiencies than propellers in freestream. Fitting the TP1 aircraft with a rear propeller may provide other benefits such as drag reduction coming from the decrease of nacelle size but also from the reduction of the vertical tail plane as less yaw moment is to be balanced in case of one engine failure. Nevertheless, those benefits are directly related to the propulsion system integration and not to the BLI concept itself. The rear propeller installation is therefore not considered in the architectures studied in the thesis.

1.4.4 Wingtip propellers

The generation of lift with finite-span wing inevitably produces lift induced drag. Due to the three dimensional nature of the flow around the finite-span wing, the difference in pressure between the upper and lower surfaces of the wing forces the air flow to curl around the wing tip resulting in wingtip vortices. The circulation around the wing tips increases the downwash velocity just behind the wing trailing edge – in particular near wing tips – without producing additional increase in lift. On the contrary, this increased downwash velocity lowers the effective angle of attack of the wing and thus

requires a physical increase in wing angle of attack to maintain the required total lift of the wing. This change in angle of attack will result in induced drag also referred to as vortex drag.

Winglets are commonly used on current aircraft to decrease the lift induced drag. This additional device at the wing tip extends the wing vertically without increasing its span. The vertical extension provides an increase of the effective aspect ratio of the wing [80] and therefore reduces the lift induced drag by further approaching the two-dimensional flow. Depending of the type of winglets, a 3-6% reduction of the total drag in cruise is usually achieved [17].

In the past, many wingtip devices have been studied: end plates [105], wing grids [65], spiroid wing tips [50], wing tip turbines [90] [48], wing tip blowing devices [68] and others. In particular, a wingtip propeller can be used as a wind turbine in order to harvest some energy from the vortex and attenuate its strength at the same time. This results in additional available power and a decrease of drag due to lift [95] [2].

But rather than adding some extra systems, can it be winglets or wingtip turbines for energy harvesting, several studies [96] [124] suggest to direct the high energy mass wake of a main aircraft powerplant into the tip vortex in order to interrupt the vortex core axial flow. This dissipates the vortex and results in a decrease in induced drag. In particular Snyder et al. [124] conducted a series of exploratory wind tunnel tests on a wingtip propeller in a tractor configuration. The results of the study confirmed the theoretical prediction that the use of a rotor turning in the direction opposite to that of the tip vortex produces a simultaneous lift increase and drag decrease. Rotating the wing tip propeller in the same direction to that of the tip vortex leads to opposite results. The rotational component of the propeller slipstream is then available for amplifying or attenuating the wing vortex system. In other words, wingtip propellers could be used as an active control of the lift over drag of the aircraft through the variation of the induced drag. This seems very attractive, as low values of induced drag are generally targeted for take-off and climb while approach and landing requires high values of induced drag.

Patterson Jr and Bartlett [93] [94] also conducted experimental studies on wingtip mounted propellers in a pusher configuration. Placing the propeller just behind the wingtip increases the propulsive efficiency of the propeller as a result of the influence of the wingtip vortex flow, and simultaneously attenuates the wingtip vortex by injection of the propeller wake into the vortex to reduce induced drag.

However, mounting all the propulsive power of the aircraft at the wing tips also raises possible disadvantages. While the wing of an aircraft with conventionally installed

powerplant near the center wing box is generally sized by positive load factor cases, a wing with tip-mounted powerplant would probably be sized by negative load factor cases (such as hard landing). Indeed, placing the engines and propellers at the wing tips will relieve the wing shear and bending moments under positive load factors but will increase them under negative accelerations. In addition, the wingtip powerplant installation highly changes the torsional moment of inertia of the wing which could result in aeroelastic problems due to unfavorable coupling of bending and torsional modes of flutter or vibration. For these two previous reasons, the structural design of the wing must be carefully considered in the evaluation of such aircraft configuration. Finally, having all the propulsive power installed at the wingtips can lead to impossible trimming of the aircraft in case of engine failure. In order to avoid this situation the propulsive power must be redistributed. Electrical power transmission provides a great flexibility for this purpose. Wingtip propellers are therefore considered in different hybrid aircraft concepts by NASA [19] (Figure 1.23) [9] (Figure 1.28). Borer et al. [19] calculated a 5-10% improvement of lift over drag in cruise coming from the wingtip propellers of the X-57. They also pointed out that this first estimation most probably under predict the effect due to the method used. Antcliff and Capristan [9] expect an increase in effective propulsive efficiency (which includes the induced drag reduction) of 18% for the wingtip propulsors on the PEGASUS concept.



Fig. 1.28 NASA PEGASUS concept [9]

At that time, no simple aerodynamic model can reflect the potential benefits of the wingtip propellers listed before with a relatively good level of accuracy. For this reason, placing the main propellers at the wing tips has not been studied in this thesis and will be considered in the prospects for future works.

1.5 Fuel replacement

In order to reduce aircraft environmental footprint, another solution consists in replacing kerosene (Jet A1) by a greener energy source. Electricity and Hydrogen, if produced from renewable resources, are interesting candidates. But air transportation requires high energies, megawatt range propulsion powers and lightweight vehicles making the transition more difficult than for other types of transport (e.g. road, railway, maritime).

Ground-based full electric vehicles use batteries to store energy from the electrical grid on-board. Today, the Lithium-Ion (Li-Ion) battery technology is widely selected for vehicle propulsion. This technology has interesting specific energy and power (150 Wh/kg at system level, 1C to 3C discharge rate), as well as good lifetime and level of safety. However, today's Li-Ion technology, and in particular the specific energy value, limits the full electric flight to small aircraft such as Pipistrel Taurus Electro, Airbus E-fan [61] or Siemens Extra 330LE. Rechargeable Lithium-Sulfur (Li-S) batteries are being increasingly studied due to their high theoretical specific energy, which is 3 to 4 times higher than that of Li-ion at cell level. Achieving the high energy density of these batteries requires challenging their high internal resistance, self-discharge and rapid capacity fading on cycling. Finally, metal-air batteries (such as Li-Air) have the highest theoretical specific energy compared to all other batteries but attaining high safety level and long cycle life seems critical. The cathode reactant (oxygen) is directly taken from the ambient air and does not need to be stored inside the battery therefore decreasing the battery weight. Finally, let us recall that designing a battery is always a trade between high specific energy and high specific power, not to mention the influence of the cycle life requirement. Table 1.5, derived from [43], provides a roadmap overview of energy-sized batteries for aircraft applications.

		Li-ion			Li-S		Li-Air	
		today	3 years	5 years	5 years	10 years	10 years	20 years
Specific Energy	cell	200-250	400	500	550	600	710	_
(Wh/kg)	system	140 - 170	280	350	300	400	280	500
Energy Density	cell	500-600	800	$1,\!000$	620	700	760	_
(Wh/L)	system	220-270	500	700	260	340	240	—

Table 1.5 High energy battery technology roadmap overview

For a first comparison with kerosene (Lower Heating Value of 43MJ/kg and density of 840 kg/m³ under standard conditions), let consider a gas turbine efficiency of 40%

and an electrical chain efficiency (from battery to electric motor output) of 90%. The volume of a 10-year-projection technology level Li-S battery providing 1 MJ at the electrical chain output is 13 times larger than the volume of kerosene providing the same energy at the gas turbine output. This battery is also 13 times heavier than the compared volume of kerosene. If reducing the battery weight is generally presented as the main challenge for full electric flight, the volume constraint is also to be carefully taken into account. Despite the good projected technology improvements, the full electric flight with batteries will most probably be limited to short range missions during the next 10 to 20 years. This point is treated more into details in the last chapter of the thesis.

With a Lower Heating Value of 120 MJ/kg, Hydrogen is a high-energy fuel. On top of challenges for delivering hydrogen to the airports, requiring new facilities and investments, the storage of hydrogen on-board remains the main concern.

High pressure gaseous hydrogen storage, primarily for its technical simplicity and fast filling-releasing rate, has become the most popular and mature method. Today's technologies are able to store gaseous hydrogen up to 70 MPa. However, even under such pressure level the volume of gaseous hydrogen storage will most probably not be acceptable for large and medium size hydrogen-powered commercial aircraft. Indeed, if the specific energy of hydrogen is 2.8 times that of kerosene, the volume of hydrogen in gaseous state (density of 42 kg/m^3 at 70 MPa (700 bar), standard temperature) containing 1 MJ of energy is 7.1 times larger than the volume of kerosene containing the same energy. This is already better than the volume ratio calculated with the advanced Li-S battery. Using liquid hydrogen storage (density of 71 kg/m³, 20 K. standard pressure) reduces this volume. However, if gaseous hydrogen can be stored at ambiant temperature, liquid hydrogen tank must be insulated to maintain cryogenic temperature (20 K). Assuming that 30% of hydrogen energy is lost in order to maintain hydrogen below its boiling point with a cooling system, the volume of liquid hydrogen containing 1 MJ of useful energy is 5.4 times larger than that of kerosene. In order to push research and development in hydrogen storage techniques, the US Department of Energy (DOE) has established some targets [31] (Table 1.6). Even if these figures concern light vehicles only (such as cars), the targeted specific energy of hydrogen storage systems exceeds by far that of advanced batteries. However, the power system for converting the hydrogen chemical energy is to be taken into account for a fair comparison.

	2020	2025	Ultimate
System Gravimetric Capacity (kWh/kg)	1.5	1.8	2.2
System Volumetric Capacity (kWh/L)	1.0	1.3	1.7

Table 1.6 DOE Hydrogen Storage System targets for Light-Duty Fuel Cell Vehicles

Hydrogen, whose combustion produces no carbon emissions, seems to be an excellent fuel for gas turbines. In 1937, Hans von Ohain's first jet engine demonstrator (HeS 1) used hydrogen as a fuel [82]. However, burning hydrogen so as to produce low NOx requires special combustion system technology [71]. A greener way of using hydrogen is to convert its chemical energy into electricity within a fuel cell, the waste products of the electromechanical reaction being water, heat, and very small amounts of nitrogen dioxide and other emissions depending on the fuel source. There exist different types of fuel cells, classified according to the nature of the electrolyte, the startup time and the operating temperature. But most research efforts are focused on polymer exchange membrane PEM fuel cells. PEM fuel cells have a low operating temperature (80-100°C), a low thermal inertia (insuring a short startup time) and a relatively mature technology level. The specific power of PEM fuel cell systems (including cooling device and auxiliary systems) is estimated to be around 1.3 kW/kg for near-term technology [114]. Assuming that the discharge rate of advanced energy-sized battery will stand between 3C to 5C, the specific power of PEM fuel cell systems is in the same order of that of batteries. Contrary to gas turbines, fuel cells show a decrease of efficiency with outgoing power. PEM fuel cells efficiency is generally 80% at minimum output power and decreases to 50% at maximum output power. Note that PEM fuel cell efficiency is also highly sensible to hydrogen impurity. While the PEM fuel cell efficiency at maximum output power is almost twice poorer than that of batteries (90%), the much higher specific energy of hydrogen storage favours fuel cell systems for the longest full electric flights if the propulsion system weight is the main challenge. Despite interesting advantages over batteries, the fuel cell system solution has not been studied in this thesis and only the full electric flight with batteries is addressed in the last chapter of the thesis. Still, analysing the feasibility of an hydrogen-powered regional aircraft is considered in the prospects for future works.

1.6 Conclusion

The first study of this chapter based on Potential for Hybridisation in Power and Potential for Hybridisation in Energy shows that energy saving potential brought by transient energy storage is much smaller for a regional aircraft than for ground-based transportation. Furthermore, various technologies enabled by electric power for the purpose of regional aircraft propulsion were analysed. Table 1.7 summarizes the benefits at technology level as well as the possible penalties that were pointed out in each study. Also, this table provides an estimation of associated overall energy savings for the 200 nm mission in the most optimistic case, by ignoring any penalty at aircraft level. The calculations are based on the TP1 aircraft fuel burn (Table 1.1). Boundary Layer Ingestion and Energy Recovering at landing show negligible benefits on the overall energy consumption that do not justify further investigations on these concepts for our hybrid aircraft. If the quick study on energy management in descent outlined poor interest regarding *Energy Recovering* in normal operation, the possibility of harvesting energy in flight is still considered in the next chapters so as to confirm these results. Also, even if the Engine Downsizing concept shows poorer energy efficiency than the TP1 aircraft at first sight, more detailed engine models are considered in the thesis to come to a conclusion. Despite the interesting figures announced by NASA with *Wingtip Propellers*, this concept is not studied in the thesis due to the lack of models. All other concepts show interesting potential energy savings and are studied more into details in the next chapters. Nevertheless, those benefits have to be put in contrast with weight and drag penalties caused by additional electrical components, energy storage devices and possible cooling systems. Also, this chapter pointed out the importance of the trajectory in the aircraft design. The optimisation of the hybrid aircraft and the evaluation of its performance require further trade studies that have to be performed with an overall aircraft design loop dedicated to hybrid aircraft. Results of this first chapter have been used for selecting the most promising hybrid architecture concepts for a regional aircraft and are presented in the next chapter.

Concept for fuel saving	Benefits	Max. Overall Energy saving	Potential penalties	Further studied in the thesis
Low propeller rpm in taxi	+49% propeller efficiency	-0.9%	Weight of electrical components Non-propulsive systems to be rethought	√
Secondary energy source	<u>Taxi</u> : -90% energy consumption <u>Descent</u> : -60% energy consumption	-2.4% -4.3%	Weight of secondary energy source Non-propulsive systems to be re- designed	√
Single engine aircraft	+12% prime mover efficiency	-8.9%	Weight of back-up system Acquisition and operating costs of two gas turbines of different size	Х
Engine downsizing	To be studied	n/a	Weight of secondary power source	√
Energy recovering	Landing: 0.2% block energy recoverable <u>Descent</u> : no benefit in normal operation	-0.2% n/a	Weight of energy harvesting system	X √
Differential thrust	Reduced skin friction drag	-4.5%	Reduced aircraft natural stabil- ity	√
Blown wing	Increased high speed aerodynamic efficiency	more than -2.5%	Drag of folded propellers Weight of powered high-lift sys- tem	\checkmark
Boundary layer ingestion	+1% propulsive efficiency	-0.8%	Rear propeller integration	Х
Wingtip propellers	+18% propulsive efficiency (PEGASUS)	-13.4%	Aircraft trimming with engine failure	Х
Fuel replacement	Decreased CO2 and NOx emissions	n/a	Weight of electrical power system	\checkmark

Table 1.7 Chapter 1 results overview

Chapter 2

Hybrid-electric regional aircraft configurations

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This chapter introduces the different aircraft configurations that are studied in more detail in the last chapters of the thesis. Each aircraft configuration is actually composed of a similar airframe and a specific propulsion system.

After describing the general airframe configuration, the conventional propulsion system that is used to optimise the reference aircraft is presented.

Then, the chapter focuses on electrified propulsion system architectures, and starts by addressing their classification. Since different technologies of electrical components can be used for the purpose of transmitting power from generators to propellers, a comparison of serial arrangements of these components is provided. Finally, the selection of electrified propulsion systems for this thesis is commented and the associated aircraft configurations are presented.

2.1 A common airframe geometry

This thesis focuses on the hybridisation of a 70-seat turboprop aircraft such as ATR72. All the studied aircraft configurations, can they be hybrid or conventional, are designed around a common airframe featuring a high-mounted wing and a T-tail. The dimensions of the airframe, illustrated in Figure 2.2, are mainly based on ATR72 geometry (Figure 2.1). In particular, the fuselage is the same as that of ATR72 with a fixed length of 27 meters and the wing span is limited to 27 meters. The wing is fitted with single slotted flaps, such as on ATR72, and the unflapped outboard portion of the wing is equipped with ailerons. The wing area but also the areas of the vertical and horizontal stabilisers, shown with red-outlines in Figure 2.2, are optimised in the aircraft design studies of this thesis. More details about the geometry assumptions are provided in Chapter 5.



Fig. 2.1 ATR72 geometry



Fig. 2.2 The common airframe—Red-outlined areas are variable

2.2 A reference propulsion system

The interest of regional aircraft hybridisation must be evaluated with respect to a reference conventional aircraft. As shown in Figure 2.3, this reference aircraft is equipped with two wing-mounted engine units, each of them including a turboshaft engine, a double-reduction double-branch power gearbox [142], a variable-pitch propeller and a nacelle. The reference aircraft is also named **REF** in the thesis.

The propulsion system of an aircraft provides the thrust required for flying but also the power for supplying non-propulsive systems. As an example, the power consumption of non-propulsive systems can be as high as 4 to 5% of the total installed power on a medium-sized conventional aircraft with propulsive power in the order of 40 MW [141] [88]. In order to account for the power consumption of non-propulsive systems in this thesis, a constant total power off-take of 140 kW is used and applied to the reference propulsion system but also to all the electrified propulsion systems that have been studied (Section 2.3.2). More details about the power off-takes positioning in the propulsion chains are provided in Chapters 5 and 6.



Fig. 2.3 Reference propulsion system REF

2.3 Different electrified propulsion systems

The usual classification of electrified aircraft propulsion systems is organised according to three main categories: All Electric, Turboelectric and Hybrid Electric. These categories can be further subdivided resulting in six different types of architecture as illustrated in Figure 2.4. The six types of architecture are commented hereafter and some examples of related studies conducted by industry, government and academia are provided.

All-electric architectures use batteries or fuel cells as the only source of energy for aircraft propulsion. As discussed in Chapter 1, today's battery technology limits the full electric flight to small aircraft (e.g. general aviation, commuter aircraft). Some prospective studies also focused on all-electric architectures for bigger aircraft such as the Bauhaus Luftfahrt Ce-Liner [56], an all electric single-aisle concept.

Turboelectric systems rely solely on fuel as energy source. **Full turboelectric** architectures commonly use turboshaft engines to drive electric generators and convert all the fuel energy into electricity to supply several electric-driven propulsors. This architecture is studied by NASA on the N3X [41] that mixes both blended wing body and turboelectric distributed propulsion concepts. Also, the ECO-150 airliner concept from ESAero [118] features a turboelectric distributed propulsion system embedded within the wing. **The partial turboelectric** system uses only part of the engine output power for electric propulsion and the rest drives a fan or a propeller mechanically connected to the engine. This propulsion system is explored by NASA on the STARC-ABL concept whose architecture consists of two under-wing turbofans with generators extracting power from the turbofan shafts and transmitting it electrically to a rear fuselage boundary-layer ingesting fan [140].



Fig. 2.4 Electrified propulsion architectures – modified from [40]

Hybrid electric systems use energy from fuel but also from one or several additional energy storage devices such as batteries. In particular, the gas turbines are available for propulsion but also to charge the batteries. These secondary power sources can also be operated for powering the aircraft during low energy phases in place of the gas turbines or providing power boost in order to improve the efficiency of the main engines resulting from new design considerations. In a **series hybrid** architecture, there is no mechanical links between the gas turbines and the fans or propellers. Similarly to the full turboelectric system the gas turbine drives exclusively a generator, but the electrical output power is then available to drive motors and/or charge batteries. The Airbus/Rolls-Royce single-aisle eConcept [113] is based on a series hybrid architecture composed of a single large turbine engine, a battery pack and six distributed ducted fans. In the **parallel hybrid** system, the gas turbine and the motor/generator are both mechanically connected to the same shaft that drives a propeller or a fan. The parallel hybrid uses both mechanical energy from the turbine and electrical energy from the batteries to drive the common output shaft. Several parallel hybrid architecture studies focuse on the single-aisle aircraft application. In particular, the Boeing SUGAR Volt [20] uses a parallel hybrid propulsion system to increase the cruise portion of the mission. Rolls-Royce North America also studied this type of architecture with the Electrically Variable Engine (EVE) technology [97]. These studies explore the hybrid engine design space with the objective to identify synergies between engine design and new energy management strategies. United Technologies Research Center also conducted studies on a Hybrid Geared Turbofan System [67]. The electric motor provides boost power during take-off and climb, resulting in a smaller core which increases the engine efficiency at cruise versus a conventional geared turbofan engine. Antcliff et al. [10] also carried out analyses of a 50-seat hybrid parallel regional aircraft. Finally, the **series/parallel partial hybrid** system is a combination of the previous hybrid electric systems. Some propellers or fans are driven directly by gas turbines while others are driven exclusively by electrical motors which can be powered by a battery or a turbine-driven generator.

The parallel hybrid architecture does not offer any of the aerodynamic improvements presented in Chapter 1. However, depending on the battery size it is most probably the lightest electrified architecture and the easiest to implement. All other architectures are well-suited to distributed propulsion which would enhance aerodynamic and propulsive efficiencies of the aircraft. Finally, "partial" systems benefit from lighter electrical components versus "full" architectures and require smaller technology advances beyond the state of the art to be implemented. The previous comments are further discussed in Chapter 6.

2.3.1 Comparison of electrical components

If long-term cryogenic electrical components such as superconducting machines and transmission lines, cryogenic power electronics and cryocoolers theoretically bring significant efficiency benefits over conventional components [22][21], most of them are at a relatively low technology readiness level. This section addresses the selection of electric components for the electrified propulsion system studies of this thesis which focuses on non-cryogenic technologies only.

Three different types of electric machines have been identified for the purpose of converting mechanical power into electrical power (i.e. generator function): Wound Rotor Synchronous Machine (WRSM), Permanent Magnet Synchronous Machine (PMSM) and Induction Machine (IM). A detailed description of these machines can be found out in [18] for example. Contrary to the other machines, a PMSM does not need electric power for creating the magnetic flux in the rotor resulting in higher efficiencies. If the IM is the most robust design of these three types of machines, the WRSM offers the possibility to control the output voltage through the variation of the rotor exciting current.

Transmitting the generator output power to an electrical load (e.g. motor) can be implemented with the help of an AC bus or a serial arangement comprising an inverter, a DC bus and a rectifier.

On the motor side, the PMSM and the IM are the most promising non superconducting technologies. In addition to the higher efficiency of the PMSM over the IM already quoted, the PMSM also features higher specific torque. With propeller rotation speeds in the order of 1,000 to 5,000 rpm, the PMSM would most probably enable the direct drive of the propellers and therefore to save both weight and efficiency loss from the gearbox. However, since the motor rotation speed has a strong influence on its design and, consequently, on its power density, combining a PMSM with a gearbox still appears as a relevant option. When used as a motor, the PMSM associated with power electronics offers an accurate control of the rotor position and is well suited for variable speed drive applications such as automotive propulsion or actuation systems. On the other hand, the IM is better suited for constant speed operation and can be directly connected to a fixed frequency network. The PMSM can also be used on a fixed frequency network without power electronics but requires an assisted-starting to reach synchronous speed. This assistance can be provided by an external starting device or by fitting the PMSM with a squirrel-cage winding for asynchronous self-starting. The resulted design provides higher efficiency in nominal operation than an equivalent induction motor but the power density could become lower.

Finally, a motor of the electrified propulsion system of a regional aircraft can drive a variable-pitch propeller or a fixed-pitch propeller. If controlling the blade pitch angle ensures a good propulsive efficiency throughout the flight (Section 1.3.1 and Figure 2.5), designing a foldable variable-pitch propeller seems hardly achievable with regard to both the blade mechanical link to the propeller shaft and the blade actuation system that should allow the blade to rotate about two orthogonal axes. In this sense, a high-lift propeller (Section 1.4.2) will most likely be a foldable fixed-pitch propeller. Figure 2.5 illustrates the evolution of propeller efficiency with aircraft speed for variable-pitch and fixed-pitch propellers.

All the components listed previously are reported in Table 2.1 which also shows the feasible combinations of them. All the nine serial arrangements of the first group


Fig. 2.5 Schematic comparison of variable-pitch and fixed-pitch propeller efficiencies

of this table feature an AC bus and a fixed-pitch propeller. The general shape of the performance map of a fixed-pitch propeller can be illustrated by considering a single solid-line of Figure 1.16 (i.e. a constant blade-pitch angle curve of the variable-pitch propeller performance map). Since the fixed-pitch propeller can only operate along such line, torque and rotation speed of the propeller input shaft are linked together. Whatever the type of electric motor, the propeller speed is tied to the electrical frequency of the AC bus. On the generator side, the constant voltage-to-frequency ratio and the field-oriented controls are the most widely used methods to control the induction machines and the synchronous machines, respectively. In both cases, the output voltage and electrical frequency are tied together. In other words, in this group of component arrangements the propeller torque is linked to the AC bus voltage. Driving the fixed-pitch propeller along its operating line then requires the voltage to be controlled accordingly, which can only be implemented with the WRSM. In this group, the combination of the WRSM and the PMSM for the motor still raises issues during the starting phase. Indeed, the assisted-starting of the PMSM to reach synchronous speed in such case may require high power levels with a fixed-pitch propeller, in particular if the AC bus frequency cannot be lowered enough. The resulting complexity of the additional starting device clearly disadvantages the PMSM over the IM.

The next nine component arrangements of Table 2.1 include an AC bus and a variable-pitch propeller. Of these nine configurations, only those featuring a synchronous generator (i.e WRSM or PMSM) are feasible. However, the use of synchronous generators without power electronics raises stability issues than can be dealt with an oversizing of the machine at first order. Indeed, as demonstrated by Fitzgerald et al.

[42] through a graphical approach, the stability of a synchronous generator subjected to a large load that is suddenly applied can be enforced if the load power is less than 72% of the maximum power capability of the generator. Having said that, the PMSM solution would still probably suffer from a narrower good efficiency domain versus the WRSM configuration because of the impossibility to control the AC bus voltage. In both cases, the power required for assisting the permanent magnet synchronous motor (PMSM) to start is expected to be less that for a fixed-pitch propeller of the same power capability since the blade pitch angle can be adjusted to minimise power absorption while spinning (i.e. low (high, respectively) pitch angle at low (high, respectively) speed). The combination of wounded rotor synchronous generators (WRSM) and permanent magnet synchronous motors (PMSM) has most probably been selected by Aurora on the LightningStrike VTOL X-Plane, an unmanned Vertical Take-Off and Landing Experimental Aircraft whose propulsion system is illustrated in Figure 2.6. The LightningStrike uses twenty four distributed variable-pitch ducted fans supplied by three Honeywell International 1 MW electric generators, all being driven by a single Rolls-Royce AE 1107C turboshaft. According to the shape of the electric motors shown in Figure 2.6, they are likely PMSM. The three configurations including an IM as generator cannot be implemented since no component is able to provide it with reactive power.

Despite the efficiency losses and major weight penalties imparted by the inverter and the rectifier versus the AC bus for the transmission of power, the use of power electronics favours the control and stability of the propulsion chain. In particular, all the last configurations of Table 2.1 comprising power electronics and a DC bus are feasible.

If the use of a direct drive or a geared drive on the motor side has been briefly commented before, a gear reducer may also be required at the generator input depending on the driving source. For a turbine-driven generator for example, the high rotation speed of the turbine output shaft (15,000 to 25,000 rpm) may favour the direct drive operation of high speed machines. All the architectures presented in the next section feature geared drives but the parametric gearbox model of this thesis, that can be found in Appendix F, can model both direct and geared drives with the objective to identify the need for using gear reducers for motors and generators.



Fig. 2.6 Full turboelectric propulsion system of the Aurora LightningStrike X-Plane – *modified from* [117]

Generator			Transr	Ν	Motor			Propeller			
WRSM^{a}	PMSM^b	IM^{c}	Rectifier+DC bus	+Inverter AC bus	IM+Gearbox	PMSM+Gearbox	PMSM	Fixed-pitch	Variable-pitch	Feasibility	Analysis criteria
•	•	•		• • • • •	•	•	•	• • • • •		✓ × × × × × × × × × × × × × × × × × × ×	Motor starting Torque-Voltage relation
•	•			•	•	•	•		• • • • •	<>><>>>>>>>>>>>>>>>>>>>>>>>>>>>>>>>>>>	Stability Efficiency
		•		•	•	٠	•		•	X X X	Reactive power supply
•	•	•			• • • •	• • • •	• • •	•	• • • • • • • • • • • • • • • • • • • •	~ ~ ~ ~ ~ ~ ~ ~ ~ ~ ~ ~ ~ ~ ~ ~ ~ ~ ~ ~	Power density & Efficiency

Table 2.1 Combination of components for electric power transmission

 $^{^{}a}$ Wound Rotor Synchronous Machine b Permanent Magnet Synchronous Machine

^cInduction Machine

2.3.2 Studied electrified propulsion systems

Four electrified architectures have been studied in this thesis: a parallel hybrid, a turboelectric, a partial turboelectric and an all electric. As shown in Table 2.2, the selection of these architectures is based on the concepts for fuel saving that have been identified in Chapter 1.

The accurate study of the *low propeller rpm in taxi* concept, that enables the propeller to operate at a slightly higher efficiency than on current turboprop aircraft, requires detailed analyses of non propulsive systems which are out of the scope of the thesis. For simplicity, this concept is supposed to be implemented on all the studied aircraft configurations, including on the reference **REF**, by allowing the propeller to operate at its optimum efficiency in taxi conditions.

The Start & Stop, the Engine downsizing and the Energy recovering concepts relate to new propulsion system energy management for which potential energy savings are independent at first order to those coming from the aerodynamic improvements brought by the *Differential thrust* and the *Blown wing* concepts. In this sense, the author focused on the study of energy savings enabled by new energy managements and improved aerodynamics separately. The identification of effective benefits from both sides would further justify the need to analyse architectures combining concepts for fuel saving related to new energy managements and improved aerodynamics. The parallel hybrid architecture is the simplest electrified propulsion system enabling all the Start & Stop, the Engine downsizing and the Energy recovering concepts to be implemented. The associated hybrid aircraft configuration considered in the thesis is named **PH**. The potential benefits brought by the *Differential thrust* and the *Blown* wing concepts are studied thanks to the the turboelectric and the partial turboelectric architectures. The aircraft featuring a turboelectric propulsion system is referred to as **TE4.** The rationale for considering a partial turboelectric propulsion system is that the weight penalties coming from the electrical components is expected to be less than for a full turboelectic. The aircraft configuration equipped with a partial turboelectric propulsion system is named **PT8**. Coupling the potential benefits identified from the parallel hybrid architecture and from one of the turboelectric or partial turboelectric architectures could be studied through series hybrid or series/parallel partial hybrid architectures.

Finally, the *Fuel replacement* concept is implemented in the all electric aircraft **AE** which features an all electric architecture. This aircraft configuration is used in the last chapter of this manuscript to identify the battery technology level that must be reached to enable the full electric flight of a 70-seat regional aircraft.

Aircraft configuration name	REF	\mathbf{PH}	TE4	PT8	AE
Type of propulsion architecture	Conventional	Parallel hybrid	Turboelectric	Partial turboelectric	All electric
Low propeller rpm in taxi	\checkmark	\checkmark	\checkmark	\checkmark	\checkmark
(Secondary energy source) Start & Stop		\checkmark			
Engine downsizing		\checkmark			
Energy recovering		\checkmark			\checkmark
Differential thrust			\checkmark	\checkmark	
Blown wing			\checkmark	\checkmark	
Fuel replacement					\checkmark

Table 2.2 Implemented concepts for fuel saving in the studied hybrid aircraft configurations

The electrified propulsion systems of Table 2.2 are detailed hereafter. Note that the relative size of the components illustrated in the next figures do not necessarily reflect that of the final design. Also, let us recall that a constant power consumption of 140 kW is considered for non-propulsive loads, which is not shown in these figures.

Parallel hybrid–PH

The parallel hybrid architecture studied in the thesis and shown in Figure 2.7 is composed of two identical wing-mounted propulsion units. Each of the units includes the same types of component as the reference propulsion system (Figure 2.3) plus a motor/generator tied to the power gearbox, a converter to control electrical power flows placed just behind the motor/generator, a battery located inside the wing and cables to connect the battery and the converter each other. Reducing cable lengths has been the the only reason for placing the converter inside the nacelle and the battery back in the wing. Despite the integration issues that may arise from the hot environment inside the nacelle, the models of the thesis let us evaluate the first-order effect of the increased volume and weight of the nacelle involved by the additional electrical components. In addition, installing the battery pack inside the wing may not be the best option if battery swapping strategy (i.e. loading a new full charged battery pack prior to each flight) is selected, not to mention the harsh thermal and pressure constraints seen by the battery throughout the mission. Still, it provides enough information to feed the aircraft models and evaluate the effect of the additional battery mass on the aircraft structural weights. Thanks to the converter, the electric machine can be either a PMSM or an IM.



Fig. 2.7 Hybrid parallel propulsion system

Full turboelectric-TE4

In the full turboelectric systems of this thesis, power is supplied by two turboshaft engines installed at each extremity of the fuselage (i.e. in the nose cone and in the rear end) as illustrated in Figure 2.8. The rationale for such configuration is to prevent the aircraft centre of gravity from being strongly shifted backward if both turbine engines are placed in the aft fuselage but also to avoid cascading failures in case of engine burst with side by side installation. Each turboshaft engine drives an in-line reducer that can be composed of several epicyclic gear trains in series, which in turn, drives a generator. The **TE4** aircraft configuration has four identical wing-mounted nacelles, each of them carrying a motor and an in-line gearbox driving a variable-pitch propeller. As illustrated in Figure 2.8, the power transmission from generators to motors is ensured by two independent AC buses. In this example, the rear generator supply the inboard propellers while the front generator supply the outboard motors, ensuring symmetrical thrust production in case of engine failure. After isolating the failed turbo-generator from its AC bus, the two buses could be connected together with switches so that the non-failed turbo-generator can supply all the motors, which could benefit the blown wing effect. According to Section 2.3.1, a generator of this propulsion system must be a WRSM while a motor can be either a PMSM with assisted-starting or an IM. These AC buses behave like flexible mechanical shafts without the need for any clutch, and



their combination with variable-pitch propellers even enable differential thrust yaw control such as implemented on the LightningStrike VTOL X-Plane (Figure 2.6).

Fig. 2.8 Full turboelectric propulsion system TE4

Partial turboelectric-PT8

The partial tuboelectric architecture of the **PT8** aircraft, shown in Figure 2.9, consists of two classical wing-mounted turboprop units (i.e. turboshaft engine, power gearbox, variable-pitch propeller and nacelle) equipped with generators extracting power from the power gearbox and transmitting it electrically to eight fixed-pitch high-lift propellers. The generator speed is therefore tied to that of the main variable-pitch propeller. Similarly to the NASA X-57 demonstrator [92], the high-lift propellers are used only during take-off and landing or emergency cases. During high speed phases the high-lift propellers can fold along their nacelles installed on the leading edge of the wing. In normal operation, each generator supplies the four high-lift propellers mounted on the same side of the aircraft. In case of failure of one engine, the associated generator can be isolated from the electrical network and the eight high-lift propellers can be connected all together thanks to switches, enabling the operated generator to supply the eight high-lift propellers and the thrust asymmetry to be reduced. The use of power electronics and DC buses to transmit power from generators to electric driven fixed-pitch propellers is to be chosen versus AC buses if an optimum control of the high-lift propeller rotation speeds is required. However, since the high-lift propellers will be commanded to provide maximum thrust when operating, they will rotate at their maximum allowable tip speed, and consequently, at constant rpm. Indeed, driving a fixed-pitch propeller at constant power results in a propeller rpm increase when flying faster and/or higher. Therefore, supplying the maximum power to a fixed-pitch propeller in order to produce the maximum thrust requires limiting its rotational speed according to the maximum allowable tip speed, which is usually given at sea level and limits the propeller rotation speed for all the flight conditions. Because of the constant speed operation of the high-lift propellers, the power transmission of the **PT8** aircraft is supposed to be ensured by AC buses which benefit from higher power density versus the other solution. Then, according to Table 2.1, a generator must be a WRSM and a motor must be an IM.



Fig. 2.9 Partial turboelectric propulsion system PT8

All-electric-AE

As shown in Figure 2.7, the all-electric aircraft **AE** features two nacelles, each of them carrying a converter, an electric motor and an in-line gearbox. The gearbox output shaft drives a variable pitch propeller and the battery pack has been arbitrarily located in place of the classical fuel tanks. Placing the battery in the fuselage would have only influence the cable length in this thesis, the models used for estimating the structural weights and the battery performances being independent of this assumption. Of course,

a detailed study of this architecture would require thermal and pressure constraints to be accounted for precisely, in particular for this battery pack location. The electric machine of this all-electric architecture can be either a PMSM or an IM.



Fig. 2.10 All electric propulsion system

Chapter 3

Multidisciplinary design optimisation for aircraft preliminary design study

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This chapter deals with deals with the aircraft design process addressed by the Overall Aircraft Design discipline, also referred to as OAD.

After presenting the constrained multidisciplinary optimisation problem that must be solved when designing an aircraft, a brief overview of the most suitable multidisciplinary optimisation methods is provided.

The second part focuses on the case of conventional aircraft drawing energy from a unique source, namely kerosene. A special attention is given to the mission performance evaluation during which the fuel burn is calculated, and to the iterative Mass-Performance Loop sub-process required to define the total structural weight. Typical constraints and optimisation criteria are also commented, and existing software for carrying out overall aircraft design studies are presented.

The last part addresses the case of hybrid-electric aircraft that can be designed with more than one type of energy storage device. Using energy from multiple sources involves additional variables in solving the design optimisation problem, requiring modifications of the classical design process, and in particular of the performance evaluation modules and of the Mass-Performance Loop. Modifications are proposed, including a new mathematical formulation of the latter. Finally, the most promising software for conducting the studies of this thesis are discussed.

3.1 Multidisciplinary design optimisation

Multidisciplinary Design Optimisation (MDO) is the field of engineering that uses numerical optimisation techniques to design systems involving multiple disciplines or components. This section introduces the particular case of aircraft design and addresses the different architectures that can be implemented for solving MDO problems.

3.1.1 Aircraft design: a multi-disciplinary optimisation problem

Aircraft design involves many disciplines such as propulsion, aerodynamics, structures, weight estimation, flight controls, manufacturing and many others. Managing the strong and numerous interactions between them is not an easy task. As an example, let us consider one of the primary functions of an aircraft: to fly.

As shown in Figure 3.1, level flight requires generating aerodynamic lift in order to compensate the aircraft weight which is mainly achieved by the wings. But the aircraft also generates some aerodynamic drag which has to be compensated by the thrust produced by the engines requiring fuel. Also, fuel, engines and wings need to be tied together with the help of structures directly impacting the aircraft weight. Wings,



Fig. 3.1 Balance of forces in steady level flight

engines and structure appear as the main components for flyability. Also, it emphasises that reducing both weight and drag is key to fly efficiently. Figure 3.2 better illustrates the complex interactions between these components.



Fig. 3.2 Interactions between components during aircraft design for the function of flying [102]

As caricatured in Figure 3.3, designing the ideal plane according to each discipline usually leads to incompatible solutions: requirements from a discipline can be constraints for an other one. Fortunately, the coupled design of two or more disciplines can also lead to more interesting solutions than the combination of individually designed solutions. This is in particular the case for propulsion and aerodynamics with more airframe-integrated propulsion concepts such as blown wing, boundary layer ingestion or wingtip propellers. Thus, aircraft design consists in finding out the best compromise between adversial optimum solutions from each discipline but also favoring synergies between them to further improve the final aircraft design.



Fig. 3.3 Caricature of ideal planes according to each discipline [102]

As a product, an aircraft is designed according to several requirements also referred to as Top Level Aircraft Requirements (TLARs). These requirements depend on the targeted market. Typical design requirements are:

- number of seats
- maximum range
- cruise speed
- take-off field length
- approach speed
- one engine inoperative ceiling
- time to climb and initial climb altitude
- ...

The TLARs of the TP1 aircraft designed in Chapter 1 are shown in Table 3.1.

From the TLARs, an inverse problem has to be solved: defining an aircraft that fulfills all the requirements. In addition to the selection of TLARs, other criteria are identified in order to make the product more attractive versus the competition. For instance, these criteria can be:

Value
70
$6,\!650~\mathrm{kg}$
$7,500 \mathrm{~kg}$
$400~\mathrm{nm}$
0.45
$17 \min$
$1,\!400~{\rm m}$
$113 \mathrm{kt}$
10,000 ft

Table 3.1 TP1 aircraft Top Level Aircraft Requirements

- fuel/energy consumption
- operating cost of the aircraft
- environmental impact
- a differentiation parameter (the biggest, the fastest, ...)

The objective in aircraft design is to optimise the defined aircraft according to one criterion or a combination of criteria. Finally, with the optimisation of an aircraft under several requirements (TLARs), the problem solved in aircraft design clearly defines a constrained multidisciplinary design optimisation problem.

3.1.2 Multidisciplinary Design Optimisation architectures

Martins and Lambe [72] provide a complete survey of architectures that can be used for solving multidisciplinary design optimisation problems. A brief overview of the most interesting architectures for the purpose of aircraft preliminary design is presented hereafter.

As illustrated in Figure 3.4, methods for solving MDO problems are divided into two categories: monolithic and distributed. Monolithic formulations, which include the multidisciplinary design feasible (MDF), the individual design feasible (IDF) and the simultaneous analysis and design (SAND) use a single optimiser for solving the MDO problem. On the contrary, distributed formulations such as Collaborative Optimisation (CO), Concurrent Subspace Optimisation (CSSO) or Bilevel Integrated System Synthesis (BLISS) decompose the MDO problem into a set of several smaller optimisation problems. A system-level optimiser is then responsible for managing the interactions between the smaller optimisation problems. The distributed formulations are particularly well suited during system development phases as they adapt well to an industrial setting where groups in charge of each discipline may work largely independently of one another based on design targets provided by the system-level optimiser. During system preliminary design phases, as the models involved in each discipline are usually simpler but also because the system optimisation is most of the time performed by one single group of designers, monolithic architectures are usually preferred over distributed methods because of their shorter optimisation time [131].



Fig. 3.4 Different MDO architectures

In a MDO problem, each discipline is responsible for solving a set of governing equations (e.g. equations of motion in a flight simulation, Navier-Stokes equations in fluid mechanics) to find the state variables from a set of inputs. The proper solving of the set of governing equations of a discipline, also referred to as discipline analysis, ensures the discipline feasibility. The set of inputs of a discipline is not only made of the MDO problem design variables¹, but also of variables related to the state of

 $^{^{1}}$ In this manuscript, the term *design variable* refers to a variable handled by the optimisation algorithm. In the literature, it is also known as *decision variable*.

other disciplines. The variables that are exchanged between the disciplines to model the interaction of the whole system are named coupling variables.

The multidisciplinary feasible (MDF) method uses a multidisciplinary analysis for evaluating the objective and constraints. With the help of a Gauss-Seidel iteration or gradient-based methods, the multidisciplinary analysis solves the governing equations for all disciplines until the coupling variables have converged. The MDF formulation enforces the multidisciplinary feasibility at each design point. The main advantage of MDF over the other monolithic architectures is that is offers the smallest optimisation problem size.

The individual discipline feasible (IDF) method only enforces discipline feasibility at each optimisation iteration by considering the coupling variables in the set of design variables. The multidisciplinary feasibility of the optimum solution is ensured by checking that the estimate of the coupling variables matches the actual coupling variables computed by each discipline through feasibility constraints added to the optimisation problem. Contrary to MDF, the governing equations of each discipline are solved only once per optimisation iteration resulting in much faster objective evaluation.

The simultaneous analysis and design (SAND) method relaxes the requirements of discipline feasibility at each design point enforced by the IDF formulation. The optimiser is then responsible for solving the governing equations of each discipline and the optimisation problem simultaneously. In addition to the coupling variables, the state variables for each discipline are considered in the set of design variables. Discipline feasibility of the optimum solution is ensured by using the residuals of the governing equations of each discipline as equality constraints in the optimisation problem. The state variables are used to compute the actual coupling variables and compared to the estimate of coupling variables through multidisciplinary feasibility constraints such as in the IDF method. SAND does not rely on any solver but requires the residuals of the governing equations to be computed which may be impossible with the use of commercial software. Finally, the resulting dimensionality of the optimisation problem can also be an issue.

MDF, IDF and SAND are therefore the most interesting architectures for aircraft preliminary design. It is also important to note that the benchmarking study carried out by Tedford and Martins [131] with different analytic problems shows that IDF and SAND outperformed MDF and other distributed formulations in term of computational efficiency. The common process used for the preliminary design of a conventional aircraft is presented in the next section.

3.2 Conventional aircraft design process

The aircraft design process starts with the identification of the TLARs and the criteria to be optimised (Section 3.1.1). In particular, as shown in the TLARs of Table 3.1, a design range must be selected. The combination of the design range and the design payload defines the design mission that is used to define the maximum structural weights of the aircraft and thus to size the structure (see Section 3.2.3). For a conventional aircraft it seems logical to choose the mission requiring the highest take-off weight which is usually the longest mission intended to be flown. As shown in Figure 3.5, the design range of Table 3.1 (i.e. 400 nm) is the longest range that must be covered by the TP1 aircraft. In view of the predicted flight occurrences of this figure, if airlines require the TP1 aircraft to be able to cover 400 nm, they are obviously willing to operate a more efficient aircraft regarding the optimisation criteria on a 200 nm mission rather than on the design mission. The missions used for evaluating the optimisation criteria are referred to as cost missions in this chapter.



Fig. 3.5 Predicted trip occurences for TP1 aircraft

The design variables for optimising the aircraft also have to be selected. As pointed out previously, wings, engines and structure are the most important components to guarantee flyability. On conventional aircraft, the wing area and the take-off thrust are commonly chosen as design variables for the wing and engine components, respectively. The Maximum Take-off Weight (MTOW), described more into details in Section 3.2.3, is one of the main parameters driving the design of the mechanical structure. This parameter is usually determined inside a sub-process called Mass-Performance Loop. The objective in OAD is to find the best values for the design variables (i.e. wing area and engine take-off thrust) so as to optimise the criteria and meet the TLARs. Of course, more design variables can be considered in this optimisation process, in particular variables related to engine ratings (max. climb, max. cruise, etc) [138] but also to aircraft operations such as climb and descent speeds. For simplicity, the design variables in this section are limited to the wing area and the engine take-off thrust already quoted. The following subsections describe the optimisation process into more details.

3.2.1 General work-flow

The classical aircraft design process work-flow is represented in Figure 3.6. The take-off thrust variable is first sent to the Propulsion module. This module designs an engine according to the take-off thrust requirement and returns its geometry as well as its performance characteristics. The other design variable (wing area) and the engine geometry then feed the Geometry module made of parametric models describing the entire shape of the aircraft. These models are updated according to the new inputs. The Geometry module then feeds both the Aerodynamics and Weights modules, the later being inside the Mass-Performance Loop. The Aerodynamics module generates all aerodynamic data, in particular drag polars, from the aircraft geometry. These data are then sent to the Evaluation of Design Mission module. The basic principle of mission evaluation is addressed in Section 3.2.2. The Weights module returns the mass of each component according to their geometry but also requires the mass of fuel loaded on the aircraft to fly the design mission for sizing the structure. This fuel mass is provided by the evaluation of the design mission. However, simulating the design mission also requires knowing the aircraft structural weights which justifies why both Weights and Evaluation of Design Mission modules are inside the Mass-Performance Loop. This particular sub-process is defined into more details in Section 3.2.3. After loop convergence, the aircraft is fully defined (i.e fixed geometry, aerodynamics, propulsion, and structural weights). The optimisation criteria are computed according to the evaluation of one or several cost missions and the constraints to be met (TLARs) are evaluated after analysing different scenarios including critical cases such as rejected take-off and one engine inoperative flight. The optimisation of the aircraft definition can be performed with a design of experiments in the case of a few design variables or with the help of an optimisation algorithm. Because the coupling between the structural weight and the aircraft performance along the design mission is solved at each optimisation iteration, the classical design process is a multidisciplinary design feasible (MDF) architecture (Section 3.1.2).



Fig. 3.6 Overall Aircraft Design Process

3.2.2 Mission performance evaluation

Mission performance evaluation is a major discipline of the OAD process, as the aircraft performance and in particular the fuel consumption must be evaluated along the design mission but also along one or several cost missions.

Aircraft performance is commonly divided into two sub-domains: High Speed and Low Speed. The Low Speed domain covers aircraft performance below 1,500 ft during take-off and landing sequences including normal and critical operations of the aircraft (e.g. engine failure, contaminated runway, etc). The High Speed domain deals with aircraft performance above this altitude in normal operations but also in case of enroute failures (e.g. engine failure, cabin pressurisation loss). More details about aircraft performance can be found out in [3]. Evaluating the aircraft fuel consumption and the fuel reserves for a given mission is one of the main task in High Speed performance studies and is described hereafter.

For the purpose of aircraft design and aircraft performance evaluation, different aircraft weights are defined:

- Manufacturer's Empty Weight (MEW): the weight of the structure, propulsion, systems and other items of equipment that are considered an integral part of the aircraft.
- **Operating Empty Weight (OEW):** the manufacturer's weight empty plus the operator's items, i.e. the flight and cabin crew, emergency equipment, galley structure, catering equipment, seats, documents and all fluids necessary for operation such as engine oil, engine coolant, water and unusable fuel.
- **Zero Fuel Weight (ZFW):** the weight obtained by addition of the total payload including cargo loads, passengers and baggages and the operating empty weight.
- Landing Weight (LW) : the weight at landing at the destination airport. It is equal to the Zero Fuel Weight plus the fuel reserves.
- **Take-Off Weight (TOW):** the weight at take-off at the departure airport. It is equal to the landing weight at destination plus the fuel needed for the trip.

Relations between aircraft weights are shown in Figure 3.7.

If one of the primary functions of an aircraft is to fly it also has to carry a payload over a range, thus composing the minimum data set for defining a mission. Let us consider a given aircraft for which the OEW, the aerodynamics and propulsion characteristics are known. Evaluating the trip fuel $(fuel_{trip})$ and and fuel reserves $(fuel_{reserves})$ for a mission consists in finding out the TOW of the aircraft satisfying Equation 3.1.

$$TOW = OEW + payload + fuel_{trip} + fuel_{reserves}$$
(3.1)

A rough approximation of the required fuel mass can be evaluated by solving the Breguet range equation [111] for $fuel_{trip}$ and defining $fuel_{reserves}$ as a function of mission range and $fuel_{trip}$. Higher accuracy results are obtained by solving the



Fig. 3.7 Aircraft weights

equations of flight mechanics at each point of the discretized mission. The effects of aircraft mass variation on aerodynamics and propulsion efficiencies are thus accurately accounted for and the total fuel consumption is calculated by integrating the fuel flow over time. This evaluation method is implemented in High Speed performance tools which also feature many algorithms for predicting properly fuel reserves based on airworthiness regulations. In any of these cases, computing the TOW requires evaluating $fuel_{trip}$ and $fuel_{reserves}$, which in turns, depend on the TOW. Figure 3.8 illustrates the computation of fuel burn in the case of mission discretization but also the algebraic loop that must be solved for the TOW which is usually performed by numerical root-finding methods. Note that more details about solving the equations of flight mechanics are provided in Chapter 4.

3.2.3 Mass-Performance loop

This section further describes the Mass-Performane Loop sub-process whose iterative aspect has been pointed out in Section 3.2.1: sizing the structure in the Weights module requires the fuel quantity computed during the evaluation of the design mission, which in turn, requires the structural weights from the Weights module.



Fig. 3.8 Iterative process for evaluating aircraft fuel burn over a discretized mission

The mass of the aircraft structure depends on the aircraft geometry but also on maximum structural weights. These weights, also referred to as aircraft characteristic weights, are defined as follows:

- Maximum Take-Off Weight (MTOW): The take-off weight (TOW) must never exceed the MTOW which is determined in accordance with structure and landing gear resistance criteria.
- Maximum Zero Fuel Weight (MZFW): Since wing root bending moments are maximum when the quantity of fuel in the wings is minimum, the MZFW limits the weight when there is no fuel in the tanks (ZFW).
- Maximum Landing Weight (MLW): The landing weight (LW) is limited by the MLW with regards to structure and landing gear resistance criteria during landing impact.

The weight models of the Weights module calculate the mass of each component from their geometry and the aircraft characteristic weights. The main task of the Weights module is to return the OEW. Since only the characteristic weights are unknown when entering the module for the first time, the OEW can be expressed as in Equation 3.2.

$$OEW = f_w(MTOW, MZFW, MLW) \tag{3.2}$$

In order to compute the OEW from Equation 3.2, three other equations are needed. As shown in Table 3.1, TLARs commonly provide a design payload $(payload_{design})$ and a maximum payload $(payload_{max})$. The design payload is used in the Evaluation of Design Mission module while the maximum payload must be carried by the aircraft, likely over shorter ranges than that of the design mission (i.e. the design range). The payload-range diagram, addressed later in this section is commonly used for representing the link between payload and range capabilities of an aircraft. From the definitions of MZFW and maximum payload, Equation 3.3 can be written.

$$MZFW = OEW + payload_{max} \tag{3.3}$$

By definition, the design mission simulated in the Evaluation of Design Mission module is the mission for which the TOW is equal to the MTOW of the aircraft. The objective of the Evaluation of Design Mission module is to compute the required quantity of fuel to perform the trip ($fuel_{trip_{design}}$) but also the fuel reserves ($fuel_{reserves_{design}}$) according to regulatory rules for a given payload ($payload_{design}$) and OEW such as in Equation 3.4.

$$MTOW = OEW + payload_{design} + fuel_{trip_{design}} + fuel_{reserves_{design}}$$
(3.4)

According to the definitions of MLW and aircraft's weights (Figure 3.7), MLW and MZFW are linked through Equation 3.5. In this equation, the fuel reserves for which the MLW is reached are usually expressed as a function of MZFW. One can therefore rewrite Equation 3.5 such as in Equation 3.6, where k is a coefficient depending on the aircraft category.

$$MLW = MZFW + fuel_{reserves_{MLW}}$$
(3.5)

$$MLW = MZFW(1+k) \tag{3.6}$$

Finally, combining Equations 3.2, 3.3, 3.4 and 3.6 yields to the system of equations 3.7.

$$OEW = f_w(MTOW, MZFW, MLW)$$

$$MZFW = OEW + payload_{max}$$

$$MTOW = OEW + payload_{design} + fuel_{trip_{design}} + fuel_{reserves_{design}}$$

$$MLW = MZFW(1+k)$$

$$(3.7)$$

Let us recall that $payload_{design}$ and $payload_{max}$ are given by the TLARs, k is a constant, and $fuel_{trip_{design}}$ and $fuel_{reserves_{design}}$ are returned by the evaluation of the design mission. Therefore solving the system 3.7 for the unknown OEW is the main objective of the Mass-Performance Loop sub-process. This is generally achieved by numerical root-finding algorithms such as Newton-Raphson's. It should be pointed out that the outputs of the Mass-Performance Loop, that are OEW and aircraft characteristic weights, fully depend on the fuel required for flying the design mission, and consequently on the choice of the design mission.

The payload-range diagram mentioned earlier plots the carrying payload capacity of one aircraft versus the mission range. It is a useful graph for comparing different aircraft and is most always used in competition studies. A classical payload-range diagram is plotted in Figure 3.9.



Fig. 3.9 A payload-range diagram

Let us consider an aircraft without payload nor fuel. This aircraft stands at the origin of the chart of Figure 3.9. At the upper left corner of the diagram the payload is maximum and is linked to the MZFW driven by Equation 3.3. However, this aircraft still needs fuel to fly a range. At maximum payload, adding fuel increases the flyable range but also increases the TOW. The fuel quantity can be increased until the TOW reaches the MTOW. At that point of the diagram, the range is equal to the maximum range that can be achieved at maximum payload. To further increase the range more fuel needs to be loaded. However, to avoid the TOW from being greater than the MTOW, some payload has to be removed from the aircraft. This part of the payload-range diagram is MTOW-limited. Finally, loading fuel indefinitely along this limit is not possible because of the finite volume of fuel tanks. Once fuel tanks are full, the only way to cover longer ranges is to reduce the payload and hence the take-off weight.

Actually, any set of payload and range inside the diagram is flyable with this aircraft. Let us recall that the design mission defined by a design payload and a design range drives the MTOW of the aircraft in the Mass-Performance Loop of the aircraft design process. Also, as shown in Section 3.2.3, the MZFW of the aircraft is computed in the Mass-Performance Loop module through several equations linked to the design mission. Consequently, the shape of the payload-range diagram is mainly driven by the design mission. This highlights once again the importance and the difficulty of selecting the design mission as it must guarantee that any mission intended to be flown by the aircraft stays inside the diagram.

3.2.4 Evaluation of the requirements

After the Mass-Performance Loop, the aircraft is fully defined and is able to fly the design mission. However, this aircraft still needs to satisfy all the TLARs so as to be considered as a viable solution. The requirements not covered by the previous steps of the aircraft design process are expressed as constraints in this optimisation problem. They commonly relate to operational performances but also safety. As a non-exhaustive list of examples, these requirements may deal with:

Time-To-Climb: the time to climb between two given altitudes shall be shorter than the value given in the TLARs. It is evaluated for a climb phase preceded by a take-off at MTOW. This requirement is mainly set for aircraft flying over short ranges. Indeed, it ensures that the cruise duration is sufficiently long versus climb and descent for passenger comfort. Note that the time-to-climb requirement may strongly constrain the engine size.

- **One Engine Inoperative ceiling:** in case of an engine failure during flight, the remaining thrust is no longer sufficient to fly at the initial altitude. The only solution is to descend to a more appropriate flight altitude. The maximum flyable altitude with one engine inoperative of the designed aircraft shall be more than the ceiling defined in the TLARs. This limit depends on the targeted routes and obstacles flown over (such as mountains).
- **Take-Off Field Length:** airport infrastructures also define operational constraints. In particular a regional aircraft shall be able to operate on isolated places equipped with very short runways. The TLARs usually set a maximum take-off field length evaluated at MTOW under specific environmental conditions that the aircraft shall not exceed. This calculation involves many different scenarios, in particular worst failure cases, and is usually evaluated with a dedicated Low Speed Performance Tool.
- **Approach speed:** the approach speed is a good indicator of aircraft performance at landing. A low approach speed yields a short landing distance and safer operations. The aircraft approach speed shall be less than the speed defined in the TLARs.
- Landing distance: the landing distance of the aircraft shall be shorter than the distance set in the TLARs. However the landing distance is most always shorter than the take-off distance. This requirement is sometimes not mentioned in the TLARs since the approach speed is already representative of landing performance.

3.2.5 Evaluation of the optimisation criteria

The optimisation of the aircraft design can be performed according to various criteria. Minimizing cost is of most interest for airlines. Cash Operating Cost (COC) and Direct Operating Cost (DOC) are commonly used in aircraft design. The COC includes all the costs associated with flying and direct maintenance: fuel costs, maintenance costs, crew costs, landing fees, navigation fees and ground handling charges. DOC includes the same items plus costs to insure and finance: capital costs and insurance costs. Due to the variation of capital costs with countries, it is easier to estimate COC than DOC. However, if the aircraft manufacturer experience can help in estimating such costs, the picture changes when evaluating long-term concepts and new technologies.

For conventional aircraft, MTOW is one of the main drivers of COC. In particular all fees and charges in COC are function of MTOW. Also increasing the MTOW requires more fuel and bigger engines to fly the design mission resulting in higher fuel burn and maintenance costs on all missions. This explains why minimising MTOW is commonly used as optimisation objective for conventional aircraft design. Of course minimizing MTOW can no longer be considered as a single optimisation criteria for hybrid aircraft due to the additional weights of electrical components and energy storage devices.

Minimizing aircraft emissions (NOx, CO2, ...) or aircraft environmental impact can also be of great interest in particular for future aircraft. At first order, reducing fuel consumption in favour of renewable energies can be targeted.

The optimisation criteria are evaluated according to one or several cost missions as already illustrated in Figure 3.6, under the condition that the cost mission falls inside the payload-range diagram of the defined aircraft. Also, a weighted objective function can be built according to a predicted flight distribution such as shown in Figure 3.5.

3.2.6 Overall aircraft design software

There exist many software to conduct preliminary aircraft design studies: AAA [112], ACSYNT [58], FLOPS [79], Pacelab APD [89], PASS [63], Piano [123], RDS-Professional [103], etc. These programming tools include built-in algorithms for the purpose of evaluating or optimising aircraft whose general work-flows are fairly similar to that presented in the previous sections. They obviously incorporate models and evaluation functions for the different disciplines mainly based on analytical formulae, also referred to as Level 0 models as defined hereafter. These tools are fast and can be very helpful in designing conventional aircraft as they also cover typical operational constraints from aircraft regulations. They also feature interactive graphic capabilities, the most elaborated tools offering simplified Geometry Computer Aided Design modules for example. Nonetheless, the scope of these conceptual design codes is primarily limited to conventional tube and wing aircraft with low coupling between aerodynamics and propulsion. Also, adding custom functions is most often impossible which bounds the user to the models already implemented in the code. Finally, aircraft manufacturers usually develop their own tools in order to include their experience in the design process.

In the case an overall aircraft design tool is developped, the evaluation of the aircraft requires models for the different disciplines whose granularity levels depend on accuracy needs versus allowable computation time. Three different levels of models can be defined:

- Level 0: this is the simplest level of models ensuring fast evaluations. The models include analytical formulae such as Torenbeek's [133], Roskam's [108] and Raymer's [104]. The aircraft definition is divided according to the main aircraft components (fuselage, wing, etc...). Uncertainty level on the provided outputs is relatively high but can be acceptable for early conceptual studies.
- Level 1: at this level the main components of the aircraft are further broken down into subcomponents (e.g. wing is seen as an assembly of wing box, movables, fixed leading edge and trailing edge panels, etc). This level relies on semi-empirical modelling and derivation methods from well known data. One discipline is generally treated by one tool which allows including an important part of the aircraft manufacturer knowledge for each of them. This level features the best compromise between accuracy and computation time.
- Level 2: this is the most accurate and refined level for which the aircraft is studied into details in the different disciplines. For example, structures are designed with Finite Element Analyses, Aerodynamics is evaluated with Computational Fluid Dynamics, etc...This level of model requiring intensive computing resources and running time is not used in aircraft preliminary design tools.

If level 1 models are chosen, the different tools associated to each discipline need to be linked together. Since handling the numerous variable flows can hardly be done manually, this task is automated inside Multidisciplinary Design Optimisation (MDO) frameworks such as ModelCenter [69], iSIGHT [27], Optimus [87], VisualDOC [13], etc. These software allow defining a series of tasks integrating cross-disciplinary models and applications with flows of parameters between them. They obviously include advanced functionalities for optimisation, design space exploration and post-processing. This thesis was initiated with the use of such MDO frameworks but new needs such as broader aircraft design space exploration and optimisation of hybrid aircraft caused the author to consider an other MDO platform as further detailed in the next section.

3.3 New needs for hybrid-electric aircraft tradespace analyses

The use of different types of energy storage devices on the aircraft involves additional variables in the aircraft weight breakdown. Also, while energy follows a unique path in the case of a conventional aircraft, from fuel tanks to gas turbines, the use of multiple energy sources increases the energy flow path options; not to mention that mixing energy

flows can be done in various proportions, bringing even more possible combinations. These numerous additional degrees of freedom pile up with the complexity of mission performance evaluations and Mass-Performance Loop solving. The next sections highlight the required changes in the OAD process to conduct hybrid aircraft analysis and optimisation.

3.3.1 Mission performance evaluation

Similarly to mission performance evaluation for conventional aircraft, one of the main tasks of mission performance evaluation in the case of hybrid aircraft is to evaluate the energy quantity required from each of the sources to perform the mission plus reserves.

As further illustrated in Figure 3.6, the mission evaluation of each of the cost missions of a conventional aircraft returns a fuel quantity that must be loaded on the aircraft to perform the mission of interest. In other words, the loaded fuel mass is adapted to energy needs required to fly the mission. But contrary to fuel such as kerosene or hydrogen, the battery weight of a hybrid aircraft will likely not be adapted prior to each flight. Sizing the battery mass is therefore not the task of the mission performance evaluation since the method used should be compatible with the performance evaluation of any mission and in particular in a multi-mission evaluation context. On the contrary the battery mass should be included in the known aircraft weights (such as the OEW) and the mission evaluation should return the temporal evolution of the exchanged power at the battery interfaces along the nominal mission but also along those required for calculating the reserves. For the purpose of this chapter the total weight of batteries equipping the aircraft is defined in the Propulsion weight of the aircraft (Figure 3.7), that is in the OEW, and is denoted $batt_{nom}$. Note that considering the battery weight outside the OEW does not change the outcomes of the following comments since it influences all the LW, ZFW and TOW in the same way. Let us recall that mission performance evaluation deals with the performance analysis of a defined aircraft whose OEW, aerodynamics and propulsion characteristics are known. In the case of hybrid aircraft, the selected energy management strategy for each of the energy sources along the mission should also be provided. To sum up, for a given hybrid aircraft (known OEW, aerodynamics and propulsion characteristics but also energy management strategy), the mission evaluation tool should be able to evaluate the trip fuel $(fuel_{trip})$, fuel reserves $(fuel_{reserves})$ and the battery power profiles $(P_{batt_{profiles}})$ for flying a mission (given range and payload) such that the TOW meets Equation 3.1. This formulation involves an algebraic loop for fuel quantities

only such as in the conventional case (Section 3.2.2). Nonetheless, $P_{batt_{profiles}}$ depends on the TOW which is expressed in Equation 3.8.

$$P_{batt_{modules}} = f_{ME}(OEW, payload, fuel_{trip}, fuel_{reserves})$$
(3.8)

The proper sizing of the battery should be checked after mission performance analysis by a Battery Sizing module that uses $(P_{batt_{profiles}})$ as input and returns the battery mass $batt_{req}$ that would result from sizing the battery according to $P_{batt_{profiles}}$ only. The evaluation of $batt_{req}$ by the Battery Sizing module is expressed in Equation 3.9.

$$batt_{req} = f_{BS}(P_{batt_{profiles}}) \tag{3.9}$$

The rationale for checking the battery size after mission performance relies on the fact that the same battery mass $batt_{nom}$ equipping the aircraft can be used to fly two or more missions of different payloads and ranges, such as several cost missions. As illustrated in Figure 3.10, the two evaluations of the Battery Sizing module enable to check that the battery equipping the aircraft is properly sized to perform both missions according to the selected energy management strategies. If the test condition is negative, the energy management strategies can be changed and/or the battery mass $batt_{nom}$ increased under the condition that all TOW, LW and ZFW are still below the aircraft characteristic weights. Increasing the aircraft characteristic weights would require re-sizing the structure pointing out the interest of considering the multi-mission mass-performance loop presented in the next section.

If fuel policies for conventional aircraft are regulated by minimum fuel requirements [3] [11], energy reserves for batteries are currently not addressed by aviation authorities. At first order, rules for defining fuel reserves can be extended to hybrid aircraft and included in $P_{batt_{profiles}}$. These rules are mainly based on simulations under nominal operation of the propulsion system (i.e. no failed components) of a continued cruise phase, an holding phase and a diversion mission occurring after the approach segment of the nominal mission. Sizing the battery capacity by considering these extended rules is probably not the hardest task since they only deal with nominal operation of the propulsion system. Aviation experience demonstrated that fuel quantities required during engine failure cases of a conventional aircraft are covered by fuel policies based on nominal operations, mainly because of the single nature of energy used. The picture can change when it comes to hybrid aircraft and multiple types of energy storage devices. In particular, if the battery is used during emergency cases, such emergency



Fig. 3.10 Interest of checking the battery size outside the mission performance evaluation

scenario can become the sizing mission for the battery capacity, requiring this mission to be simulated. The mission performance evaluation tool should therefore be able to simulate the hybrid aircraft performance in normal operations but also with enroute failures of new power sources. Identifying the worst case scenario for the battery sizing is of most importance but also a challenging task. Another difficulty comes from the fact that batteries can be recharged, the state of charge thus being not necessarily a decreasing function of time contrary to fuel quantity.

3.3.2 Towards a multi-mission Mass-Performance loop

This section analyses how the characteristic weights of a hybrid aircraft powered by different types of energy can be determined. The hybrid aircraft of interest uses energy from one or several types of fuel (total mass $fuel_{reserves} + fuel_{trip}$) and from

one or several batteries (total mass $batt_{nom}$). Since $batt_{nom}$ is included in the OEW, Equations 3.10, 3.11 and 3.12 for the aircraft's weights hold.

$$ZFW = OEW + payload \tag{3.10}$$

$$LW = OEW + payload + fuel_{reserves}$$
(3.11)

$$TOW = OEW + payload + fuel_{reserves} + fuel_{trip}$$
(3.12)

If the same mission is considered for designing the aircraft structural weights and the batteries, then the classical Mass-Performance Loop exposed in Section 3.2.3 can be used. Indeed, in this case the OEW is still a function of MZFW, MLW and MTOW but also of $batt_{nom}$, which is expressed in Equation 3.13. Since Equations 3.3, 3.4 and 3.6 still apply, the problem can be solved by adding Equation 3.14 expressing the proper sizing of the battery for this design mission, with $P_{batt_{design}}$ being the battery power profiles returned by the evaluation of the design mission. Note that coefficient kin Equation 3.6 may differ from the case of conventional aircraft sizing. In particular, for a full electric aircraft, k is zero. After replacing Equation 3.8 into Equation 3.14, the system of equations 3.15 is to be solved for OEW.

$$OEW = f_w(MTOW, MZFW, MLW, batt_{nom})$$
(3.13)

$$batt_{nom} = f_{BS}(P_{batt_{design}}) \tag{3.14}$$

$$\begin{cases}
OEW = f_w(MTOW, MZFW, MLW, batt_{nom}) \\
MZFW = OEW + payload_{max} \\
MTOW = OEW + payload_{design} + fuel_{trip_{design}} + fuel_{reserves_{design}} \\
MLW = MZFW(1+k) \\
batt_{nom} = f_{BS}(f_{ME}(OEW, payload, fuel_{trip_{design}}, fuel_{reserves_{design}}))
\end{cases}$$
(3.15)

The resulted payload-range diagram still depends on the choice of the design mission but also on the energy management strategy implemented during this mission. The numerous additional degrees of freedom introduced by the possibility of mixing different power flows further increase the risk of designing an aircraft not able to cover all the targeted missions including emergency cases and not optimised in term of energy storage capacities for the different missions of the flight distribution. To better illustrate this, let us consider a hybrid aircraft powered by batteries and equipped with a range extender such as a piston engine or a gas turbine. This aircraft shall be able to fly a nominal mission on a full electric mode and a longer mission using energy from both batteries and fuel. Limiting the Mass-Performance Loop to a single design mission forces us to select the longer mission since a higher TOW is expected in this case. From there, an energy management strategy is to be defined in order to compute both fuel consumption and battery capacities along the design mission. There is of course little chance that the resulted battery capacities answer the energy needs of the shorter mission. The aircraft characteristic weights will likely not be optimum either due to the wrong battery weight. Hence, including the shorter mission in the Mass-Performance Loop is needed. A generalised multi-mission Mass-Performance loop is presented hereafter.

Let us consider a set of N missions to conduct the sizing of batteries and aircraft characteristic weights. Starting from an hypothetical value for the OEW, running the mission evaluation for mission number i, defined by the set $(payload_i, range_i)$, returns the battery power profiles P_{batt_i} and the aircraft weights ZFW_i , LW_i and TOW_i defined by Equations 3.16, 3.17 and 3.18, respectively. Similarly to Figure 3.10, evaluating the proper sizing of the battery for mission i by running the Battery Sizing module with P_{batt_i} as input returns $batt_{req_i}$ such as in Equation 3.19.

$$ZFW_i = OEW + payload_i \tag{3.16}$$

$$LW_i = OEW + payload_i + fuel_{reserves_i}$$
(3.17)

$$TOW_i = OEW + payload_i + fuel_{reserves_i} + fuel_{trip_i}$$
(3.18)

$$batt_{reg_i} = f_{BS}(f_{ME}(OEW, payload_i, fuel_{trip_i}, fuel_{reserves_i}))$$
(3.19)

Then, the battery mass effectively included in the OEW, that is $batt_{nom}$, should be taken equal to the maximum value of $batt_{req_i}$ to be able to cover all the missions. In the same way, based on their definitions the aircraft characteristic weights should be taken equal to the maximum values of ZFW_i , LW_i and TOW_i so as to define MZFW, MLW and MTOW respectively. The new OEW can therefore be evaluated through Equation 3.13 and the process continued until convergence. The system of equations 3.20 is therefore to be solved.

$$OEW = f_w(MTOW, MZFW, MLW, batt_{nom})$$

$$MZFW = \max_i \{OEW + payload_i \}$$

$$MTOW = \max_i \{OEW + payload_i + fuel_{reserves_i} + fuel_{trip_i} \}$$

$$MLW = \max_i \{OEW + payload_i + fuel_{reserves_i} \}$$

$$batt_{nom} = \max_i \{f_{BS}(f_{ME}(OEW, payload_i, fuel_{trip_i}, fuel_{reserves_i}))\}$$

$$(3.20)$$

The use of numerical gradient-based methods for solving this system shows a high risk of non-convergence due to the definition of aircraft characteristics weights and total battery weight through the maximum function also commonly referred to as uniform norm. Another way of solving this multi-mission Mass-Performance Loop consists in formulating an optimisation problem in order to avoid using the uniform norm. This optimisation problem, illustrated in Figure 3.11, takes X = $(MTOW, MZFW, MLW, batt_{nom})$ as design variable to minimise the OEW under the constraints defined in Equation 3.21. Note that the energy management strategy also influence the evaluation of the different missions and is intentionally not included in the vector of design variables in this section for simplicity.

$$\begin{array}{ll}
\text{minimise} & OEW = f_w(X) \\
\text{subject to} & \begin{cases} ZFW_i \leq MZFW \\ TOW_i \leq MTOW \\ LW_i \leq MLW \\ batt_{reg_i} \leq batt_{nom} \end{cases} \quad (3.21)$$

This formulation increases the convergence robustness since it is now compatible with gradient-based algorithms. In addition it transforms the MDF formulation of the classical OAD process into an IDF formulation (Section 3.1.2) which is the MDO architecture selected in this thesis to optimise hybrid aircraft. In particular, as presented into more details in Chapter 4, the design variables shown in Figure 3.11 are moved further upward in the aircraft design process without compromising the proposed solution. On the contrary this multi-mission formulation tends to homogenise the resolution of the Mass-Performance coupling as it avoids to perform the full resolution


Fig. 3.11 Multi-Mission Mass-Performance Loop process

of the system of equations at each evaluation. It is also important to note that the minimisation of any objective function regarding aircraft efficiency (e.g. fuel burn, energy consumption, costs, etc...) forces the OEW to be minimised to ensure the proper sizing of the structure (i.e. not to carry oversized structural components). Finally, this generalised formulation can also be used in the case of conventional aircraft to ensure the best fitting of the payload-range diagram according to the missions included in the process.

3.3.3 Software adaptation

The exploration of hybrid aircraft concepts requires a flexible MDO platform capable to deal with different aircraft configurations, different propulsion systems, different energy management strategies and finally different flight trajectories.

In particular, the mission performance evaluation module shall be able to evaluate the energy consumption of the aircraft under any choice of energy storage device combination, energy management strategies and flight trajectories. Pornet et al. [99] proposed a methodology to quickly adapt classical mission performance evaluation tools through the definition of new propulsion system look-up tables. Yet, this methodology does not offer the required level of flexibility for the studies targeted in this thesis. In addition and in order to limit the optimisation time, the evaluation of a mission should be performed in a few seconds. Overloading classical mission performance evaluation tools with a dedicated hybrid propulsion module have also been investigated during the first part of the thesis and was finally abandoned because of the poor computational efficiency resulting from the inefficient data exchange throughout the mission evaluation.

For such studies, the different disciplines are likely to be modelled with Level 0 models, mainly because they involve short running times but also because of the difficulty to create accurate models for long term technologies from today's level of knowledge. In this sense, the MDO platform shall facilitate the use of analytical formulae. Nam et al. [85] published a generalised analytical power-based sizing method providing great flexibility in terms of propulsion system architecture exploration. Regarding the generalised equations considered and the assumptions made, this method is however best suited for early conceptual studies.

As mentioned earlier, this thesis started with the development of a design process for hybrid aircraft inside a commercial MDO framework such as ModelCenter [69] or iSIGHT [27]. The design processes that can be defined in such commercial MDO frameworks are inherently defined as a series of tasks with flows of parameters between them. The processes defined at computational level for one hybrid aircraft configuration cannot be used for a different one, thus not meeting the flexibility needs; not to mention that they have to be defined through the graphical user interface which is quite efficient to connect software together but does not favour the use of analytical formulae and scripts.

The OpenMDAO framework [84] [51] (Multidisciplinary Design, Analysis, and Optimisation) developed at the NASA Glenn Research Center would have probably met all these needs. It is an open-source platform for systems analysis and multidisciplinary optimisation written in Python. The framework works with both gradient-free and gradient-based optimisation methods and is most recognized for its automatic analytic multidisciplinary derivatives, which can be used to compute system-level gradients for Newton solvers and/or gradient-based optimisers enabling large problem solving with thousands of design variables. The OpenMDAO framework has already been used to solve many problems related to unconventional aircraft concepts like hybrid

electric propulsion [38], boundary layer ingestion for aircraft [119], and optimal aircraft trajectories [39].

Despite the high-performance computing capabilities of OpenMDAO and the Python-based implementation that would have provided enough flexibility for this thesis, an other MDO platform written in Python for studying unconventional aerospace concepts already existed as an in-house prototype at Airbus. This platform, called XMDO, was finally selected in order to carry out this research work and is described into details in the next chapter.

Chapter 4

The XMDO tool

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This chapter is dedicated to the Multidisciplinary Design Optimisation (MDO) platform that was eventually selected to carry out the studies of this thesis.

First, the origins and history of this tool, called XMDO, are introduced. After providing an overview of the tool framework, this chapter addresses the XMDO top level notions and methods that are used to model a vehicle and to evaluate its performance along a mission. The MDO architecture implemented in the tool is also discussed, and the tool optimisation features are presented.

The other sections focus on detailed implementations in XMDO: trajectory and control law modelling, aircraft equations of motion and solving, and generic propulsion system modelling. For each topic, the section begins with the presentation of the initial implementation in XMDO and follows with the main developments brought by the author of this manuscript: the selected trajectory formulations (Section 4.3.1), the new parametric control law for flap configuration (Section 4.3.2), the take-off and landing segment models (Section 4.4.3), and the improved generic propulsion system model (Section 4.5.2).

In the last section, the different constraints that are defined throughout the chapter for solving the global MDO problem are summarised.

4.1 Origins

XMDO is an Airbus proprietary tool that was created by Matthieu Meaux in 2011 and has been continuously developed by the creator and other contributors since then. Meaux's previous research works focused on aerodynamic shape optimisation of aircraft [81] [66] [86] [45] and addressed the problem of multi-point design optimisation [44]. The creation of XMDO was motivated by the need for studying disruptive aircraft configurations to answer new ecology and energy challenges. The ZEHST hypersonic transport airplane [28] was one of the first use cases of XMDO.

Although classical configurations such as tube and wing airplanes and single rotor helicopters dominate the market, their overall efficiency is reaching an asymptote, thus requiring new paradigms of aircraft and propulsion systems to be identified. However, while engineering knowledge of classical configurations allows organising the design process accordingly, the picture changes when it comes to designing disruptive concepts. As an example, the sizing procedure of a classical propulsion system usually considers the maximum power requirements at take-off and at the top-of-climb as inputs. When implementing new propulsion systems, such as electrified propulsion systems, the

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occurrence times of the worst case scenarios for sizing the different components of the propulsion chain can hardly be identified at first sight, not to mention their dependency on the selected energy management strategy. In this sense, the XMDO platform was developed to enhance the design space exploration of new aircraft concepts with low level of knowledge of sizing scenario occurrences. By removing the prediction of the sizing constraint activation times from the engineer tasks, XMDO also eases the exploration of the couplings between disciplines and the analysis of the results is a great source of knowledge to better understand the interactions. Such design approach is enabled by the use of an individual discipline feasible (IDF) problem formulation (according to the definitions of Section 3.1.2) that is explained in further detail in the next sections. To enlarge the design space, XMDO also uses a coupled optimisation of the vehicles and the trajectories. To go even further, the tool offers the possibility to define control laws (e.g. hybridisation ratio, rotation speed, flap deflection, etc) along the trajectories and to drive them with design variables. The selected MDO architecture also presents inherent opportunities to extend the evaluation of the designed vehicle to several missions, widening the design space even more. As flexibility is one of the key requirements of the XMDO implementation, it is based on object-oriented programming concepts for easy and fast reconfigurations during design space exploration analyses. The XMDO platform is written in Python and uses the scientific packages Scipy [60] and Numpy [139]. Finally, this tool dedicated to aircraft conceptual studies works with low and mid-fidelity models. The overall framework of XMDO is described in the next section.

4.2 Overall framework

The XMDO framework can be illustrated with the help of the three main objects handled by the platform: *Vehicle*, *PerfoEval* and *PerfoOptimWrapper*. As any object in object-oriented programming, each of them has its own attributes (that can be objects as well) and methods. The main relations between them are illustrated in Figure 4.1 and describe next.

- **Vehicle:** is the representation of the vehicle under study. It is made of different objects modelling the vehicle physics according to the following disciplines:
 - Geometry, dealing with the vehicle geometry
 - Mass, relating to the vehicle weight
 - Aerodynamics, covering aerodynamic forces acting on the vehicle



Fig. 4.1 Main objects handled by XMDO

- **Propulsion**, dealing with energy conversion, power transmission and thrust generation
- Energy, covering energy storage and consumption
- Thermal science, relating to thermal management
- Costs, dealing with costs estimation of the vehicle operation

Even if thrust force is presented as being part of the Propulsion discipline, the evaluation models of the generated thrust can also be placed in the Aerodynamics discipline. Indeed, as detailed in Section 4.4, a generalised Newton's second law is used such that mechanical actions can be modelled in either the Aerodynamics or the Propulsion, or both, enabling therefore the thrust-drag bookkeeping to be chosen by the vehicle designer. This provides great flexibility for the purpose of evaluating new airframe-integrated propulsion concepts such as blown wing or boundary layer ingestion.

Vehicle is also composed of methods to control the disciplines. These methods mainly relate to Newton's laws of motion but also to execution order of disciplines and data passing. More information about the vehicle description is provided in Section 4.2.1.

PerfoEval: comprises several Mission objects. Each Mission object includes a Vehicle object, several Segment objects describing the vehicle trajectory, a PostMissionAnalysis object containing the methods to post-process the mission evaluation (e.g. calculation of constraints and objectives), and methods for managing the vehicle performance evaluation along the trajectory. PerfoEval also contains methods to aggregate the post-treated outputs of the different missions. Since one vehicle is associated to one mission, it should be pointed out here, that XMDO enables to optimise not only a vehicle in a multi-mission context but also a vehicle family (Figure 4.2).



Fig. 4.2 Airbus A320 family

PerfoOptimWrapper: is composed of a **PerfoEval** object and different methods used to call the selected optimisation algorithm and adapt the **PerfoEval** outputs through penalty functions if required.

The next sections further describe the vehicle representation, the mission evaluation, the IDF formulation, and the optimisation features.

4.2.1 General vehicle description

The previous paragraph pointed out a vehicle representation according to disciplines, which is not the most obvious way of describing it. As illustrated in Figure 4.3, XMDO actually uses a cross representation of the vehicle: according to components and disciplines. Note that the Costs and Thermal Science disciplines are not shown in Figure 4.3 as they have not been used in the thesis.

From a standard user point of view, the vehicle is described according to physical parts called *Components* such as fuselage, wings, propulsion train, etc. The propulsion train component is further split into *Subcomponents* like propeller, gearbox, turboshaft,



Fig. 4.3 XMDO general vehicle description and modelling

battery, etc. XMDO uses this description to model the vehicle according to the different disciplines in the **Vehicle** object. To do so, *Components* and *Subcomponents* are modelled by discipline, meaning that for a fuselage component for example, there is a mass model, an aerodynamic model, a cost model, etc. The vehicle description also offer the possibility to associate a particular model of a discipline to a component or subcomponent in the case several models are available, or not to link any if the component or subcomponent of interest does not influence the discipline. As an example, in the classical thrust-drag bookkeeping, the fuselage does not transmit mechanical power for thrust generation and, consequently, does not affect the Propulsion discipline. The object representing a discipline at vehicle level includes all the objects modelling the same discipline at component level. Similarly, objects representing a discipline at subcomponent level are included in the object of the similar discipline at component

level. The rationale in this cross representation is that the outputs of a discipline at vehicle level can be seen as the sum of the outputs at component levels. For examples, the total mass of the vehicle can simply be defined as the sum of the component masses, or the total thrust at vehicle level as the sum of all the thrust forces generated by the components. Even if this simplified view may supposed so, interactions are not neglected in XMDO. On the contrary, since the object-oriented programming enables to access an object representing a discipline anywhere in the code, inside a discipline but also between them, interactions between components and disciplines can be modelled easily. This requires of course a proper execution order of the components but also of the disciplines as discussed in Sections 4.2.2 and 4.5. In particular, the Aerodynamics and Propulsion disciplines can be coupled efficiently to analyse new aircraft configurations featuring blown wing, boundary layer ingestion or wing-tip propeller concepts. In addition, this vehicle cross representation provides an efficient and flexible way to perform trade studies on vehicle configurations by changing only some components of the vehicle description without the need to create new global models, the vehicle disciplines being updated accordingly. As an example, studying a twin fuselage aircraft using the same physical models as a conventional tube and wing configuration only requires adding a second fuselage in the vehicle description. If particular interaction forces have to be considered, only the concerned component models have to be updated. The vehicle cross representation has been particularly useful during the thesis to carry out the comparison of different propulsion systems for a common regional aircraft application.

Finally, it is important to note that the Geometry is chosen as the piloting discipline of the aircraft characteristics. In other words, all other disciplines are expressed as a function of the Geometry, requiring the component models to be oriented accordingly. In XMDO, most of the design variables regarding the vehicle drive the Geometry discipline, which in turn, drives the other disciplines. As tangible examples, a more powerful turboshaft engine results from the increase of its length and/or diameter, a higher battery capacity is achieved by increasing its volume, etc. The Geometry is therefore the first discipline to be run during the performance evaluation as addressed in the next section.

4.2.2 Mission evaluation

This section focuses on the definition of a mission and the process implemented to evaluate the vehicle performances along it.

In XMDO, the concept of mission is developed to define a scenario of use for the vehicle under study (i.e. *Vehicle* object that is an attribute of the *Mission* object as shown in Figure 4.1). A mission is defined as a list of segments (i.e. different phases) along which the vehicle equations of motion are solved according to the temporal discretization of the segments. The high-speed segments (climb, cruise and descent) and the taxi segment differ from the take-off and landing segments by the way of solving the equations regarding the selection of inputs and outputs (Section 4.4). From the list of segments provided by the user, XMDO generates the **Segment** objects made of parametric trajectory models (Section 4.3.1) and specific methods for the evaluation of the vehicle performances. Contrary to the distinction made by classical overall aircraft design between high-speed and low-speed performances, the continuity of flight conditions between segments is ensured in XMDO. The rationale in this continuity is to avoid missing sizing constraints or design improvement opportunities when selecting only parts of the mission. Along each segment, it is also possible to specify parametric temporal control laws that may relate to the control of propulsion components (power level, hybridisation ratio), movable surfaces (deflection angle), variable-geometry components (wing tilt angle), or any item of interest. These control laws, modelled in the corresponding *Segment* object, have the same temporal discretization as the associated segment. Their complexity can be easily adapted to the study needs thanks to their mathematical formulation presented in Section 4.3.1. As mentioned previously, optimising the mission is also part of the design space exploration with XMDO. The parametric solution bring by XMDO and presented in Section A enables to include not only the vehicle description parameters as design variables but also those of the segments driving the trajectory and the control laws.

The general process implemented in XMDO for solving the vehicle equations of motion along high-speed and taxi segments is presented hereafter. The specific methods used during take-off and landing segments, that have been developed during this thesis, are presented in Section 4.4.

For high-speed and taxi segments, the operating altitude, the vehicle speed and flight path angle (i.e. angle between speed vector and horizon) as well as all their time derivatives are provided by the trajectory parametric models of the segments and thus considered as inputs. As illustrated in Figure 4.4, after execution of the Geometry, the Mass discipline is updated to account for the vehicle mass variation that may come from the fuel consumption calculated at the previous trajectory point by the Energy discipline. Then, a numerical solver based on Newton-Raphson's method is



(1) Flight Path Angle (2) Angle of Attack

Fig. 4.4 Process to solve vehicle equations of motion at a given trajectory point

used to solve the equations of motion by varying the vehicle angle of attack and the propulsion rating (i.e. throttle lever position, detailed in Section 4.5). One can notice that this process requires the Aerodynamics and Propulsion disciplines to be controlled according to the vehicle angle of attack and propulsion rating, respectively, that is further described in the next sections. Also, the angle of attack being constant during a taxi phase, only the propulsion rating is to be returned by the numerical solver in this case. For the evaluation of the system of equations, the Propulsion discipline is executed before the Aerodynamics in order to take into account the propulsive power effect on aerodynamic forces. After solver convergence, the Energy discipline is finally run to update the stored energy quantity. Running the disciplines in this particular order at each evaluation point provides great flexibility regarding the implementation of the interactions between Propulsion and Aerodynamics but also of the control laws. For examples, the defined process would enable to study the performance of a tilt-wing aircraft or a variable-sweep wing fighter, the Geometry being run first and the other disciplines evaluated accordingly. At each evaluation point, the updated data for each discipline (i.e. outputs of the evaluation) are saved in a log file.

Once the vehicle motion has been studied over each segment of the mission, the different methods of the **PostMissionAnalysis** object post-process the outputs of the segment evaluations and calculates the constraints and objectives of the optimisation problem with regard to this mission. More details about constraint and objective formulations are provided in Section A.

4.2.3 (X)MDO architecture

As illustrated in Figure 4.5 and according to the definitions of Section 3.1.2), XMDO uses an IDF formulation (Section 3.1.2) for solving the MDO problem of vehicle design. Similarly to the classical OAD process (Figure 3.6) the multi-mission evaluation of XMDO can include nominal missions from the predicted flight distribution but also special scenarios that can relate to missions along which particular vehicle components are failed (e.g. rejected take-off) or sequences requiring to be evaluated under particular conditions (e.g. landing at Maximum Landing Weight). A weighted objective can also be calculated from the performance evaluation outputs of the nominal missions to get an optimal vehicle (or vehicle family) regarding the trip distribution. The major difference with the classical MDF architecture (Figure 3.6) relies on the use of the multi-mission performance loop presented in Chapter 2 that is integrated in the optimisation problem by considering the aircraft structural weights (i.e. MTOW, MZFW and MLW) in the design variables and by checking the correct sizing of the structure with the additional constraints of Equations 4.1-4.3.

$$TOW \le MTOW$$
 (4.1)

$$ZFW \le MZFW \tag{4.2}$$

$$LW \le MLW \tag{4.3}$$

Also, to limit the number of iterations over the fuel quantity that must be loaded on the aircraft during a mission performance evaluation, XMDO considers the loaded fuel of each nominal mission in the design variables and checks that the fuel quantity $(fuel_{req})$ calculated by the mission evaluation is less than the fuel effectively loaded $(fuel_{in})$ at each design point such as expressed in Equations 4.4.

$$fuel_{req} \le fuel_{in} \tag{4.4}$$

While the different missions are evaluated sequentially in today's version of XMDO, the selected architecture is fully compatible with a parallel evaluation of the missions. The IDF formulation results in faster objective evaluation than the classical MDF method, in particular if the structural weights are converged (Figure 3.6) according to several missions in the MDF method.

As shown in Figure 4.5, a straight forward parallel can be drawn with the structure of the *PerfoEval* object (Figure 4.1). The methods of the *PerfoEval* object manage the mission evaluation order and aggregate the post-treated outputs of the different



Fig. 4.5 Illustration of the multi-mission optimisation with nominal and special scenarios

missions to form the objective function and the various constraints used by the optimiser of the *PerfoOptimWrapper* object.

More details (e.g. design variables, objective function, constraints, etc...) about the multi-mission optimisation process that has been implemented to carry out the aircraft studies of this thesis are provided in Chapter 5 and Chapter 6.

4.2.4 Optimisation features

For clarity, this section only provides a brief overview of XMDO's optimisation features. A more detailed description is given in Appendix A.

A fully parametric optimisation case definition

A case study is described with the help of configuration files that can be seen as templates used by XMDO to generate the objects handled by the code. A parameter file can also be associated with each configuration file. Once an object has been built from the configuration file by XMDO, the default attributes of the object are overloaded with the parameter file. Since these files are defined at different levels in XMDO (e.g. subcomponent, component, vehicle, segment, mission, etc...), they provide high flexibility for optimisation case definition. In the parameter file, a given parameter can be fixed to a constant value, defined as a function of other parameters (e.g. propeller span-wise position according to its diameter) or set as a design variable. The *PerfoOptimWrapper* object considers all the parameters defined as design variables in its *PerfoEval* object to perform the optimisation.

Optimisation methods

Since the design space exploration of unconventional vehicles may involve large numbers of design variables, XMDO was primarily developed to be used with gradient-based algorithms that are usually more efficient in such cases. In this tool, the derivatives for the construction of gradients or jacobians can be estimated with the Finite Differences or Complex Step methods (Appendix A).

As already pointed out in Chapter 3 and similarly to many engineering design optimisation problems, XMDO must solve a constrained optimisation problem. Not to mention that the constraints involves in vehicle design are most often non-linear. In addition, the selected optimisation algorithm in XMDO must be able to handle bounds on the design variables (i.e. constraints on the design variables) in particular to avoid failures of the design process due to improper vehicle geometry description or trajectory definitions. All of these restricts the selection of the optimisation algorithms. The first optimisation studies of XMDO have been performed with the NLopt [59] and Scipy [60] packages, two of the numerous available optimisation librairies in Python. Of the local gradient-based optimisation algorithms provided by these libraries and tested by the developer (see Appendix A), L-BFGS-B of Scipy combined with external penalties was one of the most successful in solving XMDO studies.

Because the *PerfoOptimWrapper* object was initially developed to handle different optimisation algorithms and different penalty methods, local and global derivativefree optimisation methods have also been tested during the thesis. The selected algorithms and penalty formulations are further discussed in Chapter 5.

4.3 Trajectories and control laws

As discussed previously, a mission is represented by a series of segments. Except for the particular cases of take-off and landing that are detailed in Section 4.4, the vehicle trajectory along each segment is defined with a B-spline curve. B-splines are widely used in geometrical modelling for Computer-Aided Design (CAD) but also in aircraft trajectory optimisation problems [29] [39]. These parametric curves, presented more in details in Appendix B, feature interesting properties, and, in particular, their first or higher order derivatives can be evaluated easily. Then, if the position of an object is described by a B-spline curve, speed an acceleration are known at any time of the trajectory. The different control laws associated to each segment are also modelled with such parametric curves.

XMDO uses clamped B-spline curves that are defined from physical input data as described below. Let t_i and t_f be the start and end times of the segment, respectively, and $\Delta t = t_f - t_i$ its duration. Based on the notations of Appendix B, let C be the clamped B-spline function from $[t_i, t_f]$ to \mathbb{E} , with $\mathbb{E} = \mathbb{R}^2$ for trajectories and $\mathbb{E} = \mathbb{R}$ for control laws. By default, XMDO generates uniform knot distributions (i.e. $t_i = u_0 = \ldots = u_p$ and $t_f = u_{m-p} = \ldots = u_m$) but a non-uniform knot vector has been used in the thesis for the particular case of the flap angle control as discussed in Section 4.3.2. Also, limiting the degree p to three (i.e. cubic B-splines) for $n \ge 4$ was found to provide enough flexibility in the previous studies carried out with XMDO.

4.3.1 Trajectories

General formulation

For the purpose of two-dimensional time trajectory modelling, let e_x and e_z be the standard basis vectors of \mathbb{R}^2 , with e_x aligned with the ground to measure distance and e_z pointing skyward to measure elevation. In this reference frame, let C(t) = (x(t), z(t)) and $C'(t) = (V_x(t), V_z(t))$ be the position and velocity of the vehicle at time t, respectively. Figure 4.6 illustrates the following description for the altitude coordinate.

The positions and velocities at the segment extremities are the first parameters required to describe a trajectory in XMDO and enable to compute P_0 , P_1 , P_{n-1} and P_n according to Equations 4.5, 4.6, 4.7 and 4.8 derived from Appendix B.

(

$$C(t_i) = P_0 \tag{4.5}$$



Fig. 4.6 Illustration of the B-spline definition for the altitude with n = 4

$$C(t_f) = P_n \tag{4.6}$$

$$C'(t_i) = Q_0 = \frac{p}{u_{p+1} - u_1} (P_1 - P_0)$$
(4.7)

$$C'(t_f) = Q_{n-1} = \frac{p}{u_{n+p} - u_n} (P_n - P_{n-1})$$
(4.8)

For a given set of parameters at the extremities, the segment duration Δt has also a strong influence on the vehicle trajectory. For this reason, an additional parameter dt_{fact} can be used to control the segment duration Δt . The relation between dt_{fact} and Δt that has been implemented in the thesis is presented next. To represent a large variety of trajectories with more degrees of freedom, the number of control points can be increased. An additional control point P_i is defined as a barycentre of P_0 and P_n with the set of parameters $(\alpha_{i,x}, \alpha_{i,z}) \in [0,1]^2$ such as written in Equation 4.9, where < .|. > is the scalar product.

$$j = x, z \quad < P_i | e_j > = \alpha_{i,j} < P_0 | e_j > + (1 - \alpha_{i,j}) < P_n | e_j >$$

$$(4.9)$$

The trajectory in XMDO can therefore be fully controlled with $x(t_i)$, $z(t_i)$, $x(t_f)$, $z(t_f)$, $V_x(t_i)$, $V_x(t_i)$, $V_x(t_f)$, $V_x(t_f)$, $V_z(t_f)$, dt_{fact} and $\{(\alpha_{i,x}, \alpha_{i,z}), i \in [\![2; n-2]\!]\}$. Because the continuity in position and velocity norm is ensured between the segments composing the mission, the number of design variables that must be handled by the algorithm to optimise the trajectory is reduced. The continuity in flight path angle is however relaxed in order to improve the flexibility of the trajectory control.

Selected parametric trajectories

As mentioned previously, a relation between dt_{fact} and Δt has been used in this thesis to control the segment duration. The duration Δt is expressed as a cubic function of $dt_{fact} \in [0,1]$ with the help of three characteristic durations Δt_{av} , Δt_{min} and Δt_{max} . The average duration Δt_{av} is the duration that would result from travelling a straight path between $C(t_i)$ and $C(t_f)$ at the average speed $\frac{1}{2}(C'(t_i) + C'(t_f))$. The minimum segment duration Δt_{min} has been chosen equal to $\Delta t_{av}/3$ while the maximum duration Δt_{max} is defined as $3\Delta t_{av}$. The coefficients of the cubic function are selected such that $\Delta t = \Delta t_{min}$ for $dt_{fact} = 0$, $\Delta t = \Delta t_{av}$ for $dt_{fact} = 0.5$ and $\Delta t = \Delta t_{max}$ for $dt_{fact} = 1$. The resulted range of duration $[\Delta t_{min}, \Delta t_{max}]$ was found to be sufficiently large for the chosen B-spline formulations. When dt_{fact} is not included in the design variables, 0.5 is used as a default value. Figure 4.7 shows the effect of the variation of dt_{fact} on the flight path and velocity during climb, the other parameters defining the trajectory being kept constant. One can notice that some values of dt_{fact} can lead to inappropriate trajectories for the vehicle under study, which will be easily handled by the optimisation algorithm thanks to the selected method for solving the equations of flight mechanics (Section 4.4). In these examples, the case $dt_{fact} = 0$ is unsuitable for subsonic aircraft while the trajectory described by $dt_{fact} = 1$ is not flyable by fixed wing aircraft as the Mach number can be zero. As illustrated, the parameter dt_{fact} enables to describe a wide variety of flight paths that is further enlarged with the modification of the control points.

Table 4.1 summarizes the B-spline formulations chosen in the thesis for each type of segment of the mission. The number of points N_d of the time discretization of a segment is also provided. A smaller N_d can be picked out for the descent phase than for the other high speed phases in order to reduce the computation time because shorter duration and lower energy consumption are expected during the descent. More degrees of freedom can of course be considered for modelling the vehicle trajectory but the formulations of Table 4.1 have provided suitable trajectory exploration capability from a limited number of design variables. As an example, Figure 4.8 shows a possible trajectory when the assumption $V_z = 0$ is released along the cruise phase.

4.3.2 Control laws

General formulation

While the n + 1 control points of the B-spline modelling a flight path are defined according to the positions and velocities at the segment extremities, they are provided



as a list in the case of control laws. By default, XMDO generates uniform B-splines of degree p from the list of control points, with p = n (i.e. Bézier curve) for n < 4and p = 3 otherwise. To allow for modelling discrete laws (e.g. state of power sources, component failure) the continuity of control laws between segments is not imposed by XMDO.

Selected parametric control laws

Except for the particular case of flap configuration controlling described hereafter, all the control laws of this thesis are represented by a constant value along each segment (i.e. p = 1 and $P_0 = P_1$) in order to limit the number of design variables.

The development of an appropriate control law for the flap deflection angle was motivated by the need to avoid improper performance evaluation between low speed and high speed phases. Indeed, since velocity continuity is enforced, an important change in flap deflection angle between two evaluation points could lead to stall.

The change in flap configurations has been included in the climb (descent, respectively) sequence with the help of a clamped B-spline of degree two, defined by four control points { $\delta_{TO}, 0, 0, 0$ } ({ $0, 0, 0, \delta_{LG}$ }, respectively) where δ_{TO} (δ_{LG} , respectively) is the flap deflection angle of the take-off (landing, respectively) configuration, and a non uniform knot vector $U = (t_i, t_i, t_i + u_{fact}\Delta t, t_f, t_f)$. While Δt is imposed by the segment

Segment	p	n	N_d	Specificity	9 7	
Taxi	1	2	5	$dt_{fact} = 0.5$ $V_z = 0$ $V_x(t_i) = V_x(t_f)$		
Climb	3	4	25		() april ()	\backslash
Cruise	3	4	25	$dt_{fact} = 0.5$ $V_z = 0$ $V_x(t_i) = V_x(t_f)$	Distance (km)	
Descent	3	4	20		0 200	400

Table 4.1 Selected trajectory formulations



along which the control law applies, the parameter $u_{fact} \in]0,1[$ is used to control the temporal evolution of the flap configuration such as illustrated in Figure 4.9 in descent. Note that such curves can also be produced with a uniform knot distribution by increasing the number of control points with zero value, but the optimisation algorithm should be able to handle integers in such case. In addition, smaller developments in XMDO have been required for driving the knot distribution.



Fig. 4.9 Effect of u_{fact} on flap deflection control law along a descent segment

This formulation enables not only to cope with stall issues when considering discrete values of flap deflection angles with the trajectory formulation of XMDO but also to evaluate the energy consumption during transition and low speed phases with a good level of fidelity relatively to the rest of the mission. In particular, the simple energy allowances on low speed phases of classical OAD workflows cannot be used any more since high power levels are expected during those phases with new aircraft concepts (e.g. blown wing) and energy consumption must be evaluated accordingly.

4.4 Solving aircraft dynamic equations

Two-dimensional aircraft trajectories are commonly studied relatively to a ground reference frame R_0 that is fixed to the surface of the earth with the *x*-axis aligned with the ground and the *z*-axis oriented upward. The body and aerodynamic reference frames defined such as in Figure 5.2 with the aircraft centre of gravity as origins are also used for the projection of forces acting on the aircraft. The next equations are derived by modelling the aircraft as a variable-mass particle.

4.4.1 Airborne

In flight, the forces acting on the aircraft are the weight W, the thrust T, the aerodynamic drag D and the aerodynamic lift L.



Fig. 4.10 Forces acting on the airplane in climb

Based on the notations of Figure 5.2, the speed components \dot{x} and \dot{z} in the ground axes system are linked to the speed magnitude V and the flight path angle γ through Equations 4.10 and 4.11.

$$\dot{x} = V \cos \gamma \tag{4.10}$$

$$\dot{z} = V \sin \gamma \tag{4.11}$$

Applying Newton's second law in the ground reference frame R_0 with a projection of forces in the aerodynamic reference frame yields to the dynamic Equations 4.12 and 4.13, where ϵ is the angle between the thrust vector and the aircraft longitudinal axis, and g the gravitational acceleration.

$$\dot{V} = \frac{g}{W} (T\cos(\alpha + \epsilon) - D - W\sin\gamma)$$
(4.12)

$$\dot{\gamma} = \frac{g}{WV} (T\sin(\alpha + \epsilon) + L - W\cos\gamma)$$
(4.13)

Finally, the rate of change of the aircraft weight can be directly linked to the fuel mass flow rate \dot{m}_f such as in Equation 4.14.

$$\dot{W} = -\dot{m}_f g \tag{4.14}$$

Equations 4.11 to 4.14 then constitute the system summarised in Equation 4.15 that must be solved when evaluating the aircraft motion in flight.

$$\begin{cases} \dot{x} = V \cos \gamma \\ \dot{z} = V \sin \gamma \\ \dot{V} = \frac{g}{W} (T \cos(\alpha + \epsilon) - D - W \sin \gamma) \\ \dot{\gamma} = \frac{g}{WV} (T \sin(\alpha + \epsilon) + L - W \cos \gamma) \\ \dot{W} = -\dot{m}_f g \end{cases}$$
(4.15)

The angle ϵ is assumed constant in this chapter since it only varies over time for vectored thrust aircraft (e.g. VTOL¹ aircraft). Also, the thrust *T* and the fuel mass flow rate m_f obey relationships shown in Equations 4.16 and 4.17, where χ is the power rating and can be seen as the position of the throttle lever.

$$T = T(z, V_a, \chi) \tag{4.16}$$

$$\dot{m_f} = \dot{m_f}(z, V_a, \chi) \tag{4.17}$$

For a given airplane geometry (including the configuration of the high-lift system), aerodynamic forces depend on altitude z, air speed V_a , angle of attack α and power rating χ due to the aero-propulsive coupling, such as written in Equations 4.18 and 4.19.

$$L = L(z, V_a, \alpha, \chi) \tag{4.18}$$

 $^{^{1}\}mathrm{Vertical}$ Take-Off and Landing

$$D = D(z, V_a, \alpha, \chi) \tag{4.19}$$

If the atmosphere is at rest (i.e. no wind component) such as in the thesis, V_a simply reduces to V, the aircraft speed relative to the ground reference frame R_0 .

The system in Equation 4.15 thus involves seven variables (i.e. $x, z, V, \gamma, W, \alpha$ and χ) and five equations. Providing two additional relations between existing variables or the time histories of any two of the seven variables then enables to solve the equations of motion. In practice, the two supplementary equations relate to the pilot orders on both the control column and the throttle lever that command the angle of attack α and the power rating χ , respectively. In classical high-speed performance tools used for the evaluation of aircraft fuel burn, the cruise part is a series of constant altitude and constant speed phases while the climb and descent phases are flown at constant engine rating and specific speeds provided by a speed profile. This choice of additional relations for the climb phase raises failure issues in aircraft optimisation problems solved with gradient-based algorithms. Indeed, the evaluation of the objective function or its derivatives is made impossible if the set of inputs selected by the optimiser does not allow the mission evaluation (e.g. undersized engine). To cope with this, XMDO uses x and z as supplementary inputs for solving the system in Equation 4.15 and requires continuous extrapolations of the propulsion and aerodynamic models. In particular the propulsion system must return increasing power values for $\chi > 1$, where $\chi = 1$ results in maximum available output power, and the aerodynamic lift model must return increasing lift values for $\alpha > \alpha_{max}$, where α_{max} is the stall angle of attack. This formulation enforces the existence of a solution for the system of equations, and the feasibility of the motion is ensured by adding the constraints expressed in Equations 4.20 and 4.21 to the optimisation problem.

$$\forall t, \quad \chi(t) \le 1 \tag{4.20}$$

$$\forall t, \quad \alpha(t) \le \alpha_{max} \tag{4.21}$$

In the current version of XMDO, α_{max} is a constant and must be provided by the user. It is therefore assumed independent of the aircraft speed, which does not reflect the real physics of aircraft stall which predicts a reduction of the stall angle of attack with higher Mach numbers [16]. This assumption seems however acceptable at first order since much smaller angles of attack are expected during high speed phases than during take-off and landing. The stall angle α_{max} is then also assumed independent of the high-lift system deflection angles which holds true in the thesis since only flaps

are considered. The aerodynamic models implemented for the thesis are presented in Chapter 5 .

The process shown in Figure 4.4 and used by XMDO for solving the equations of motion can be commented more into details here. Since the set (x,z) is provided under the form of a B-spline curve, the analytical formulations of V and \dot{V} are perfectly known and are used to compute γ and $\dot{\gamma}$ thanks to the kinematic Equations 4.10 and 4.11. The execution of the Mass discipline updates the aircraft weight according to Equation 4.14 assuming a constant fuel mass flow rate between two calculation points. Then, χ and α are found by solving the dynamic Equations 4.12 and 4.13 with a Newton-Raphson solver.

Since XMDO enables not only to design airplanes but also VTOL vehicles such as helicopters, multi-rotors or tilt-wing aircraft, equations 4.12 and 4.13 are actually replaced by a generalised Newton's second law written in Equation 4.22. In the latest solution proposed by XMDO, a force $\vec{F_i}$ is not explicitly linked to a discipline and can be computed at component or subcomponent levels, which provides great flexibility for the control of force orientations but also for the definition of propulsion systems and custom thrust-drag bookkeepings.

$$(W/g)\vec{a}_{G/R_0} = \sum_i \vec{F_i}$$
 (4.22)

In the particular case of two-dimensional motions of airplanes modelled by point particles, such as described before, the Newton-Raphson solver finds χ and α to satisfy Equation 4.22.

4.4.2 Ground run

During a ground run, the forces acting on the aircraft are the usual forces involved in flight (i.e. weight W, thrust T, drag D and lift L) plus the landing gear reaction force R and the rolling ground resistance equals to $\mu_g R$ where μ_g is the rolling ground friction coefficient. Also, if the runway slope is zero, such as in the thesis, the altitude is constant and the flight path angle γ is zero. The system of equations modelling the aircraft motion on ground is written in Equation 4.23.

$$\begin{cases}
\dot{x} = V \\
\dot{V} = \frac{g}{W} (T \cos(\alpha + \epsilon) - D - \mu_g R) \\
0 = T \sin(\alpha + \epsilon) + L + R - W \\
\dot{W} = -\dot{m}_f g
\end{cases}$$
(4.23)

By combining the third and second equations of Equation 4.23, the system reduces to Equation 4.24

$$\begin{cases} \dot{x} = V \\ \dot{V} = \frac{g}{W} [T\cos(\alpha + \epsilon) - D - \mu_g (W - L - T\sin(\alpha + \epsilon))] \\ \dot{W} = -\dot{m_f}g \end{cases}$$
(4.24)

Except during rotation phases of take-off and landing, the angle of attack α is constant. In the thesis, constant values are also given to the rolling ground friction coefficient μ_g depending on the nature of the rolling motion (i.e. propulsion or braking). Finally, since the relations of Equations 4.16 to 4.19 still hold and the altitude is constant, the system in Equation 4.24 is composed of three equations and involves four variables (i.e. x, V, W, and χ). Solving the system of equations thus requires one additional relation. In taxi, the equations of motion are solved at constant speed V. The take-off and landing ground runs are however evaluated for a given power rating χ , such as discussed more into details in the next section. Similarly to the flight case, the dynamic equations of the ground motion are actually solved with the help of the generalised Newton's second law of Equation 4.22 in XMDO.

4.4.3 Take-Off and Landing segments

During the thesis, take-off and landing segment models have been developed to better suit the needs for evaluating the low-speed performance of the studied aircraft configurations in XMDO and are described next.

One-G stall speed calculation

By definition, the one-G stall speed V_{s1g} is the stall speed at which the airplane can develop a lift force (normal to the flight path) equal to its weight. It is used to define regulatory speeds along the take-off and landing segments. According to the CS-25 regulation [34], a stall speed must be determined with "engines idling, or, if that resultant thrust causes an appreciable decrease in stall speed, not more than zero thrust at the stall speed". The current certification specifications therefore clearly limit the interest of the implementation of blown wing concepts which benefit from reduced stall speeds with high thrust levels. For the purpose of this thesis, it has been assumed that aviation regulations would allow the use of positive thrust for the calculation of stall speeds in the long-term.

The new calculation of V_{s1g} for a given set of aircraft weight, altitude and slat/flap settings, is then performed by solving the equations of steady level flight written in Equation 4.25. By definition of the stall angle α_{max} , the system is solved with $\alpha = \alpha_{max}$.

$$\begin{cases} W = T\sin(\alpha + \epsilon) + L \\ D = T\cos(\alpha + \epsilon) \end{cases}$$
(4.25)

Since the relations of Equations 4.16 to 4.19 still hold and the altitude is constant, the system in Equation 4.25 only involves two variables (i.e. V and χ). These variables are found by a Newton-Raphson solver set with an upper bound on χ equal to 1 (i.e. maximum thrust). Since L and D depend on χ , the new definition of V_{s1g} includes the power effect on aerodynamic forces and is associated to a power rating χ_{s1g} .

Take-off

Take-off performance calculation involves many regulatory speeds and requires the evaluation of several critical scenarios related to engine failures and runway conditions (e.g. dry, wet or contaminated) such as described in [3]. If regulatory rules are well defined for conventional airplanes, it is not the case for new aircraft concepts. Based on current practices, one of the objectives of the thesis has been to develop a simple model for the evaluation of take-off distances and associated energy consumptions for new aircraft configurations.

The proposed model includes three generic scenarios on dry runways which differ from each other by the operating components of the propulsion system after V_{EF} (i.e. the speed at which components are assumed to fail) and by the series of phases constituting the take-off segment. These three scenarios are shown in Figure 4.11 and detailed hereafter.

Nominal take-off: The aircraft is first accelerated on the ground at constant power rating χ_{TO} from break release to V_{LOF} (i.e. the speed at which the airplane first becomes airborne). It is important to note that the rotation phase that occurs during the ground run acceleration prior lift-off is neglected here for



Fig. 4.11 The three generic scenarios considered in the take-off segment model

simplicity. The flight path angle γ is therefore instantaneously switched from zero to γ_{TO} at V_{LOF} . The ground run is followed by an airborne acceleration phase at constant power rating χ_{TO} and constant flight path angle γ_{TO} . In this scenario, the take-off segment ends when the aircraft reaches an altitude of 35 ft above the runway. The aircraft regulatory speed at this altitude is called V_2 . In airworthiness standards the selection of V_{LOF} is constrained by both the minimum unstick speed V_{MU} [3] and the maximum tire speed V_{TIRE} [3]. Since the evaluation of these two regulatory speeds requires detailed analyses and data, V_{LOF} is taken equal to $1.1V_{s1g}$ which provides a first order estimate of the lift-off speed. The one-G stall speed V_{s1g} is calculated at the take-off weight with slats/flaps in take-off configuration. Also, since aviation regulations require 15% margin on the take-off distance with all engine operating, the take-off field length returned by the model is $1.15L_N$.

Take-off with component failure: This scenario differs from the previous one by the failure at V_{EF} of one or several components of the propulsion system provided as a list by the user. To account for the pilot recognition time, one second margin

between V_{EF} and the decision speed V_1 (i.e. the maximum speed at which the crew can decide to reject the take-off) is considered in today's standards. In the thesis it has been assumed that future improvements in pilot assistance and autonomous flight technologies would help reducing the recognition time, and V1 is thus taken equal to V_{EF} . The lift-off speed is here again taken equal to $1.1V_{s1g}$, but V_{s1g} is calculated with the propulsion system in failure configuration. For this scenario the take-off field length returned by the model is L_F .

Rejected take-off: In order to evaluate the accelerate-stop distance resulting from an aborted take-off at V_1 , the ground acceleration phase is followed by a ground deceleration at constant power rating χ_I . Note that χ_I refers in this scenario to the idle ratings of the non-failed power sources. The braking action is included by increasing the rolling ground friction coefficient μ_g from 0.025 (i.e. no braking) to 0.6. The model returns a runway length equal to L_R which includes a distance margin equivalent to two seconds at V_1 such as required by current standards for the evaluation of accelerate-stop distances.

Regarding the additional relations that are chosen for solving the equations of motion along the take-off segment (i.e. χ for the ground phase and (χ, γ) for the air phase), the trajectory (and so the field length) is an output of the successive solving of the system of equations. A 0.5 second time step is used for the field length evaluation FL_{out} and the total segment duration is limited to five minutes. In case the aircraft has not been able to take-off within the maximum duration or to simply accelerate because of inappropriate set of design variables (e.g. wing area, engine size), a distance of ten times the targeted field length FL_{req} is returned. Considering the take-off segment alone make this type of penalty unsuitable for the use of gradient-based algorithms. However, if the evaluation of a take-off segment is paired with the evaluation of a series of climb, cruise and descent segments, which has always been the case in XMDO studies, the constraints of Equations 4.20 and 4.21 drive the gradient-based algorithm. The take-off field length design requirement is then ensured by adding Equation 4.26 to the optimisation problem constraints.

$$FL_{out} \le FL_{reg}$$
 (4.26)

An alternative would be to provide the time history of (x, z) along the take-off segment with a set of B-spline curves, instead of (χ, γ) , and use a constraint on $\chi(t)$ (such as in Equation 4.20), instead of Equation 4.26. This formulation would then be compatible with the use of gradient-based algorithms for a take-off segment evaluation alone but may require more design variables to control the B-spline curves and to get a constant power rating which minimises the take-off distance.

In the considered scenarios, γ_{TO} and V_1 are key parameters for the optimisation of the take-off trajectories. It is first important to point out that the distance required to take-off at MTOW is usually limited by the regulatory requirements on V_2 or on the second segment gradient γ_2 . According to aviation standards, V_2 must be greater than $1.13V_{s1g}$ and γ_2 —relating to the flight path angle along the segment following the take-off sequence, and flown at V_2 with landing gear up [3]—must be more than γ_{min} which depends on the number of engines. All other things held constant, the take-off flight path angle γ_{TO} is therefore to be adjusted to fulfill the requirement on V_2 . Also, since neither the ground effect nor the landing gear drag are accounted for in the thesis, keeping γ_{TO} greater than γ_{min} ensures that the requirement on γ_2 is met. Instead of using a local solver to find the appropriate value of γ_{TO} , which would slow down the objective evaluation of the MDO problem, γ_{TO} is included in the design variables and both Equations 4.27 and 4.28 are added to the optimisation problem constraints.

$$V_2 \ge 1.13 V_{s1q}$$
 (4.27)

$$\gamma_{TO} \ge \gamma_{min} \tag{4.28}$$

For a given take-off weight, any increase in V_1 leads to a reduction in L_F . Indeed, a higher V_1 speed involves a longer acceleration phase with all components operating and, consequently, the same V_2 speed can be achieved at a shorter distance in case of component failure. On the contrary, any increase in V_1 leads to an increase in L_R because of the longer acceleration segment from brake release to V_1 , the longer deceleration segment from V_1 to the complete stop, and the longer 2 second segment at V_1 . Hence, for a given take-off weight, there exists an optimum value of V_1 for which the distance, called balanced field length, satisfies $L_F = L_R$. With the proposed take-off model, V_1 can be included in the design variables of the MDO problem for the consideration of balanced field lengths.

Landing

A simple landing segment model has also been developed during the thesis for the evaluation of landing distances and associated energy consumptions. The proposed segment is shown in Figure 4.12.



Fig. 4.12 Decomposition of the landing segment

The airborne phase begins 50 ft above the runway and is flown at constant flight path angle (-3°) and constant speed equals to the minimum regulatory speed $1.23V_{s1q}$. The one-G stall speed V_{s1q} is calculated at MLW with slats/flaps in landing configuration. The reason why V_{s1g} is not calculated at the first aircraft weight of the landing segment comes from both the possible B-spline formulation of the preceding segment and the speed continuity enforced between segments by XMDO. Indeed, the aircraft performance evaluation along such previous segment (e.g. descent) requires the last velocity of this segment (i.e. first velocity of the landing segment) to be known. Since the aircraft landing weight is unknown prior this evaluation, it has been chosen to calculate the first speed of the landing segment according to MLW which is always known. For simplicity, the round out manoeuvre is neglected in the proposed model and the flight path angle γ is instantaneously switched from -3°to zero at touchdown. Similarly to the rejected take-off scenario, the second part of the landing segment is a ground deceleration at constant power rating χ_I which relates to the idle ratings of the operating power sources. The landing distance L_L is also an output of the successive solving of the equations of motion with a 0.5 second time step. While the landing distance is always shorter than the take-off distance and, consequently, fulfils Equation 4.26, the speed along the airborne phase $V_{L_{out}}$ must be smaller than the landing speed of the design requirements $V_{L_{reg}}$. Equation 4.29 is therefore to be added to the MDO problem constraints.

$$V_{L_{out}} \le V_{L_{reg}} \tag{4.29}$$

4.4.4 Note on Newton-Raphson solver

The different methods that have been discussed throughout Section 4.4 all involve a Newton-Raphson solver. This well know gradient-based technique is a powerful root-finding algorithm. However, it may fail to converge for several reasons:

- Derivative issues: in most cases, the derivative is evaluated with Finite Differences (Appendix A) which may slow down the convergence due to approximation errors. In addition, if the function is not continuously differentiable, the Newton's method may diverge and fail.
- Stationary point: if a stationary point of the function is encountered, the derivative is zero and the method terminates (division by zero). Even if the point is not stationary but the derivative is small, the next iteration will move far from the current approximation.
- Starting point: as for all gradient-based techniques, the convergence depends on the initial point. For some functions, some starting points may enter an infinite cycle, preventing convergence.

In XMDO, the risk of non-convergence is mitigated by placing limits on the number of iterations, and bounding the solution to an interval known to contain the root. For example, the angle of attack α is bounded to [-90°,90°]. In addition, the component models are built to avoid stationary points or multiple roots. The propeller performance model (Appendix E) that has been developed during the thesis is a tangible example. In some rare cases the Newton-Raphson still failed in solving a very small number of flight points (one or two per nominal mission) for the optimum aircraft design found by XMDO, because of bad starting points. This could be improved with multi-starting (at computational cost) or by combining the method with a more robust root finding technique (e.g. bisection).

4.5 Propulsion system and power flows

The independence of XMDO core functions with the studied vehicle configuration, which forms one of the main assets of the tool, is ensured by the use of a generalised Newton's second law for the study of the vehicle dynamics. Such as discussed in Section 4.4, the magnitude of the resultant force is then controlled by the parameter called power rating and denoted χ . A generic formulation for the transformation of the power rating into forces is therefore also implemented. The solution proposed by XMDO to model the propulsion system and the different power flows is presented in the next section. The second section focusses on the specific developments of this thesis that have been required to carry out the studies of the considered hybrid-electric propulsion systems.

4.5.1 Rating formulation in XMDO

In XMDO, each power source of the propulsion system can actually be controlled by a local power rating r. This local power rating defines the amount of power delivered by the source according to a linear function which returns the maximum available power for r = 1 and zero power for r = 0. The power rating of a propulsion system composed of s power sources is then the vector $\chi = (r_1, ..., r_s)$.

The generic transformation of local power ratings into forces in XMDO is achieved by cascading the power delivered by each source across the downstream subcomponents such as in Figure 4.13. Note that the current version of XMDO only deals with power flows, but exchanging data sets between subcomponents such as (torque,speed) or (voltage,current) is easily achievable. Each subcomponent of the power chain is characterised by an input power P_{in} and an output power P_{out} that are reset to zero before each run of the propulsion chain. From the dimensions of the subcomponent, the model of the Geometry discipline returns the maximum power $P_{max}(t)$ it can receive or provide under the flight conditions at time t. The correct sizing of the subcomponent at the optimum solution is ensured by including Equation 4.30 into the optimisation problem constraints.

$$\forall t, \quad \max[P_{in}(t), P_{out}(t)] \le P_{max}(t) \tag{4.30}$$

When running the propulsion system evaluation, the output power of the power source is first computed according to its local power rating and added to the input power of the next subcomponent. From the input power, the subcomponent model of the Propulsion discipline returns the output power and the forces $\vec{F}_{i,j}$ that are produced by the subcomponents and act as external forces on the vehicle (e.g. cooling drag, thrust). Once the output power of the subcomponent has been evaluated, its value is added to the input power of the downstream subcomponent and the process continued until the last component of the chain.

The interest of this generic propulsion system formulation combined with the use of the power rating χ to control the magnitude of the resultant force can be commented more into details here. Indeed, one may want to use directly the thrust T as a control variable in the equations of motion (Equation 4.15) instead of χ and find the associated fuel mass flow rate and energy consumption by going upstream the propulsion chain from the propulsors (e.g. propeller, ducted fan). In the case of a turboprop engine for example (Figure 4.13a), which is an assembly of a turboshaft engine, a power gearbox and a propeller, both the propeller and the turboshaft engine generate thrust. The



Fig. 4.13 Illustration of simple power chains in XMDO

propulsion force created by the turboshaft engine is called residual thrust and comes from the turbine exhaust gas. Starting from the total thrust requirement requires an additional solving of the propulsion system power flows because the relation between residual thrust and propeller thrust is unknown. The solution proposed by XMDO enables to directly account for all the forces generated by each subcomponent of the propulsion chain in the resultant force, not to mention that the orientation of each force $\vec{F}_{i,i}$ can additionally be controlled with the help of control laws.

In order to represent a large variety of architectures with the concept of power cascading from sources, the notions of power summation and power split are used by XMDO as shown in Figure 4.14. For each subcomponent of the propulsion chain a list of input subcomponents and a list of output subcomponents are defined. The power split is performed directly at the output of the subcomponent and no intermediate object acts as power node. By default, power is uniformly distributed between the non-failed output subcomponents but time-dependent weighted distributions can also be defined and driven by optimisation design variables. The architectures can therefore be relatively complex such as illustrated in Figure 4.15. Let us recall that XMDO uses configuration files (Section 4.2.4 and Appendix A) for the definition of the optimisation cases. The configuration file of the propulsion chain contains the list of subcomponents

composing the system and the causal relations between them (i.e. input and output lists of subcomponents). Switching from one propulsion chain study to an other then requires only the configuration file of the propulsion system to be changed (under the condition that all component models are available).

The use of input and output lists of subcomponents however requires to manage the execution order of the subcomponents. To do so, a flag "is to run" is associated to each subcomponent and a subcomponent can be run only if the flag "is to run" of all the input subcomponents are false (i.e. all the input components have already been run). Looping over the list of subcomponents of the power chain provides an efficient way to execute the entire propulsion system in the correct order.



Fig. 4.14 Representation of a subcomponent in the propulsion chain



Fig. 4.15 Example of a complex propulsion chain in XMDO

If the architecture includes s power sources, then s-1 additional relations involving the local power ratings are required to solve the equations of motion. For a twin turboprop aircraft, the relation is of the form $r_1 = r_2$ in nominal operation or $r_1 = 0$ in case of failure of engine 1. These relations that relate to control laws can of course be provided using B-spline curves along each segment of the mission.

Finally, XMDO also provided a simple way to manage power at battery interfaces. A battery is a particular subcomponent that can be both a power source and a load. The solution proposed by XMDO was to use the sign of the battery local power rating to specify its operating mode (charge for r > 0 or discharge for r < 0). This formulation actually defines two different architectures which depend on the battery mode. As an example, if subcomponent 2 of Figure 4.15 is a battery then the link with power source 1 is deactivated when discharging while the links to subcomponents 5 and 7 are deactivated when charging. However this solution raises issues when the battery is in a branch with other subcomponents such as subcomponent 6 of Figure 4.15. Indeed, in such case the local power rating of the battery must be known by the upstream subcomponents connected to the branch to prevent them from sending power when the battery is in discharge mode. Since no information is sent backward in the presented solution, the evaluation of such architecture was impossible. Modifications of the propulsion system evaluation to cover architectures with one or several branches with batteries have therefore been implemented and are presented in the next section.

4.5.2 New propulsion system modelling

The method developed in this thesis for the transformation of local power ratings into forces still relies on power cascading ,and thus, on the orientation of the propulsion chain with input and output lists of subcomponents for each subcomponent of the propulsion system (Figure 4.14). While power cascading starts from the power sources in the initial formulation proposed by XMDO, it starts from the *priority subcomponents* in this model. *Priority subcomponents* include power sources (i.e. turboshafts and batteries) but also NPL (Non-Propulsive Load) subcomponents that are introduced to model the power consumption of non-propulsive systems. An example of the implementation of these *priority subcomponents* is shown in Figure 4.16. A propeller can also be defined as a *priority subcomponent*, as depicted in Figure 4.17, since the direct control of the propeller state can be useful in some cases (e.g. high-lift propellers). Due to the use of power cascading, a *priority subcomponent* must be at the extremity of a branch (i.e. no input list of subcomponents).

In the proposed model the orientation of the propulsion system in the architecture description (i.e. input and output lists of subcomponents) can be opposite to the real power flow: if so, the powers sent to the downstream subcomponents according to the architecture description are negative. For example, a NPL subcomponent which is a power sink in reality always sends negative power values to its downstream subcomponents. The subcomponent models have then been made compatible with both positive and negative power values, which has also been useful to model subcomponents that can operate in both directions such as electric machines, propellers, batteries, etc. The constraint of Equation 4.30 used to ensure the correct sizing of a subcomponent in XMDO has also been rewritten with absolute values such as in Equation 4.31.

$$\forall t, \quad \max[|P_{in}(t)|, |P_{out}(t)|] \le |P_{max}(t)|$$
(4.31)

Contrary to the rating formulation presented before, the sign of the transmitted power does not define different architectures. Since only powers are handled by the current propulsion system model, the basic operations that are implemented to account for the sign of the subcomponent input power P_{in} are written in Equation 4.32, where P_{out} is the subcomponent output power and $\eta_{i\to o}$ ($\eta_{o\to i}$, respectively) is the subcomponent efficiency when operating in the same direction as (opposite direction to, respectively) the architecture description. The subcomponent models used for the calculation of $\eta_{i\to o}$ and $\eta_{o\to i}$ are described in the next chapters.

$$\begin{cases}
P_{in} \ge 0, & P_{out} = P_{in}\eta_{i \to o} \\
P_{in} < 0, & P_{out} = \frac{P_{in}}{\eta_{o \to i}}
\end{cases}$$
(4.32)

In the selected propulsion system model, a subcomponent output power is uniformly distributed between the non-failed subcomponents of its output list because no power node needs to be defined and controlled with a proper orientation of the architecture description. In particular, the correct evaluation of the architecture requires at least one branch of the architecture description to include a subcomponent that is not a *priority subcomponent* at its extremity. In addition, solving the equations of motion for an architecture containing p priority subcomponents requires p-1 additional relations involving the local power ratings of these subcomponents. In this thesis, the Newton-Raphson solver used for Newton's second law solving (Figure 4.4) manages only two variables: the aircraft angle of attack α and the local power rating r_{NR} of one of the priority subcomponents. The local power ratings of the other priority subcomponents are expressed as a function of r_{NR} or defined with B-spline curves that can be controlled by design variables. For example, the hybrid parallel architecture shown in Figure 4.16 contains six *priority subcomponents*, namely two turboshaft engines, two batteries and two NPL subcomponents. The battery local power rating r_{B1} is chosen as r_{NR} , the turboshaft local power rating r_{T1} is provided by a B-spline curve and the local power
rating r_{NPL1} is taken equal to unity to model a constant power consumption of NPL1. In symmetric flight, the relations $r_{B2} = r_{B1}$, $r_{T2} = r_{T1}$ and $r_{NPL1} = r_{NPL2}$ are used. The partial turboelectric architecture of this thesis illustrated in Figure 4.17 includes twelve *priority subcomponents*: two turboshaft engines, hight high-lift propellers and two NPL subcomponents. The turboshaft local power rating r_{T1} is selected as r_{NR} , and the relations $r_{T1} = r_{T2}$ and $r_{NPL1} = r_{NPL2} = 1$ hold in nominal flight. The local power ratings of the height high-lift propellers are set to unity during take-off, landing, and one engine failure scenario, and to zero otherwise. More details about the assumptions used to carry out the evaluations of these architectures can be found out in Chapter 5 and Chapter 6.



Fig. 4.16 Orientation of the hybrid parallel architecture HP

The real operating domain of the *priority subcomponent* that is controlled through r_{NR} by the Newton-Raphson solver is characterised by an interval of local power ratings $[r_{min}, r_{max}]$. In the thesis, $r_{max} = 1$ for all *priority subcomponents*, but r_{min} can be -1 for batteries or 0.05 for turboshaft engines that cannot be turned-off during flight (i.e. no start and stop). For the correct solving of the equations of motion even in case of improper set of design variables, the Newton-Raphson solver is allowed to search



Fig. 4.17 Orientation of the partial turboelectric architecture **PT8**

a solution for r_{NR} outside the interval. As detailed in Section 4.4, the propulsion system returns increasing propulsive power for $r_{NR} > r_{max}$ and Equation 4.33 (which is equivalent to Equation 4.20) is included in the MDO problem constraints to prevent overworking the controlled *priority subcomponent*.

$$\forall t, \quad r_{NR}(t) \le r_{max} \tag{4.33}$$

In some cases the thrust produced by the propulsion system at r_{min} is still too high to satisfy Newton's second law. To cope with this, the propulsion system is evaluated at r_{min} and the drag force D_{brk} of Equation 4.34, that could be produced by air-brakes or wheel brakes, or both, is added along the aerodynamic x-axis (Figure 5.2) to the resultant force .

$$\begin{cases} r_{NR} \ge r_{min}, & D_{brk} = 0\\ r_{NR} < r_{min}, & D_{brk} = W(r_{min} - r_{NR}) \end{cases}$$
(4.34)

The maximum drag force $D_{brk_{max}}$ that can be effectively added is estimated with Equation 4.35, where D_{glider} is the aerodynamic drag of the aircraft without propulsors. The factor 1.5 defines a speed brake that can be easily designed with today's technologies [126].

$$D_{brk_{max}} = 1.5D_{glider} + (0.6 - \mu_g)R \tag{4.35}$$

Finally, to avoid unrealistic values of D_{brk} at the optimum design, Equation 4.36 is also included into the optimisation problem constraints.

$$\forall t, \quad D_{brk}(t) \le D_{brk_{max}}(t) \tag{4.36}$$

4.6 Constraints summary

Throughout this chapter several equations relating to constraints added to the global optimisation problem to perform the design of the different hybrid aircraft configurations of the thesis have been quoted. For clarity, names are given to these constraints and are summed up in Table 4.2. Let us recall that they apply to all the missions included in the design process.

Name	Criteria	Equation	No.
R_{MTOW}	Max. structural weight	$TOW \leq MTOW$	4.1
R_{MZFW}	Max. structural weight	$ZFW \leq MZFW$	4.2
R_{MLW}	Max. structural weight	$LW \le MLW$	4.3
R_{Fuel}	Fuel quantity	$fuel_{req} \leq fuel_{in}$	4.4
R_{AoA}	Aerodynamic stall	$\forall t, \alpha(t) \le \alpha_{max}$	4.21
$R_{r_{max}}$	Max. local power rating	$\forall t, r_{NR}(t) \le r_{max}$	4.33
$R_{D_{brake}}$	Speed-brake design	$\forall t, D_{brk}(t) \le D_{brk_{max}}(t)$	4.36
$R_{SubComp}$	Subcomponent design	$\forall t, \max[P_{in}(t) , P_{out}(t)] \le P_{max}(t) $	4.31
R_{TOFL}	Take-off field length	$FL_{out} \leq FL_{req}$	4.26
R_{V2}	V_2 speed at take-off	$V_2 \ge 1.13 V_{s1g}$	4.27
$R_{\gamma_{TO}}$	$2^{\rm nd}$ segment slope	$\gamma_{TO} \ge \gamma_{min}$	4.28
R_{V_L}	Landing speed	$V_{L_{out}} \le V_{L_{req}}$	4.29

Table 4.2 Constraints of this chapter included into the global optimisation problem

Chapter 5

Reference aircraft optimisation

Contents

5.1	Aircraft modelling
5.2	Component modelling
5.3	Optimisation
5.4	Optimisation results

This chapter focuses on the optimisation of the reference aircraft **REF** with XMDO. The design models that are used at aircraft level but also at component levels are first presented. A great deal of work was dedicated to the development of these models. For the sake of clarity, most of them are placed in appendices describing their building methods.

Then, the optimisation cases are defined: the different missions, the constraints and the objective function, and the optimisation algorithms that were selected.

Lastly, two optimisation cases deferring from each other by the set of missions included in the optimisation process are compared. In the first case, the objective function is calculated according to a single mission while the second case uses a weighted objective function. The main outputs of these two optimisations are provided and discussed.

Notation: except for the notion of *priority subcomponent* that is used in reference to the generic propulsion system model presented in Chapter 4 (Section 4.5.2), the subcomponents of the propulsion chain (defining the Prop Train component in XMDO's vehicle description, Figure 4.3) are named components in this chapter. This simplifies the notations and the equations by avoiding double indexes and double summations.

The aircraft is therefore represented as an assembly of components: wings, fuselage, horizontal stabiliser, vertical stabiliser, propeller, gearbox, turboshaft, etc. The subscript C is used in the formulas to refer to one of the components composing the aircraft.

5.1 Aircraft modelling

5.1.1 Geometry

The aircraft geometry is modelled by an assembly of basic geometrical shapes as shown in Figure 5.1. For subsonic aircraft, the quarter of the wing Mean Aerodynamic Chord (MAC) [133] is most always used for placing the wing versus the aircraft centre of gravity as it is a first rough indicator of the position of the aircraft centre of pressure (i.e. acting point of aerodynamic forces) which moves with the aircraft angle of attack. Since detailed space allocation studies are out of the scope of the thesis, it has been assumed that moving the wing fore or aft in order to match the aircraft centre of gravity with the approximated wing centre of pressure would have brought low added value and induced second order effects on the results. Therefore, the wing is fixed versus the fuselage according to the 25% of the wing MAC and the aircraft centre of gravity is assumed to be vertically down to this point along the fuselage axis (Subsection 5.1.2). The position of the vertical stabiliser relative to the fuselage is fixed through the intersection of its trailing edge with the fuselage rear end. Finally, the leading edge of the horizontal stabiliser also intersects the leading edge of the vertical stabiliser at the tip, fixing their relative positioning.

5.1.2 Mass

As mentioned previously, the x-coordinate of the aircraft centre of gravity in the body reference frame (Figure 5.2) is supposed to be the same as that of the 25% of the wing MAC. The aircraft OEW is simply the sum of the weights of all the components constituting the aircraft and presented in Section 5.2.

5.1.3 Aerodynamics

As a remainder of the previous chapter, the different coordinate systems that are used for the definition of the main external forces are depicted in Figure 5.2.



Fig. 5.1 Main parameters for aircraft geometry description

Let us recall that for a given flap configuration and a given engine power rating, the angle of attack α is used to control the magnitude of the aerodynamic forces acting on the aircraft. The aerodynamic resultant at aircraft level is commonly divided into a lift force L_A and a drag force D_A . In the thesis, L_A and D_A are defined by Equation 5.1, where L_C and D_C are respectively the lift and drag forces returned by the aerodynamic model of component C.

$$L_A = \sum_C L_C \quad D_A = \sum_C D_C \tag{5.1}$$

At component level, the drag force D_C is subdivided into:

• D_{C_f} : the friction drag generated by skin friction between the molecules of the air and the solid surface



Fig. 5.2 External forces and reference frames

- D_{C_p} : the viscous pressure drag (or form drag) arising from the varied pressure distribution which depends on the body shape
- D_{C_a} : the additional drag (or excrescence drag) created by surface imperfections (e.g. panel joints, gaps around doors and control surfaces, etc...).
- D_{C_i} : the induced drag (or lift-induced drag) resulting from the creation of lift on a three-dimensional lifting body (e.g. wing, tail plane, etc...).

The wave drag, also called transonic compressibility drag, is neglected in the aerodynamic models because of the subsonic flight conditions of the aircraft of interest (Mach<0.6). More physical details about the above aerodynamic drag components can be found in [7] for example. Classical methods, detailed by Torenbeek [133], Roskam and Lan [111], Raymer [104] or Kroo and Shevell [62], are used for the evaluation of the drag components and can be found in Appendix C.

5.1.4 Propulsion

The resultant force \vec{T}_A produced by the propulsion system and acting on the aircraft is defined by Equation 5.2, where \vec{T}_C is the force returned by the propulsion model of subcomponent C (i.e. $\vec{T}_A = \vec{R}$ and $\vec{T}_C = \sum_j \vec{F}_{C,j}$ in reference to Figure 4.13b of Chapter 4).

$$\vec{T}_A = \sum_C \vec{T}_C \tag{5.2}$$

The orientation of the reference propulsion system in XMDO is shown in Figure 5.3. The local power rating $r_{T_2}^{-1}$ of Turboshaft₂ is chosen as r_{NR} , the local power rating controlled by the Newton-Raphson for solving the equations of motion. The minimum power rating $r_{NR_{min}}$ (i.e. idle rating) for r_{T_2} is set to 0.05. If Turboshaft₂ is never switched off or put into failure mode with this convention, it is not the case for Turboshaft₁. In symmetrical flight, the relation $r_{T_1} = r_{T_2}$ is used, but r_{T_1} can be set to zero for single engine taxi or engine failure scenario. For simplicity, the local power rating r_{NPL_i} of the non-propulsive load NPL_i is 1 while $r_{T_i} > 0$, and zero otherwise. In other words, if Turboshaft₁ is not operating, only half of the total non-propulsive power demand is supplied. The power off-take on Turboshaft₂ is therefore underestimated in single engine operations since, in practice, this engine shall be able to supply all the non-propulsive systems operating in degraded modes (e.g. lower cabin pressure).



Fig. 5.3 Orientation of the reference propulsion system architecture

5.2 Component modelling

Component models are described for the following disciplines: Geometry, Mass, Aerodynamics and Propulsion. When the influence of a component on a particular discipline is assumed to be none, this discipline is not mentioned.

¹Reminder: Turboshaft₂ delivers its maximum power when $r_{T_2} = 1$, and is off when $r_{T_2} = 0$

5.2.1 Fuselage

Geometry. As indicated in Chapter 2, the fuselage geometry is fixed (Figure 5.1).

Mass. The fuselage mass is estimated thanks to a confidential semi-empirical formula taking the fuselage geometry and the aircraft characteristic weights (i.e. MTOW, MZFW and MLW) as inputs. Similar formulas can be found in [133] and [104] for examples.

Aerodynamics. The fuselage contribution to lift is supposed to be zero. Hence, $L_C = D_{C_i} = 0$. The other drag components are evaluated thanks to Appendix C.

5.2.2 Nacelle

Geometry. The nacelle length and diameter are expressed as a function of the dimensions of the carried components (i.e. turboshaft and power gearbox for the reference aircraft). In addition, the spanwise location of the nacelle is defined according to the propeller diameter (Figure 5.1).

Mass. The nacelle mass is calculated with a confidential semi-empirical formula defined as a function of the nacelle geometry and the total mass of carried components.

Aerodynamics. The nacelle contribution to lift is supposed to be zero. Therefore, $L_C = D_{C_i} = 0$. The other drag components are evaluated according to Appendix C.

5.2.3 Wings

Geometry. Wings are made of a rectangular central part (i.e. $c_{b_W} = c_{r_W}$) and two trapezoidal external parts. As already mentioned in Chapter 2, each wing is fitted with single slotted flaps with a 30% flap chord ratio (i.e. c_{f_i}/c_{r_W}) and inboard and outboard limits of 10% and 70% of the semi-span, respectively (i.e. $\eta_i = 0.1b_W/2$ and $\eta_o = 0.7b_W/2$). With the wing aerodynamic model implemented in this thesis, the flap deflection angle δ can be varied from 0° to 80°. The root chord c_{r_W} and the span b_W are selected as design variables to change the wing geometry, the taper ratio of the external panels (i.e. c_{t_W}/c_{b_W}) being kept constant versus ATR72.

Mass. The wing mass M_W is evaluated from a formula of similar form as the one proposed by Kroo and Shevell [62]. In this formula, written in Equation 5.3, S_W is the wing planform area, and X_W and Y_W are data-fitted coefficients. It should be noted that if the validity of this expression based on classical propulsion system installations cannot be guaranteed with distributed propulsion configurations, it has nevertheless been used for them as well.

$$M_W = X_W S_W + Y_W b_W^3 \frac{\sqrt{MTOWMZFW}}{S_W}$$
(5.3)

Aerodynamics. The generalised method of Appendix C including the blown wing effect is implemented for the evaluation of aerodynamic forces acting on the aircraft wings. For a given aircraft geometry and under given flight conditions, L_c and D_C are then functions of α , δ and r_{NR} .

5.2.4 Horizontal stabiliser

Geometry. The tail plane is a simple trapezoidal lifting surface. Taper ratio (i.e. c_{t_H}/c_{r_H}), sweep angle φ_H , and aspect ratio (i.e. b_H^2/S_H) are all kept constant versus ATR72, and the root chord c_{r_H} is chosen as design variable to modify the surface area.

Mass. The mass of the horizontal stabiliser M_H is given by the polynomial fit of Equation 5.4 built during the thesis, where S_H is the planform area, and A_H and B_H are the regression coefficients.

$$M_H = (A_H S_H + B_H) S_H \tag{5.4}$$

Aerodynamics. Appendix C is used for the evaluation of L_C and D_C . Let us recall that the correct sizing of the horizontal stabiliser is ensured if the modified volume coefficient \overline{V}_H , defined in the same appendix, meets Equation C.8 for all points of the trajectory.

$$\forall t, \quad \overline{V}_H(t) \ge \overline{V}_{H_{min}} \tag{5.5}$$

5.2.5 Vertical stabiliser

Geometry. The vertical stabiliser is modelled by a trapeze. Similarly to the horizontal tail plane, the root chord c_{r_V} is chosen as design variable while the taper ratio (i.e. c_{t_V}/c_{r_V}), the sweep angle φ_V and the aspect ratio (i.e. b_V^2/S_V) are fixed with ATR72 as reference.

Mass. The mass M_V of this component is also estimated from a polynomial regression written in Equation 5.6, where S_V is the planform area, and A_V and B_V are the regression coefficients.

$$M_V = (A_V S_V + B_V) S_V \tag{5.6}$$

Aerodynamics. L_C and D_C are calculated thanks to the formulas of Appendix C and the sizing constraints of Equations C.12 and C.13 (also defined in the appendix) are evaluated.

$$V_{V_s} \ge 0.02 \tag{5.7}$$

$$\forall t, \quad V_{V_c}(t) \ge V_{V_{c_{min}}} \tag{5.8}$$

5.2.6 Equipments

Mass. The mass of this component includes the contribution of: landing gears, environmental control systems, furnishings, crew members and operational items. Except for the mass of the landing gears that is evaluated as a function of the aircraft MTOW, the mass of the other contributors is assumed constant throughout the studies of the thesis. Hence, the total mass M_E of these equipments is calculated thanks to a formula of the form of Equation 5.9, where K_E is the constant part of the total mass and f_{LG} returns the mass of the landing gears.

$$M_E = K_E + f_{LG}(MTOW) \tag{5.9}$$

5.2.7 Non-propulsive load

Propulsion. Such component only influences the propulsion discipline but does not create any external force acting on the aircraft (i.e. $\vec{T}_C = 0$). This component being defined as a *priority subcomponent* (Section 4.5.2), as shown in Figure 5.3, the characteristic power values for each non-propulsive load NPL_i are: $P_{max} = 70,000$ W, $P_{in} = 0$, and $P_{out} = -r_{NPL_i}P_{max}$.

5.2.8 Turboshaft

Geometry. The geometrical model described in Appendix D is implemented. The diameter D_{gt} is used as design variable to control the turboshaft size. The specific

rotation speed of the component N_{gt_s} is calculated according to the expression of $N_{LP_{design}}$.

Mass. The component mass is evaluated thanks to the simple formula of Appendix D.

Propulsion. The propulsion model is based on the specific fuel consumption model of Appendix D. It is important to note that the thrust produced by the exhaust gas of the turboshaft is neglected, and, therefore, $\vec{T}_C = 0$. The characteristic power values of this *priority subcomponent* (Section 4.5.2) are: $P_{max} = P_{MCL}$, $P_{in} = 0$, and $P_{out} = r_{T_i}P_{max}$. The fuel consumption is of course taken into account thanks to the specific fuel consumption model.

5.2.9 Propeller

Geometry. A 6-bladed variable-pitch propeller is considered. The propeller size is controlled through its diameter D_p . The specific rotation speed of the component N_{ps} is taken equal to the maximum propeller speed N_{max} defined in Appendix E.

Mass. The mass estimation model of Appendix E is used.

Propulsion. The selected model for variable-pitch propellers described in Appendix E is implemented. The propeller force is defined as $\vec{T}_C = T\vec{e}_{x_B}$ (i.e. $\epsilon = 0$, Figure 5.2), where T is the value returned by the performance model of the appendix. The maximum power P_{max} that can be absorbed by the propeller is also evaluated from the formula of this appendix.

5.2.10 Power gearbox

Geometry. This component is a "turboprop" category gearbox, defined in Appendix F. The associated geometrical model is used, and the gearbox diameter D_{gbx} is selected as design variable. The gear ratio M_0 is taken equal to N_{qt_s}/N_{p_s} .

Mass. The mass is calculated according to the mass model of Appendix F.

Propulsion. The component efficiency is evaluated from the simple performance model of Appendix F. The maximum power P_{max} that can flow through the gearbox is taken equal to the product of the maximum input torque capability $T_{in_{max}}$, given by the same appendix, and the specific rotation speed of the turboshaft. Finally, $\vec{T}_C = 0$ for gearboxes.

5.3 Optimisation

After defining the different missions that are used for the optimisation of the reference aircraft, the constraints and the objective function that are considered in the optimisation problem are exposed. Finally, the selection of the optimisation algorithms is discussed.

5.3.1 Missions

Nominal mission

A nominal mission #i can be used to evaluate the aircraft performance for the payload_i carried over the range_i. Such mission is made of the following sequence of segments (Figure 5.4):

- **Taxi-out:** In taxi, the aircraft speed is set to 10 m/s. During this first taxi phase of 2.5 minutes, the turboshaft engines are allowed to be stopped. In the case of the reference aircraft, r_{T_1} is set to zero for single engine taxiing. By default the take-off flap configuration is used for all the taxi phases.
- **Taxi-out_H:** Along this second taxi phase, all the gas turbines that will be used for the take-off must be on while the aircraft is still moving at 10 m/s during 1.5 minutes. For the reference aircaft, $r_{T_1} = r_{T_2}$ (≥ 0.05).
- **Take-off:** As described in the previous chapter, the trajectory of a nominal take-off is controlled through the flight path angle γ_{TO} . In addition, a take-off rating $r_{TO} \leq 1$ is defined for two reasons:
 - the take-off power of a turboprop engine is usually mechanically limited by the gearbox: if sized by the power requirements at high altitude, the gas turbine may not be operated at maximum power capability at take-off to reduce mechanical loads applied to the gearbox.
 - for an all electric aircraft using batteries, the maximum rating being defined according to the maximum discharge power (see Chapter 6), performing a take-off at maximum discharge power can lead to tremendous power values for energy-sized batteries.

Hence, for the reference aircraft, $r_{T_1} = r_{T_2} = r_{TO}$. Finally, the flap deflection angle during the take-off phase is controlled by the design variable δ_{TO} .

- Climb: The climb trajectory is a B-spline (Chapter 4) controlled at the first point by the vertical speeds Vz_{CL_i} and at the last point by the distance defined through x_{CR_i} (Figure 5.4), the altitude set by the cruise altitude z_{CR} , the vertical speeds Vz_{CL_f} and the cruise speed M_{CR} . The segment duration is modified through the duration factor dt_{CL} (i.e. dt_{fact} in Chapter 4). The flap retraction law is also a B-spline (Chapter 4) with an initial value of δ_{TO} and a final value of zero. The flap retraction rate is controlled by the uniformity factor u_{CL} (i.e. u_{fact} in Chapter 4).
- **Cruise:** The cruise is a level flight at the altitude z_{CR} and Mach M_{CR} . The first and last distances from the last point of the take-off segment are illustrated in Figure 5.4. The aircraft is of course in clean configuration.
- **Descent:** The B-spline for the descent trajectory is controlled at its first point by the vertical speed Vz_{DSC_i} (in addition to the parameters driving the cruise trajectory), and at its last point by the vertical speed Vz_{DSC_f} . The segment duration is adjusted thanks to the duration factor dt_{DSC} . The flap extension law has initial and final values of zero and δ_{LG} , respectively, the latter being the landing flap deflection angle. The flap extension rate is controlled by the uniformity factor u_{DSC} .
- **Landing:** The landing phase is performed with the landing flap configuration. For the reference aircraft, and according to Chapter 4, $r_{T_1} = r_{T_2} \ge r_{NR_{min}}$ while airborne and $r_{T_1} = r_{T_2} = r_{NR_{min}}$ for the braking phase.
- **Taxi-in_H** This phase is evaluated under the same assumptions as $Taxi-out_H$.
- **Taxi-in** This phase is evaluated under the same assumptions as *Taxi-out* with a duration reduced to 1.5 minutes.

Table 5.1 summarises the parametric control laws for the reference aircraft along a nominal mission. Finally, the fuel quantity $fuel_{in}$ loaded prior to the start of the taxi-out phase is also a design variable for a nominal mission. It is important to note that no energy reserves are considered in the thesis for the sake of reducing the computation time. Their correct evaluation according to current regulatory rules Airbus [3] would have required the evaluation of additional segments. The fuel quantity after the last evaluation point of the taxi-in phase can therefore be zero for the reference aircraft. More comments about this assumption are provided in the conclusion of this manuscript.



Fig. 5.4 Definition of the main distances along a nominal mission

Segment	Propulsion	Flap configuration
Taxi-out	$r_{T_1} = 0$	$\delta = \delta_{TO}$
$\operatorname{Taxi-out}_{\mathrm{H}}$	$r_{T_1} = r_{T_2}$	$\delta = \delta_{TO}$
Take-off	$r_{T_1} = r_{T_2} = r_{TO}$	$\delta = \delta_{TO}$
Climb	$r_{T_1} = r_{T_2}$	$\delta(t_i) = \delta_{TO} \ \delta(t_f) = 0 \ u_{CL} \in]0,1[$
Cruise	$r_{T_1} = r_{T_2}$	$\delta = 0$
Descent	$r_{T_1} = r_{T_2}$	$\delta(t_i) = 0 \ \delta(t_f) = \delta_{LG} \ u_{DSC} \in]0,1[$
Landing	airborne: $r_{T_1} = r_{T_2}$ ground: $r_{T_1} = r_{T_2} = 0.05$	$\delta = \delta_{LG}$
$\mathrm{Taxi}\text{-}\mathrm{in}_{\mathrm{H}}$	$r_{T_1} = r_{T_2}$	$\delta = \delta_{TO}$
Taxi-in	$r_{T_1} = 0$	$\delta = \delta_{TO}$

Table 5.1 Control laws for the reference aircraft along a nominal mission

One Engine Inoperative (OEI) take-off

The OEI take-off mission is a take-off sequence at MTOW with the failure of one of the prime movers (Section 4.4). Regarding the numerous degrees of freedom that are considered in this thesis for solving the aircraft design problem, the decision speed V_1 is finally not included into the set of design variables, and is defined as a function of the lift-off speed V_{LOF} instead ($V_1 = 0.99V_{LOF}$). For the reference aircraft, the power rating r_{T_1} is therefore switched from r_{TO} to zero at $V_{EF}(=V_1)$. As already said, Propeller₁ is then instantaneously feathered.

Rejected take-off

The rejected take-off mission is a take-off sequence at MTOW with the failure of one of the prime movers (Section 4.4) at $V_{EF}(=V_1)$, instantaneously followed by the take-off

abortion. For the reference aircraft, at V_{EF} , r_{T_1} is then switched to zero while r_{T_2} is set to 0.05.

Approach at MLW

This mission is actually made of a single evaluation point at MLW in landing configuration. The equations of motion are solved for the first point of the airborne phase of a landing sequence (i.e. z = 50 ft, $V = 1.23V_{s1g}$, $\gamma = -3^{\circ}$). However, since the models take into account the effect of the propulsion in the calculation of V_{s1g} , this flight point is evaluated with the failure of one of the prime movers. For the reference aircraft, $r_{T_1} = 0$.

OEI ceiling

The OEI ceiling mission is also composed of a single flight point. For simplicity this point is evaluated at MTOW, which is a conservative assumption for variable mass aircraft. The aircraft is supposed to be in level flight at the targeted OEI ceiling altitude, in clean configuration. The flight mach number M_{OEI} is however included into the set of the design variables. The equations of motion are then solved under the above conditions with one of the prime movers in failure mode. For the reference aircraft, $r_{T_1} = 0$.

5.3.2 Constraints and objective function

The reference aircraft is designed under the Top Level Aircraft Requirements of Table 5.2. As can be seen from this table, the maximum payload is supposed to be the same as the nominal payload for simplicity.

Name	Value
Nominal number of passengers	70
Design payload	$6,\!650~\mathrm{kg}$
Maximum payload	$6,\!650~\mathrm{kg}$
Maximum range	400 nm
Take-Off Field Length (MTOW, Sea Level, ISA)	$1,400 {\rm m}$
Approach speed (MLW, Sea Level, ISA)	113 kt (58 m/s)
One engine inoperative ceiling (ISA)	10,000 ft

Table 5.2 Reference aircraft Top Level Aircraft Requirements

The sizing constraints presented in this chapter, and added to the constraints discussed in Chapter 4, are listed in Table 5.3. A constraint applies to all the missions included in the design process under the condition that it can be defined for the mission of interest (e.g. R_{V_2} is not defined for the OEI ceiling mission). In Table 5.3, FL_{req} and $V_{L_{req}}$ are therefore 1,400 m and 58 m/s, respectively. In addition, for all the aircraft configurations of the thesis, $r_{max} = 1$. For confidentiality reasons, α_{max} , $\overline{V}_{H_{min}}$ and $V_{V_{c_{min}}}$ are not provided in the manuscript.

Name	Criteria	Equation	No.
R_{MTOW}	Max. structural weight	$TOW \leq MTOW$	4.1
R_{MZFW}	Max. structural weight	$ZFW \leq MZFW$	4.2
R_{MLW}	Max. structural weight	$LW \le MLW$	4.3
R_{Fuel}	Fuel quantity	$fuel_{req} \leq fuel_{in}$	4.4
R_{AoA}	Aerodynamic stall	$\forall t, \alpha(t) \le \alpha_{max}$	4.21
$R_{r_{max}}$	Max. local power rating	$\forall t, r_{NR}(t) \le r_{max}$	4.33
$R_{D_{brake}}$	Speed-brake design	$\forall t, D_{brk}(t) \le D_{brk_{max}}(t)$	4.36
$R_{SubComp}$	Subcomponent design	$\forall t, \max[P_{in}(t) , P_{out}(t)] \le P_{max}(t) $	4.31
R_{TOFL}	Take-off field length	$FL_{out} \le FL_{req}$	4.26
R_{V2}	V_2 speed at take-off	$V_2 \ge 1.13 V_{s1g}$	4.27
$R_{\gamma_{TO}}$	$2^{\rm nd}$ segment slope	$\gamma_{TO} \ge \gamma_{min}$	4.28
R_{V_L}	Landing speed	$V_{L_{out}} \leq V_{L_{reg}}$	4.29
R_{HTP}	Horizontal tail sizing	$\forall t, \overline{V}_H(t) \ge \overline{V}_{H_{min}}$	C.8
R_{VTPs}	Vertical tail sizing (stability)	$V_{V_s} \ge 0.02$	$\mathrm{C.12}$
R_{VTPc}	Vertical tail sizing (control)	$\forall t, V_{V_c}(t) \geq V_{V_{c_{min}}}$	C.13

Table 5.3 Constraints for the reference aircraft optimisation problem

If minimising the aircraft MTOW is classically chosen for conventional aircraft design (Chapter 3), such objective function does not make sense any more with hybrid aircraft. While many minimisation criteria can be considered such as energy, costs and environmental impacts, this thesis focuses on the minimisation of fuel consumption along the nominal missions. The objective function f is then defined through Equation 5.10, where F_{M_n} denotes the aircraft fuel burn for the nominal mission M_n , and w_{M_n} is a weighting factor that can be chosen according to the predicted flight distribution.

$$f = \sum_{M_n} w_{M_n} F_{M_n} \tag{5.10}$$

5.3.3 Optimisation algorithms and penalty functions

Since XMDO was primarily developed to be used with gradient-based algorithms and, if required, external penalties, these optimisation features (Appendix A) have first been tested on the reference aircraft case.

For the evaluation of derivatives, the failure rate of a given algorithm appeared to be greater when provided with Finite Differences than with Complex Steps, because of the higher sensitivity of Finite Differences to step size. On the optimiser side, both the SLSQP (Sequential Least-Squares Quadratic Programming) from the NLopt [59] library and the L-BFGS-B (Limited-memory BFGS) from Scipy [60] have been tried. Similarly to the first studies carried out with XMDO, SLSQP always failed in solving the problem. On the contrary, the combination of the L-BFGS-B algorithm with the external penalty methods (kept confidential) of XMDO most always succeeded in returning a local optimum within a reasonable computation time. However, with regards to the potential presence of multimodality in the design space and the need to implement multi-starting for the research of a global optimum with local-based methods, different local and global derivative-free algorithms have additionally been tested; requiring minor modifications of XMDO as it was initially developed to handle different optimisers and penalty methods.

Of the NLopt gradient-free algorithms handling variable bounds, the BOBYQA [100] and the Spblx (Based on Subplex [115]) methods have been tried. Even from a bad starting point, BOBYQA was most often able to quickly find a feasible solution (i.e. with constraints satisfied) but which was rarely optimum. Oppositely, when starting from a relatively good solution (feasible or not), Spblx drastically improved it. Because these methods are still local, global optimisation algorithms have also been looked at: the Differential Evolution [129] from Scipy, a Clearing method from Sareni et al. [116] recoded in Python during the thesis, and the CMA-ES [52] from its Python distribution. While the well known Differential Evolution method was hardly able to find a feasible solution, the Clearing method always managed to do it. Yet, the Clearing method suffered from a very low rate of convergence for this optimisation problem that was not acceptable. Finally, the CMA-ES was found to be the most robust and efficient algorithm of these global methods, with a much cheaper computational cost than the two others.

Afterwards, a comparison of CMA-ES and L-BFGS-B for this optimisation problem demonstrated that the CMA-ES solution was at least as good as the solution returned by a multi-started L-BFGS-B, and most often better. In addition, running the local method Spblx with the CMA-ES solution as a starting point enabled to further improve the solution in the most complicated cases such as with the hybrid aircraft configurations. The combination of CMA-ES and Sbplx has finally been selected to carry out the optimisation studies of the thesis.

Because CMA-ES and Sbplx are only able to solve unconstrained optimisation problems, an external penalty method is used for the chosen optimisation strategy and is presented next. Let \overline{f} be the normalised objective function (i.e. f/f_{ref}) and $x \in \mathbb{R}^n$ the vector of design variables. Also, let M and R be respectively one of the missions and one of the constraints (e.g. from Table 5.3 for the reference aircraft) that are considered in the optimisation problem. The new objective function π is then written such as in Equation A.4, where K is a global penalty factor and $\phi(R, x)$ is a penalty function. The general definition of $\phi(R, x)$, where the constraint R is time-dependent, is given by Equation 5.12.

$$\pi(x) = \overline{f}(x) + K \sum_{M} \sum_{R} \phi(R, x)$$
(5.11)

$$R: \forall t, h(x,t) \le h_R(x,t), \qquad \phi(R,x) = \sum_t \max\left[0, \frac{h(x,t) - h_R(x,t)}{h_R(x,t)}\right]$$
(5.12)

Unlike the implementation of penalty methods with gradient-based algorithms (Appendix A), a very high value can be given to K relativity to \overline{f} in order to force the constraints to be met.

5.4 Optimisation results

5.4.1 Optimisation cases and design variables

In this chapter two different sets of missions are compared:

- S1 includes five missions: a nominal mission at maximum range (400 nm), a OEI take-off, a rejected take-off, a landing at MLW and a OEI ceiling.
- S2 includes six missions: the missions of S1 plus a nominal mission of 200 nm. For the evaluation of the objective function (Equation 5.10), and based on the flight distribution of Figure 3.5, weighting factors of 0.7 and 0.3 are assigned to the 200 nm and 400 nm missions, respectively.

The design variables used for these two optimisation cases are shown in Table 5.4. For these sets of missions, the maximum payload being equal to the nominal payload, the aircraft MTOW is defined through Equation 5.13, where F_{400} is the fuel loaded to fly the 400 nm mission. In addition, the MLW is defined by Equation 5.14 for simplicity (i.e. MLW includes 10% fuel margin). As a consequence, the aircraft characteristic weights (i.e. MTOW, MZFW, MLW) are all defined from MZFW and F_{400} , explaining why MTOW and MLW are not in the set of design variables of Table 5.4.

$$MTOW = MZFW + F_{400} \tag{5.13}$$

$$MLW = MZFW + 0.1(MTOW - MZFW)$$

$$(5.14)$$

The group of global design variables refers to the design variables that could have been duplicated for each mission of the optimisation case. The rationale of this group is of course to limit the number of design variables.

5.4.2 Convergence rate

Figure 5.5 shows the convergence rates of two optimisation runs for S1 and S2. As expected, case S1 requires less evaluations of the objective function than S2 to converge because of the smaller number of design variables (i.e. 26 for S1 and 37 for S2). The Sbplx algorithm was not able to further improve the solution found by CMA-ES for S1, and thus, is not shown in the graph. For S2, the improvement brought by Sbplx is in the order 0.25% of the optimum solution, showing that CMA-ES was already able to find a very good solution. Also, while the switch from CMA-ES to Sbplx was done manually, it could have been automated according to the CMA-ES convergence rate in order to save computation time. About time, the complete evaluation of S1 with a serial running of the missions took between 7 to 10 seconds and between 13 to 16 seconds for S2 on a high-end workstation computer². Of course, these figures depends on the number of points considered in mission discretization. Finally, although the results of only two optimisation runs are provided in this chapter, it should be noted that the optimum solutions are highly reproducible even from different starting points.

 $^{^2}$ Intel Xeon E5-1630 V4 (4 Core, 3.7-4.0GHz Turbo, 10Mo, 35W/C, 2400 Mhz), 32Go RAM DDR4 ECC 2400 MHz



Fig. 5.5 Convergence rates for S1 and S2

5.4.3 S1 versus S2

As can be seen from Table 5.4, optimising the reference aircraft according to S1 or S2 results in a fairly similar aircraft. Indeed, all the design variables relating to the aircraft geometry are very close from one set of missions to the other, and so are the weights summarised in Table 5.5. This is actually not surprising since even for S2 the 400 nm mission drives the aircraft characteristic weights that are used in the evaluation of the critical scenarios, which, in turns, constraint the aircraft component sizes. This observation confirms the rationale of the classical OAD process (Chapter 3), based on a single design mission for sizing the structure, when using a single source of energy.

Regarding the wing span, it is set to the maximum value by the optimiser to benefit from a high aspect ratio, and, consequently, from a low induced drag. However, it should be noted that the wing models of the thesis does not account for limiting criteria on the aspect ratio that may arise from buckling or minimum allowable thicknesses for examples. In particular, the wing mass is most probably underestimated from Equation 5.3 for wings of high aspect ratios.

From the main outputs of the missions depicted in Table 5.6, one can see that the TOFL and the landing speed constraints (i.e. R_{TOFL} and R_{V_L}) are active for the OEI take-off and the approach at MLW, respectively. Coming back to Table 5.4, it can be noticed that optimum flap settings are not only similar between S1 and S2, but also to those of ATR72 (i.e. 15° for take-off and 30° for landing). The difference in OEI ceiling Mach number can be simply explained by the fact that the OEI ceiling case is

not a sizing mission in both S1 and S2. The other major differences between S1 and S2 relate to the trajectory that can be further analysed thanks to the output plots of Figure 5.7.

In both cases, the 400 nm mission is flown at the maximum enabled altitude (i.e. 30,000 ft). Note that this upper bound was defined according to the turboshaft model whose validity is not guaranteed above this altitude. If the climb and descent phases differ between S1 and S2, so does the mission duration. The shorter time on 400 nm for S2 can be associated with the steeper descent but also with the slightly higher cruise speed. Moreover, the optimum cruise speeds are of the same order as those of current turboprop aircraft of this category. On the aerodynamic side, the high aspect ratio of approximately 15 (Figure 5.6) combined with the 30,000 ft cruise altitude enable these aircraft to achieve a high lift-to-drag ratio of almost 18.

Finally, the small higher fuel consumption for S2 along the longest mission can also be linked to the lower weight of F_{400} in the objective function than F_{200} .



Fig. 5.6 Geometry of the reference aircraft optimised according to S1 (m)

5.4.4 Deeper analysis of S1

This section analyses more in details the reference aircraft optimised according to S1. In particular, some of the most interesting outputs for the nominal mission (Figure 6.7)

	Name	Notation	Unit	Min.	Max.	S1	S2	$\Delta\%$ vs. S1
	MZFW	MZFW	kg	17000.0	35000.0	19174.4	19148.4	-0.1%
raft	Wing span	b_W	m	13.0	27.0	27.0	27.0	0.0%
	Wing chord	c_{r_W}	m	0.80	7.00	2.04	2.09	2.7%
	HTP chord	c_{r_H}	m	0.70	3.00	1.66	1.67	0.8%
irc	VTP chord	c_{r_V}	m	0.80	8.00	3.31	3.30	-0.3%
A	Propeller diam.	$\dot{D_p}$	m	3.00	5.50	3.83	3.87	1.0%
	Gearbox diam.	D_{qbx}	m	0.55	0.90	0.75	0.75	0.0%
	Turboshaft diam.	$\tilde{D_{gt}}$	mm	650	850	763	760	-0.3%
	Take-off rating	r_{TO}	-	0.10	1.00	0.99	0.99	0.6%
	Take-off slope	γ_{TO}	0	1.37	10.00	4.59	4.62	0.6%
bal	Take-off conf.	δ_{TO}	0	0.00	80.00	14.79	14.02	-5.2%
Glo	Climb unif. fact.	u_{CL}	-	0.01	0.99	0.03	0.05	82.6%
\cup	Landing conf.	δ_{LG}	0	0.00	80.00	29.48	27.96	-5.2%
	Descent unif. fact.	u_{DSC}	-	0.01	0.99	0.93	0.76	-18.9%
	OEI ceiling Mach	M_{OEI}	-	0.20	0.45	0.20	0.27	30.6%
	Fuel loaded	F_{400}	kg	200.0	3000.0	807.7	821.0	1.7%
	Climb vert. speed	Vz_{CL_i}	m/s	0.00	20.00	9.49	9.51	0.2%
Ъ	Climb vert. speed	Vz_{CL_f}	m/s	0.00	15.00	3.04	2.24	-26.2%
nn	Climb duration fact.	dt_{CL}	-	0.35	0.65	0.45	0.43	-4.3%
00	Cruise Mach	M_{CR}	-	0.35	0.60	0.43	0.40	-6.5%
ul 4	Cruise altitude	z_{CR}	m	5000.0	9140.0	9135.8	9046.1	-1.0%
inŝ	Cruise start fact.	x_{CR_i}	-	0.05	0.45	0.21	0.43	105.2%
om	Cruise end fact.	x_{CR_f}	-	0.55	0.95	0.78	0.55	-29.3%
Z	Desc. vert. speed	Vz_{DSC_i}	m/s	-15.00	0.00	-5.53	-5.61	$1.5 \ \%$
	Desc. vert. speed	$V z_{DSC_f}$	m/s	-15.00	0.00	-1.10	-3.09	179.7%
	Desc. duration fact.	dt_{DSC}	-	0.35	0.65	0.44	0.44	-0.5%
	Fuel loaded	F_{200}	kg	200.0	3000.0		480.2	
	Climb vert. speed	Vz_{CL_i}	m/s	0.00	20.00		9.34	
Г	Climb vert. speed	Vz_{CL_f}	m/s	0.00	15.00		0.06	
Nominal 200 nm	Climb duration fact.	dt_{CL}	-	0.35	0.65		0.42	
	Cruise Mach	M_{CR}	-	0.35	0.60		0.44	
	Cruise altitude	z_{CR}	m	5000.0	9140.0		9120.1	
	Cruise start fact.	x_{CR_i}	-	0.05	0.45		0.31	
	Cruise end fact.	x_{CR_f}	-	0.55	0.95		0.84	
	Desc. vert. speed	Vz_{DSC_i}	m/s	-15.00	0.00		-9.58	
	Desc. vert. speed	$Vz_{DSC_{f}}$	m/s	-15.00	0.00		-2.26	
	Desc. duration fact.	dt_{DSC}	-	0.35	0.65		0.41	

Table 5.4 Design variables: bounds and solutions for S1 and S2 $\,$

	S1	S2	$\Delta\%$ vs. S1
Total structure ^{a}	5545.2	5538.9	-0.1%
Propeller (x2)	350.7	358.0	2.1%
Gearbox $(x2)$	263.5	262.0	-0.6%
Turboshaft $(x2)$	599.0	582.7	-2.7%
Nacelle $(x2)$	364.3	356.2	-2.2%
Total propulsion	1577.6	1558.9	-1.2%
Equipments	5401.4	5401.0	0.0%
OEW	12524.2	12498.8	-0.2%

Table 5.5 Weight breakdown for the reference aircraft $\left(\mathrm{kg}\right)$

Mission	Output	S1	S2	$\Delta\%$ vs. S1
OEI talaa aff	FL_{out}	1399.9	1400.0	0.0%
OEI take-on	V_2	62.9	62.7	-0.3%
Rejected take-off	FL_{out}	1352.5	1343.2	-0.7%
	$V_{L_{out}}$	58.0	58.0	0.0%
Approach at MLW	α	4.14°	4.23°	2.2%
	r_{T_2}	0.31	0.30	-3.1%
OFL coiling	α	5.47°	5.28°	-3.5%
OEI tennig	r_{T_2}	0.71	0.73	3.3%
	TOW	19981.9	19969.4	-0.1%
	LW	19189.7	19163.8	-0.1%
	ZFW	19174.2	19148.4	-0.1%
Nominal 400 nm	$fuel_{req}$	807.7	818.6	1.3%
	FL_{out}	1114.6	1111.5	-0.3%
	V_2	64.0	62.8	-1.9%
	$V_{L_{out}}$	57.7	57.7	0.1%
	TOW		19628.5	
	LW		19163.8	
	ZFW		19148.4	
Nominal 200 nm	$fuel_{req}$		480.2	
	FL_{out}		1069.0	
	V_2		62.3	
	$V_{L_{out}}$		57.7	

 a Fuselage+Wings+Vertical stabiliser+Horizontal stabiliser

Table 5.6 Examples of mission outputs for optimum solutions (SI units)

and for the OEI take-off (Figure 5.9) are discussed.

As can be seen from Figure 6.7, the turboshaft component is sized by the climb phase and is used near maximum power capability during take-off (Table 5.4).

All along the descent, the turboshaft rating is close to its minimum value (i.e. 0.05). During the last part, the aircraft even has to generate braking drag (Equation 4.34, Section 4.5.2) to follow the optimum trajectory, which does not seem to be the most energy efficient descent at first sight. However, integrating the power dissipated (i.e. braking drag times aircraft speed) over time yields to a dissipated energy of 89.7 MJ, representing 0.26 % of the total energy consumed based on a 43 MJ/kg fuel lower heating value. Even if this very low ratio does not justify the implementation of energy recovering systems in descent, it reopens new opportunities for hybrid-electric aircraft (Chapter 6). In addition, this solution found by the optimiser most probably results in lower overall fuel burn than with a longer descent flown at idle rating without braking drag regarding the simple parametric curves that are used to model the flight path.

On the flap configuration control, flaps are quickly retracted after the take-off to benefit from a higher lift-to-drag ratio in clean configuration, resulting in a large increase of the angle of attack which is still below the limit. During the descent, the clean configuration is used until the aerodynamic stall constraint (R_{AoA} , Table 5.3) is active, coinciding with the first point of the flap extension phase prior to landing.

Finally, as can be seen from the last plot of Figure 6.7, the horizontal tail volume gets closer to the sizing constraint (R_{HTP} , Table 5.3) at the approach phase. This constraint is actually active for the mission modelling the approach at MLW. Note that a default (and non-physical) value is given to the horizontal tail volume on the ground.

The OEI take-off mission involves the maximum input powers for the gearboxes and the propellers. As can be seen from Figure 5.9 the gearbox is exactly sized for the take-off. Let us recall that the gearbox input power depends not only on the turboshaft size, but also on the take-off power rating r_{TO} which is slightly less than unity (Table 5.4). The propeller diameter, however, has not been chosen according to the maximum power constraint ($R_{SubComp}$, Table 5.3). The resulting propeller diameter is therefore a compromise between weight and efficiency.

For this conventional aircraft, the fin is also sized by OEI take-off. Indeed, the controllability criterion expressed through R_{VTPc} (Table 5.3) becomes active right after lift-off, where the thrust is maximum and the speed low. Note that because the

landing gear contribution to yaw equilibrium is not modelled, a default value satisfying the constraint is also given to V_{V_c} on the ground.

Figure 5.9 also shows the evolution of external forces acting on the aircraft along this mission. At the failure time of Turboshaft₁, Propeller₁ is feathered creating negative thrust (i.e. drag). At the same time the vertical tail counters the yaw moment from the operating engine and adds trim drag accordingly. However, note that this drag component may be underestimated because of the simple aerodynamic model used for its evaluation. As a result of the lower mass flow blowing the wing, the loss of a propulsor induces a lift reduction, but also, surprisingly, a drag reduction. In other words, even if the total drag shown in the graph does not include the negative thrust of the feathered propeller, the additional forces created by both the feathered propeller and the vertical stabiliser seem to not overcome the drag reduction of the wing-propeller interaction. Still, the lift is less than with all engine operating, and, above all, the total thrust is reduced by half.

Finally, the effect of the engine failure on the aircraft acceleration is clearly noticeable on the last plot. Although the ground effect and the landing gear drag are neglected in the aerodynamic model, the simple low-speed models seem satisfactory for the purpose of this thesis.



Fig. 5.7 Output plots for nominal missions of optimum solutions



Fig. 5.8 Output plots for the nominal mission of S1



5.4.5 Comparison with ATR72

Based on the data of [4], it can be seen that the reference aircraft of this chapter burns 22 % less fuel than an AT72-600 on a 200 nm mission (S2, Table 5.6). This large fuel burn reduction comes from several reasons.

First, the top level aircraft requirements (TLARs) are different: the ATR72 can carry a 850 kg heavier maximum payload, and can cover a maximum range with its nominal payload that is twice that of the reference aircraft. In addition, all other parameters being equal, the reference aircraft is lighter in flight because fuel reserves are not taken into account. All of these results in lower characteristic weights for the reference aircraft: -8 % MZFW, -14 % MLW, -11 % MTOW.

Secondly, both the aerodynamic model and the turboshaft model include technology improvements versus ATR72 whose maiden flight dates back to October 1988.

Thirdly, the one-g stall speed calculation in the thesis includes the positive thrust effect which is not allowed by current aviation regulations to which ATR72 is subjected. The reference aircraft then benefits from improved low-speed performance enabling to reduce the wing area to meet the same landing speed and take-off field length requirements.

Finally, the reference aircraft flies at a higher cruise altitude than the ATR72 and then benefits from a better aerodynamic efficiency. As an example, the reference aircraft improves its lift-to-drag ratio by 18 % when flying at 30,000 ft instead of 20,000 ft at the same Mach number.

Chapter 6

Hybrid-electric aircraft optimisations

Contents

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This chapter deals with the optimisation of the different hybrid-electric aircraft configurations presented in Chapter 2: the turboelectric **TE4**, the partial turboelectric **PT8**, the parallel-hybrid **PH** and the all-electric **AE**. In the description of each optimisation case, a particular attention is given to the assumptions relating to the propulsion system modelling and the constraints considered in the optimisation problem. Throughout this chapter, results are compared to the conventional aircraft **REF** optimised in Chapter 5.

First, the models and assumptions applying to all the aircraft configurations are exposed, and the different missions considered in the optimisation cases are presented.

Then, the component models that were developed and implemented for the purpose of electrified propulsion system studies are addressed. For the sake of clarity, the electrical machine model is placed in Appendix G. In order to include the effect of technology improvements with time, two technology levels are defined according to the service entry date of the aircraft: 2025 and 2030+. The third section presents and analyses the optimisation results for the turboelectric **TE4** and the partial turboelectric **PT8**, the two architectures that were selected in Chapter 2 to better understand the potential aerodynamic improvements identified in Chapter 1.

The fourth section deals with the optimisation of the parallel-hybrid that was defined in Chapter 2 for the study of fuel savings brought by new energy management strategies. In fact, two sets of constraints differing from each other by the allowed state of charge of the batteries at the end of the nominal mission are used, resulting in two optimisation cases named **PH** and **PH-R**.

Afterwards, the results of the previous sections are all compared in terms of fuel and energy consumptions versus the reference aircraft, and conclusions are drawn.

The last section is dedicated to the all-electric aircraft, and aims at identifying the minimum specific energy required for batteries as a function of the aircraft design range. A trade study is also carried-out in accordance with the service entry date for the other electrical components.

Notation: like in Chapter 5, the subcomponents of the propulsion chain are named components, and the notion of *priority subcomponent* is still used in reference to the generic propulsion system model presented in Chapter 4 (Section 4.5.2).

6.1 General optimisation case

6.1.1 Airframe modelling

For the different aircraft configurations of this chapter, the aircraft model is composed of the same estimation models as the ones used for the reference aircraft optimisation introduced in Chapter 5. Additional component models for the studied electrified propulsion systems are presented in Section 6.2. The same design variables relating to the airframe are therefore considered, except the wing span that is fixed to the maximum value (27 m) in order to limit the number of design variables. Despite the small weight penalty returned by the wing mass model when increasing the aspect ratio at iso-area, the optimiser would drive the wing span towards the upper bound for all the optimisation cases in order to minimise the lift-induced drag. Considering the wing span as design variable would make sense with a more detailed wing model including limitations (e.g. mechanical, manufacturing) on the aspect ratio.

6.1.2 Missions and objective function

As pointed out in the previous chapter, optimising the reference aircraft according to the set of missions S1 (including a single nominal mission) or S2 (including two nominal missions) results in the same aircraft with regard to the objective function and the set of constraints of the optimisation problem. This would hold true for all the aircraft configurations using a single type of energy source since the structural weights are only driven by the amount of energy required to fly the longest mission. However, for the parallel hybrid architecture using energy from fuel and batteries, the aircraft design would most probably be different for the two sets of missions, as discussed in Chapter 3.

While the reference aircraft optimisation case features the smallest number of design variables of all the studied configurations, its convergence time for S2 took between three to four days with a non-parallel CMA-ES algorithm on a high-end workstation computer. Because the implementation of a parallel version of CMA-ES was not possible regarding the time-frame of the thesis, it was decided to limit all the optimisation cases to S1. This assumption is of course taken into account in the conclusions drawn from the optimisation results of the parallel hybrid.

Except for the all electric aircraft (Section 6.6), the objective function is still the fuel consumption along the nominal mission as defined in Chapter 5. Finally, all the following results are compared to the reference aircraft REF optimised according to S1 in the previous chapter.

6.2 Additional component models

This section presents the different component models that were developed and implemented for the purpose of electrified propulsion system studies. The next component descriptions employ the notation \vec{T}_C , introduced in Section 5.1.4, for the thrust force at component level. Note that the models and the associated assumptions presented in Chapter 5 for the turboshaft engine, the non-propulsive load, the power gearbox, the nacelle and the variable-pitch propeller are also used in the following studies. For the electrical components (electric machines, cables, power electronics and batteries), two level of technologies are defined regarding the potential service entry date: EIS¹2025 and EIS2030+.

¹Entry Into Service
6.2.1 Electric machine

The permanent magnet synchronous machine (PMSM) model of Appendix G is used to model electrical machines. While the PMSM technology does not suit to all the selected architectures as discussed in details in Chapter 2, the use of this model for all of them provide optimistic efficiencies and torque densities against other machine technologies; which is taken into account in the analyses of the optimisation results. The technology improvement relative to the EIS assumption is modelled through the cooling factor k_C (Appendix G) that is increased from 1.0 to 1.6. The associated maximum power densities and efficiencies in continuous operation are summed up in Table 6.1. For simplicity, an electrical machine is supposed to have the same efficiency in both motor and generator modes.

Geometry. The motor size is modified thanks to its diameter D_m and length L_m , that are chosen as design variables. The annular geometry assumption of the PMSM model is selected, and the specific speed of the machine is taken equal to the speed of maximum efficiency ω_{opt} (Appendix G).

Mass. The mass is calculated with the scaling law of Appendix G.

Propulsion. The maximum power P_{max} that can be applied to the electric machine is taken equal to the maximum power in continuous operation defined as $\omega_{Copt} T_{Copt}$ in the appendix. Also, such component does not generate external forces and $\vec{T}_C = 0$. Finally, the component efficiency is supposed to be constant and equal to the maximum efficiency in continuous operation (i.e. η_{Copt} in Appendix G). Note that the proposed PMSM model is also able to capture the effect of torque and speed variations on the efficiency and can be used in more detailed studies for which (torque, speed) and (current, voltage) are exchanged between components.

6.2.2 In-line gearbox

The in-line gearbox component was developed to model geared drives of electric-driven propellers or turbo-generators which may require smaller gear ratios than turboprop engines.

Geometry. The in-line category gearbox is defined in Appendix F. The associated geometrical model is used, and the gearbox diameter D_{gbx} is selected as design variable. The gear form factor Ψ is fixed to a typical value but could be taken as design variable

if space allocation is one of the main concerns. The gear ratio M_0 is taken equal to the ratio of the specific rotational speeds (e.g. max. speed for a propeller, optimum speeds for electric machines and turboshaft engines) of the input and output components.

Mass. The mass is evaluated according to the mass model of Appendix F.

Propulsion. The component efficiency is calculated from the simple performance model of Appendix F. The maximum power P_{max} that can flow through the gearbox is taken equal to the product of the maximum input torque capability $T_{in_{max}}$, given by the same appendix, and the specific rotation speed of the input component (in the sense of reduction gearing i.e. $M_0 > 1$). Finally, the thrust force $\vec{T}_C = 0$.

6.2.3 High-lift propeller

Geometry. Like variable-pitch propellers, high-lift propellers feature six blades. The propeller size is controlled through its diameter D_p . The specific rotation speed of the component is taken equal to the maximum propeller speed N_{max} defined in Appendix E.

Mass. The mass estimation model of Appendix E is used.

Propulsion. In the next optimisation cases (Section 6.3.2), high-lift propellers are defined as *priority subcomponents* (Section 4.5.2). Thrust and output power are therefore controlled by a local power rating r_{HLP} . The maximum power P_{max} that can be absorbed by the propeller is first evaluated from the formula of Appendix E. The propeller force is defined as $\vec{T}_C = T\vec{e}_{x_B}$, where T is the value returned by the performance model of the same appendix for a shaft power equal to $r_{HLP}P_{max}$. The output power sent to the downstream component is $P_{out} = -r_{HLP}P_{max}$.

6.2.4 Cables

Cables are characterised by the type of current, AC or DC, and the voltage level² V_c which depends on the EIS assumption (Table 6.1). The simple parametric model that is used in the thesis is described next.

²Even for AC cables, V_c is defined as the DC bus voltage (i.e. the DC bus voltage that would exist with power electronics connected to the AC bus)

Geometry. The length L_c of a cable component is defined according to the locations of its input and output components, and the diameter D_c of a single phase is used as design variable.

Mass. The mass M_c of this component is evaluated from Equation 6.1, where ρ_c is the cable density and A_c is the section area of a single phase. Cables are supposed to be made of copper and ρ_c is 9000 kg m⁻³.

$$M_c = \begin{cases} 2\rho_c A_c L_c & \text{if DC} \\ 3\rho_c A_c L_c & \text{if AC} \end{cases}$$
(6.1)

Propulsion. The maximum power P_{max} that can be applied to the cable is defined by Equation 6.2, where $j_{c_{max}}$ is the maximum current density equal to 6 A/mm².

$$P_{max} = \begin{cases} V_c j_{c_{max}} A_c & \text{if DC} \\ 3 \frac{V_c}{2\sqrt{2}} j_{c_{max}} A_c \times 0.9 & \text{if AC} \end{cases}$$
(6.2)

In this model, power losses only include joule losses P_J that are calculated from Equation 6.3, where R_c is the resistance of a single phase. The equation for the AC case assumes that the frequency is constant and equal to its maximum value.

$$P_{J} = \begin{cases} 2R_{c}(P_{in}/V_{c})^{2} & \text{if DC} \\ 3R_{c} \left(\frac{P_{in}}{3V_{c}/(2\sqrt{2})0.9}\right)^{2} & \text{if AC} \end{cases}$$
(6.3)

The output power P_{out} of the cable component is estimated according to Equation 6.4, and $\vec{T}_C = 0$ for this component.

$$P_{out} = P_{in} - P_J \tag{6.4}$$

6.2.5 Power electronics

Converters are used in the parallel-hybrid (PH) and all electric (AE) architectures. The converter model is probably one of the simplest model of this chapter that uses a specific power \overline{P}_{pe} and a constant efficiency η_{pe} which are both provided by the EIS assumption (Table 6.1). **Geometry.** The converter is modelled by a box of volume $0.2 \times 0.7 \times L_{pe}$ m³, where the length L_{pe} is used as design variable to modify the maximum power capability of the component. Contrary to motors and gearboxes that must be carried inside a nacelle to drive a propeller, and whose dimensions can change the aircraft aerodynamics, more freedom is given to the designer on the shape of this component but also on its location on the aircraft. Therefore, the size of the power converter is assumed to have no influence on the aircraft drag, and a virtual density $\rho_{v_{pe}}$ (3000 kg m⁻³) serves as a link between the maximum power capability P_{max} and L_{pe} , as expressed in Equation 6.5.

$$P_{max} = \rho_{v_{pe}} \overline{P}_{pe} (0.2 \times 0.7 \times L_{pe}) \tag{6.5}$$

Mass. The mass of this component is simply evaluated from the specific power \overline{P}_{pe} and P_{max} (see Table 6.1).

Propulsion. The calculation of the component output power is straight forward thanks to the constant efficiency assumption. Here again, the thrust force $\vec{T}_C = 0$.

6.2.6 Battery

From the EIS assumption (Table 6.1), the battery model uses a specific energy \overline{E}_b (in Wh/kg) and the efficiency η_b which is the same for charging or discharging.

Geometry. The battery size is controlled by the length L_b of the box (volume $0.5 \times 1.0 \times L_b \text{ m}^3$) modelling the battery. Like power electronics, the battery is supposed to fit inside the aircraft whatever its geometry, and L_b is used to drive the battery capacity E_{max} according to Equation 6.6, where ρ_{v_b} is a virtual density of 2500 kg m⁻³.

$$E_{max} = \rho_{v_b} \overline{E}_b (0.5 \times 1.0 \times L_b) \tag{6.6}$$

Mass. The battery mass is calculated from the specific energy \overline{E}_b and E_{max} (see Table 6.1).

Propulsion. The battery is defined as a *priority subcomponent* (Section 4.5.2) and its output power is driven by a local power rating r_B . The maximum power P_{max} (in W) that can be received or sent is defined by Equation 6.7, and depends on the sign of r_B (i.e. discharge: $r_B > 0$, and charge: $r_B < 0$). As can be seen from this equation, maximum rates of discharge and charge are 5C and 3C, respectively. The battery output power is $P_{out} = r_B P_{max}$.

$$P_{max} = \begin{cases} 5E_{max}10^3 & \text{if } r_B > 0\\ 2E_{max}10^3 & \text{otherwise} \end{cases}$$
(6.7)

The battery state of charge SOC is defined by Equation 6.8, where E(t) is the energy stored in the battery at time t. By defining dt as the time step between two evaluation points of the mission, the stored energy E at t + dt is given by Equation 6.9.

$$SOC(t) = \frac{E(t)}{E_{max}}$$
(6.8)

$$E(t+dt) = E(t) - \begin{cases} \frac{P_{out}(t)dt}{\eta_b} & \text{if } r_B > 0\\ \eta_b P_{out}(t)dt & \text{otherwise} \end{cases}$$
(6.9)

For this component, the external force \vec{T}_C is also zero. The different constraints of the optimisation problem relating to the implementation of batteries are described in Section 6.4.1.

6.2.7 Summary of electrical component assumptions

Table 6.1 summarises the main assumptions considered in the electrical component models. Let us recall that the figures for the electric machine are outputs of the PMSM model of Appendix G, obtained with a cooling factor of 1.0 and 1.6 for EIS2025 and EIS2030+, respectively.

		EIS2025	EIS2030+
Electric machine	Specific power Efficiency	$\begin{array}{c} 7 \hspace{0.1cm} \mathrm{kW/kg} \\ 96 \hspace{0.1cm}\% \end{array}$	$\frac{11 \text{ kW/kg}}{98.5 \%}$
Power electronics	Specific power \overline{P}_{pe} Efficiency η_{pe}	$15 \ { m kW/kg}$ 99 %	$20 \ {\rm kW/kg}$ $99.5 \ \%$
Battery	Specific energy \overline{E}_b Max. ch./disch. C Efficiency η_b	280 Wh/kg 2/5 90%	380 Wh/kg 2/5 95 %
Cable	DC bus voltage V_c	540 V	$1500 \mathrm{V}$

Table 6.1 Electrical component assumptions according to EIS

Finally, it should be pointed out that no cooling component has been taken into account. Although it can be assumed that the mass of the electrical component includes that of the cooling system, the cooling drag is missed. This can actually be improved by using a cooling drag model returning an external force \vec{T}_C as a function of the flying conditions and the power losses of the component.

6.3 Aerodynamic improvements: TE4 and PT8

This section focuses on the study of the two architectures enabling potential benefits on aircraft aerodynamics: the turboelectric TE4 and the partial turboelectric PT8.

6.3.1 Turboelectric TE4

Propulsion

The orientation of the propulsion system in XMDO of the TE4 aircraft is shown in Figure 6.1 (N=4). In this architecture, all the gear reducers are in-line gearboxes. Similarly to the reference aircraft REF, the local power rating r_{T_2} of Turboshaft₂ is chosen as r_{NR} and the minimum power rating $r_{NR_{min}}$ (i.e. idle rating) is 0.05. Here again, the relation $r_{T_1} = r_{T_2}$ is used in nominal operation, and r_{T_1} is zero during single engine taxiing and engine failure scenarios. The local power rating r_{NPL_i} of the non-propulsive load NPL_i is still 1 while $r_{T_i} > 0$, and zero otherwise.

Actually, this optimisation case uses the same assumptions on control laws (local power ratings and flap setting) along the different missions of S1 as for the reference aircraft (Section 5.3.1). The single difference relies on the control of propellers that are never feathered, since only the failure of Turboshaft₁ is considered in the missions of S1.

Constraints

The TE4 aircraft is optimised according to the same constraints as the REF aircraft case, provided in Table 5.3.

Optimisation results

While the optimisation run of the REF aircraft with S1 involves 26 design variables (Aircraft: 8, Global: 6, OEI ceiling: 1, 400 nm mission: 11; see Table 5.4), 30 design variables are considered in the optimisation of the TE4 configuration. The definitions of the design variables for the last three groups (Global, OEI ceiling and 400 nm mission) are the same as in the REF case, and the major differences relate to the Aircraft group



Fig. 6.1 Orientation of the turboelectric architecture TE

as can be seen from Table 6.2. For the sake of clarity, only the Aircraft group and three design variables of the Global group are shown in this table. In the Aircraft group, the cable diameter D_c refers to the sizing parameter of a cable connected to a motor, the diameter of the generator cable (Figure 6.1) being defined accordingly.

The resulting geometry is shown in Figure 6.2 for EIS2025 and the areas of lifting surfaces are summarised in Table 6.3. Unlike the vertical tail plane of the reference aircraft, the fin area of the TE4 is driven by the stability criterion (R_{VTPs} , Table 5.3) because the loss of one engine does not involve any yaw moment under the assumptions of this thesis. Hence, the vertical stabiliser area is greatly reduced versus the reference. However, despite the four propellers and the higher flap deflection at landing favouring

	Name	Notation	Unit	Min.	Max.	EIS2025	EIS2030+
	Wing chord	c_{r_W}	m	0.80	7.00	2.11	1.98
	HTP chord	c_{r_H}	m	0.70	3.00	1.76	1.67
	VTP chord	c_{r_V}	m	0.80	8.00	1.34	1.30
	Propeller diam.	D_p	m	3.00	5.50	4.33	4.40
ц.	Prop. gearbox diam.	D_{gbx_p}	m	0.55	0.90	0.34	0.32
raf	Motor diam.	D_{mot}	m	0.20	1.50	0.56	0.44
irc	Motor len.	L_{mot}	m	0.14	2.40	0.81	0.62
A	Cable diam.	D_c	mm	1.0	50.0	21.2	12.8
	Generator diam.	D_{gen}	m	0.20	1.50	0.95	0.89
	Generator len.	L_{gen}	m	0.14	2.40	0.99	0.62
	Gen. gearbox diam.	D_{gbx_q}	m	0.55	0.90	0.52	0.43
	Turboshaft diam.	\tilde{D}_{gt}	mm	650	850	769	769
al	Take-off conf.	δ_{TO}	0	0.0	80.0	17.7	15.0
do	Landing conf.	δ_{LG}	0	0.0	80.0	46.1	36.0
Ð	Take-off rating	r_{TO}	-	0.0	1.0	0.96	0.96

Table 6.2 Design variables: main solutions for TE4

the blown wing effect, the wing area for EIS2025 is still bigger than that of the REF aircraft. This can be simply explained by the tremendous weight penalty imparted by the electrified propulsion system, as shown in Table 6.4.



Fig. 6.2 Geometry of the TE4 aircraft (m) - EIS2025

	EIS2025	$\Delta\%$ vs. REF	EIS2030+	$\Delta\%$ vs. REF
Wings	50.1	$3.7 \ \%$	46.9	-2.9 %
Fin	1.9	-83.2 %	1.7	-84.2 %
Tail plane	9.2	13.0~%	8.2	$1.2 \ \%$

Table 6.3 Area summary for TE4 (m^2)

About the sizing conditions of the different components of the propulsion system, the turboshaft is still sized by the climb phase and the power constraint on the propeller is never active such as in the REF case. The propeller diameter is actually even larger than for the reference propulsion system, most probably to increase the blown portion of the wing. All the gearboxes and other electrical components are sized by the take-off power. Regarding gearboxes, the ratio is 2.68 (4.75, respectively) on the propeller side, and 8.5 (5.81, respectively) on the generator side for EIS2025 (EIS2030+, respectively). But let us recall that these figures fully depend on the PMSM model, which is a technology that is not likely to be implemented on this architecture for the generators (Chapter 2).

	EIS2025	$\Delta\%$ vs. REF	EIS2030+	$\Delta\%$ vs. REF
Total structure	5620	1.3~%	5559	0.3~%
Equipments	5505	1.9~%	5421	0.4~%
Propeller (x4)	898		713	
Prop. gearbox $(x4)$	226		225	
Motor $(x4)$	715		421	
Nacelle $(x4)$	297		192	
Cable (tot.)	592		216	
Generator $(x2)$	749		440	
Gen. gearbox $(x2)$	192		127	
Turboshaft $(x2)$	640		640	
Total propulsion	4309	173.1~%	2974	88.5 %
MZFW	22088	15.2~%	20560	7.2~%
Fuel burn F_{400}	906	12.2~%	846	$4.7 \ \%$

Table 6.4 Weight summary for TE4 (kg)

As can be seen from Table 6.3, the TE4 aircraft burns 12.2 % more fuel than the reference for EIS2025. To better understand this figure, let us analyse key performance data along the nominal mission.



Fig. 6.3 TE4 - EIS2025: main trajectory outputs vs. REF

The trajectory selected by the optimiser, and plotted in Figure 6.3, is fairly similar to that of the REF aircraft with the cruise altitude and cruise speed being respectively slightly lower and higher for the TE4. Along this trajectory, the TE4 configuration actually shows a higher lift-to-drag ratio than the REF aircraft (Figure 6.4) that mainly comes from the smaller vertical stabiliser. In addition, the TE4 benefits from a higher specific fuel consumption (Figure 6.4) due to its bigger engines. However, the much larger aircraft weight combined with the lower efficiency of the power chain between propellers and prime movers results in higher power demands for the TE4 aircraft (Figure 6.5). The greater power demands, though associated to the better specific fuel consumption of the turboshaft engines, lead to a less fuel efficient aircraft (Figure 6.5).

With a 4.7% larger fuel burn than the reference, the expecting technology improvements on electrical components for EIS2030+ do not enable TE4 to win against REF.



Fig. 6.4 TE4 - EIS2025: aerodynamic and engine performance data vs. REF



Fig. 6.5 TE4 - EIS2025: power profile and fuel burn vs. REF

Focus on blown wing effect

On the Breguet 941 [26], one of the most famous quad-turboprop aircraft featuring blown flaps, flaps can be extended up to approximately 90°. While the upper bound for the landing flap setting δ_{LG} was set to 80°, in line with the aerodynamic model of the thesis, the maximum value selected by the optimiser is only 45° (Table 6.2). Although the Breguet 941 features a more complex flap system than TE4, the low flap deflection value had to be understood.

First, the one-g stall speed for the EIS2025 solution was evaluated (according to the definition of the manuscript, Section 4.4.3) for different flap settings at MLW, 50 ft, ISA, with Turboshaft₁ inoperative. The results, shown in Figure 6.6, demonstrate that choosing a higher flap deflection angle would enable to decrease the one-g stall speed. In other words, based on the landing speed criteria (R_{V_L} , Table 5.3) which is active for the TE4 aircraft, increasing δ_{LG} would result in a smaller wing area.



Fig. 6.6 TE4 - EIS2025: one-g stall speeds at MLW, 50 ft, ISA, Turboshaft₁ inoperative

Afterwards, a new optimisation case, named TE4-65°, was run with δ_{LG} removed from the set of design variables and fixed to 65°. As expected, the resulting wing area (Table 6.5) is much smaller than with the 46° flap extension. But the fuel consumption (Figure 6.7a) is also 0.44 % more for TE4-65° (910 kg) than for TE4 (906 kg). The take-off flap configuration is now 35° (18° for TE4) and the maximum angle of attack constraint (R_{AoA} , Table 5.3) is still active after the take-off and prior to the landing (Figure 6.7b). Regarding the aerodynamic efficiency, shown in Figure 6.7c, the liftto-drag ratio of TE4-65° is of course higher than that of TE4. In the same figure, one can notice the smaller specific fuel consumption of the TE4-65° engines in climb. Lastly, Figure 6.7c reveals the much larger power demand along the climb path for the larger flap setting, requiring the engines to be oversized. In spite of the engine efficiency gain resulting from its oversizing, the greater power demand induces a larger fuel consumption during climb as depicted in Figure 6.7a.

If the fuel consumptions of the two optimisation cases are eventually very close to each other, the TE4-65° case enabled to check the consistency of the results for the turboelectric configuration. It also illustrates the interest of evaluating the constraints for each time of the missions—requiring their complete simulation—when the occurrence of the sizing constraints are unknown; which can be the case when designing new aircraft concepts.

	TE4	$\Delta\%$ vs. REF	TE4-65°	$\Delta\%$ vs. REF
Wings	50.1	$3.7 \ \%$	42.5	-11.9 %
Fin	1.9	-83.2 %	1.6	-85.3 %
Tail plane	9.2	13.0~%	8.5	$4.5 \ \%$

Table 6.5 Area summary (m^2) for TE4 and TE4-65, EIS2025



6.3.2 Turboelectric PT8

Propulsion

The orientation of the PT8 propulsion system in XMDO is described in Figure 6.1. In this architecture, the gear reducers driving the high-lift propellers are in-line gearboxes and the power gearboxes, connected to the turboshaft engines, are of the turboprop category (Appendix F).

Similarly to the reference aircraft, the local power rating r_{T_2} of Turboshaft₂ is chosen as r_{NR} and the minimum power rating $r_{NR_{min}}$ (i.e. idle rating) is 0.05. The relation $r_{T_1} = r_{T_2}$ is also used in nominal operation, and r_{T_1} is set to zero in taxi or during engine failure scenario. The local power rating r_{NPL_1} is 1 while $r_{T_1} > 0$, and zero otherwise. Like for the reference propulsion system, Propeller₁ is instantaneously feathered if $r_{T_1} = 0$. In order to represent the action of switches mentioned in Chapter 2, Cable Gen₁ is additionally turned into failure mode if $r_{T_1} = 0$, preventing Cable₁ to Cable₈ from sending information to it.

On top of the assumptions about local power ratings $(r_{T_1} \text{ and } r_{T_2})$ and flap setting (δ) defined along the different missions of S1 (Section 5.3.1) for the reference aircraft, the local power ratings r_{HLP_i} of Table 6.6 are used. For the segments that are not included in this table: $\forall i, r_{HLP_i} = 0$. Finally, the local power ratings $r_{HLP_{taxi}}$ and $r_{HLP_{OEI}}$ are included in the set of design variables.

	Mission/Segment	$\forall i, r_{HLP_i}$
inal	$\begin{array}{c} {\rm Taxi}~({\rm out/out_H/in_H/in})\\ {\rm Take-off} \end{array}$	$r_{HLP_{taxi}}$ 1
Nom	Landing	airborne: 1 ground: 0
	OEI Take-off	1
	Rejected Take-off	$V \le V1: 1$ $V > V1: 0$
	Landing at MLW	1
	OEI ceiling	$r_{HLP_{OEI}}$

Table 6.6 Control laws for the high-lift propellers of PT8

Constraints

The PT8 aircraft is optimised under the same constraints as the REF aircraft, provided in Table 5.3.



Fig. 6.8 Orientation of the partial turboelectric architecture PT8

Optimisation results

The optimisation run of the TE8 aircraft involves 33 design variables (Aircraft: 13, Global: 6, OEI ceiling: 2, 400 nm mission: 12). The main differences with the REF case concern the Aircraft group, and the two additional ratings $r_{HLP_{taxi}}$ and $r_{HLP_{OEI}}$ involved in the evaluations of the nominal mission and the OEI ceiling, respectively. The main design variables and their optimum values are provided in Table 6.2. Here again, the cable diameter D_c refers to the sizing parameter of a cable connected to a motor, the diameter of the generator cable (Figure 6.8) being defined accordingly.

The output geometry is shown in Figure 6.9 for EIS2030+ and the areas of lifting surfaces are summarised in Table 6.8. The use of the high-lift propellers in case of

	Name	Notation	Unit	Min.	Max.	EIS2025	EIS2030+
	Wing chord	c_{r_W}	m	0.80	7.00	1.91	1.91
	HTP chord	c_{r_H}	m	0.70	3.00	1.65	1.64
	VTP chord	c_{r_V}	m	0.80	8.00	2.93	2.93
	Propeller diam.	D_p	m	3.00	5.50	4.14	4.14
	Power gearb. diam.	D_{qbx}	m	0.55	0.90	0.75	0.75
aft	Turboshaft diam.	D_{gt}	mm	650	850	767	767
CL	Generator diam.	D_{gen}	m	0.20	1.50	0.69	0.52
Aiı	Genrator len.	L_{gen}	m	0.14	2.40	0.49	0.36
	Cable diam.	\tilde{D}_c	mm	1.0	50.0	6.4	4.0
	Motor diam.	D_{mot}	m	0.20	1.50	0.20	0.20
	Motor len.	L_{mot}	m	0.14	2.40	0.14	0.14
	HLP gearbox diam.	$D_{qbx_{HLP}}$	m	0.01	1.50	0.15	0.14
	HLP diam.	D_{HLP}	m	0.50	1.50	0.78	0.76
	Take-off conf.	δ_{TO}	0	0.0	80.0	15.8	14.8
	Landing conf.	δ_{LG}	0	0.0	80.0	33.9	33.0
	HLP taxi rating	$r_{HLP_{taxi}}$	-	0.0	1.0	0.00	0.00
	HLP OEI rating	$r_{HLP_{OEI}}$	-	0.0	1.0	0.43	0.57

Table 6.7 Design variables: main solutions for PT8

failure of one engine reduces the thrust asymmetry versus the conventional propulsion system, and, therefore, enables the fin area (still sized by the controllability criterion R_{VTPc}) to be downsized by more than 20 %. It should be noted that the maximum shaft power ($r_{HLP_i} = 1$) commanded by the high-lift propellers, and resulting from the choice of the propeller diameter D_{HLP} (Section 6.2.3), is only 99 kW per propeller for EIS2025 and 93 kW for EIS2030+. With a total shaft power of less than 800 kW for the high-lift propellers (representing about 16 % of the take-off power all-engine operating), the wing area is still decreased by 6 % versus the REF aircraft. So, it can be said that the blowing effect outweighs the mass penalty (Table 6.9) which is much lower than for the turboelectric architecture.

	EIS2025	$\Delta\%$ vs. R	EF EIS2030+	- $\Delta\%$ vs. REF
Wings	45.4	-6.0 %	45.3	-6.2 %
Fin	8.7	-21.4 %	8.7	-21.7 %
Tail plane	8.1	-0.5~%	8.0	-2.2 %

Table 6.8 Area summary for PT8 (m^2)





	EIS2025	$\Delta\%$ vs. REF	EIS2030+	$\Delta\%$ vs. REF
Total structure	5592	0.8~%	5559	0.3~%
Equipments	5432	0.6~%	5421	0.4~%
Propeller (x2)	410	16.9~%	410	16.9~%
Gearbox $(x2)$	256	-2.9 %	257	-2.4 %
Turboshaft $(x2)$	630	$5.1 \ \%$	627	4.6~%
Nacelle $(x2)$	523	43.4~%	463	27.2~%
Generator $(x2)$	272		148	
Cable (tot.)	56		22	
Motor $(x8)$	130		89	
HLP gearbox $(x8)$	0		0	
HLPropeller $(x8)$	58		55	
HLP nacelle $(x8)$	43		32	
Total propulsion	2378	50.8~%	2104	33.4~%
MZFW	20056	4.6 %	19738	2.9 %
Fuel burn F_{400}	824	2.1~%	814	0.8~%

Table 6.9 Weight summary for PT8 (kg)

In addition to the optimistic electric machine model for this architecture, let us remind that the high-lift propeller performance model (Appendix E) also overestimates the propeller efficiency for a wide range of conditions as it is based on that of the variable-pitch propeller. Even so, the PT8 aircraft burns more fuel (Table 6.9) than the reference, including under EIS2030+ technology assumptions. Despite the higher aspect ratio (smaller wing area) and smaller fin area than REF, the high-lift prop nacelles adds wetted area, leading to a roughly equivalent lift-to-drag ratio to that of the reference aircraft (Figure 6.10). With lightly bigger engines, the PT8 turboshafts show a better specific fuel consumption in cruise (Figure 6.10). But the additional weights of the electrical components requiring more power in cruise, the PT8 looses the advantage of the prime mover efficiency and is less fuel-efficient as plotted in Figure 6.11.



Fig. 6.10 PT8 - EIS2030+: aerodynamic and engine performance data vs. REF



Fig. 6.11 PT8 - EIS2030+: power profile and fuel burn vs. REF

6.4 New energy management: PH and PH-R

This section deals with the optimisation of the PH aircraft configuration under two set of constraints differing from each other by the allowed state of charge of the batteries at the end of the nominal mission. The two optimisation cases are referred to as PH and PH-R.

6.4.1 Parallel-hybrid PH

Propulsion

The orientation of the parallel-hybrid propulsion system in XMDO is illustrated in Figure 6.12. In this architecture, the power gearboxes are of the turboprop category (Appendix F). Turboshaft₁ is chosen as the most critical engine that is turned into failure mode in all the missions of S1 excluding the nominal one. When Turboshaft₁ is failed, all the *priority subcomponents* (Section 4.5.2) of the same power unit are turned off (i.e. $r_{B1} = r_{NPL1} = 0$), and Propeller₁ is feathered.



Fig. 6.12 Orientation of the parallel-hybrid architecture PH

The local power rating r_{B2} of Battery₂ is chosen as r_{NR} , the local power rating controlled by the Newton-Raphson solver, and the minimum power rating $r_{NR_{min}}$ is taken equal to $r_{B_{min}} \in [-1,0]$, one of the design variables of this optimisation case. The reason for considering this design variable comes from the implementation of the braking drag (Equation 4.34). Since the braking drag is created when $r_{NR} < r_{NR_{min}}$, the use of $r_{B_{min}}$ relaxes the battery sizing constraint relatively to braking actions, which necessarily becomes active with $r_{NR_{min}} = -1$ if such drag needs to be generated. The local power rating r_{NPL2} is always 1, but r_{T2} is controlled according to Table 6.10 including new design variables.

	Mission/Segment	r_{T2}
	Taxi-out	$r_{T2_{TXO}} \in [0,1]$
	$Taxi-out_H$	$r_{T2_{TXOH}} \in [0.05, 1]$
	Take-off	$r_{T2_{TO}} \in [0.5, 1]$
It	Climb	$r_{T2_{CL}} \in [0.5, 1]$
ina	Cruise	$r_{T2_{CR}} \in [0.5, 1]$
om	Descent	$r_{T2_{DSC}} \in [0,1]$
Ζ	Landing	airborne: $r_{T2_{LG}} \in [0, 1]$ ground: 0.05
	$Taxi-in_H$	$r_{T2_{TXIH}} \in [0.05, 1]$
	Taxi-in	$r_{T2_{TXI}} \in [0,1]$
	OEI Take-off	$r_{T2_{TO}}$
	Rejected Take-off	$V \le V1: r_{T2_{TO}}$ V > V1: 0.05
	Landing at MLW	$r_{T2_{LG}}$
	OEI ceiling	1

Table 6.10 PH: control laws for Turboshaft₂ and bounds for new design variables

In nominal operation, the relations $r_{T_1} = r_{T_2}$ and $r_{B_1} = r_{B_2}$ are used. Lastly, the same parametric control of flap deflection angle as all the other optimisation cases is implemented.

Constraints

The PH aircraft is optimised under the same constraints as the reference aircraft (Table 5.3), plus a constraint relating to the battery capacity. Whereas the power sizing of the battery (i.e. max. C-rates) is enforced by $R_{SubComp}$, the energy capacity must also be checked. In the thesis, a minimum state of charge of 20 % is considered

in normal conditions, and the energy sizing for the nominal mission is ensured by the constraint $R_{EnergyB}$, defined in Equation 6.10.

$$R_{EnergyB}: \max SOC(t) - \min SOC(t) \ge 0.8 \tag{6.10}$$

For simplicity, this constraint is only applied to the nominal mission since the other ones rather relate to power sizing missions. For the PH case, the aircraft is allowed to arrive at the destination airport with empty batteries (i.e. $SOC \ge 0.2$); requiring them to be swapped or recharged before the next flight.

Optimisation results

The PH optimisation run involves 36 design variables (Aircraft: 8, Global: 8, OEI ceiling: 1, 400 nm mission: 18). The main differences with the REF case (26 design variables) deal with the propulsion system (i.e. Aircraft group) and the power ratings to control the turboshaft engines. The optimum solutions for the Aircraft group are shown in Table 6.11.

Name	Notation	Unit	Min.	Max.	EIS2025	EIS2030+
Wing chord	c_{r_W}	m	0.80	7.00	2.45	2.42
HTP chord	c_{r_H}	m	0.70	3.00	1.89	1.87
VTP chord	c_{r_V}	m	0.80	8.00	3.88	3.87
Propeller diam.	D_p	m	3.00	5.50	4.99	5.0
Gearbox diam.	D_{gbx}	m	0.55	0.90	0.88	0.89
Turboshaft diam.	$\tilde{D_{gt}}$	mm	650	850	767	766
Motor diam.	D_{mot}	m	0.2	1.5	0.46	0.43
Motor len.	L_{mot}	m	0.14	2.4	0.33	0.32
Converter len.	L_{conv}	m	0.01	1.5	0.06	0.07
Battery len.	L_{bat}	m	0.01	10	0.60	0.66

Table 6.11 Design variables: main solutions for PH propulsion system

From the weight breakdown (Table 6.12), one can notice the 15 % higher MZFW than the reference. With a total mass of approximately 1,500 kg, batteries bring most of the added mass from electrical components. Despite the implementation of batteries, the turboshaft engines are slightly oversized versus the reference. Because of the higher turboshaft output power, but also because of the additional gearbox input power from the electric motor, gearboxes are heavier. More surprisingly, propellers are also major contributors to the weight increase. Once again, the power sizing constraint on the propellers is never active, and the choice of the bigger diameter can

only be explained by the greater propeller efficiency. Nonetheless, with the geometrical parametric model of the aircraft, a larger propeller diameter induces a longer lever arm between the propeller axis and the aircraft CG. Hence, as can be seen from Table 6.13, the combination of the bigger engine and the longer lever arm involves a larger fin area than on the reference aircraft. These drag and weight penalties are most probably somehow outweighed by the propeller efficiency augmentation. In this table, one can also note the large increase of the other lifting surfaces versus the reference to meet the low-speed requirements with the higher mass. The aerodynamic efficiency of the PH aircraft is therefore poorer than for the reference, and shows a 5.4 % lower L/D ratio in cruise. In addition, with a cruise altitude of 30,000 ft and a cruise speed of about M0.45 (Figure 6.13), the output trajectories are actually fairly similar to those of the reference case. Still, as indicated in Table 6.12, the PH aircraft burns 9 % and 14 % less fuel than the reference for EIS2025 and EIS2030+, respectively. These figures can be better understood thanks to the analysis of the energy management strategies selected by the optimiser.

	EIS2025	$\Delta\%$ vs. REF	EIS2030+	$\Delta\%$ vs. REF
Total structure	5914	6.6~%	5902	6.4~%
Equipments	5500	1.8~%	5496	1.8~%
Propeller (x2)	599	70.7~%	599	70.8~%
Gearbox $(x2)$	362	37.5~%	350	32.8~%
Turboshaft $(x2)$	624	$4.1 \ \%$	622	3.8~%
Nacelle $(x2)$	364.3	33.9~%	485	33.0~%
Mot./gen. $(x2)$	120		107	
Converter $(x2)$	49		93	
Cable $(x2)$	82		107	
Battery $(x2)$	1409		1651	
Total propulsion	3814	141.7~%	3983	152.5~%
MZFW	22110	15.3~%	22035	14.9 %
Fuel burn F_{400}	735	-9.0 %	696	-13.9 %

Table 6.12 Weight summary for PH (kg)

The energy management strategy along the nominal mission for EIS2025 is provided in Figure 6.14. As can be seen from Figure 6.14a, all the design variables driving the local power rating r_{T2} are set to their lower bounds for all the taxi phases but also for the descent. During the descent, the Taxi-out and the Taxi-in segments (Figure 5.4),

		EIS2025	$\Delta\%$ vs.	REF	EIS2030+	$\Delta\%$ vs.	REF
	Wings	58.2	20.4	%	57.5	19 0	70
	Fin	15.3	37.6	%	15.2	36.9	%
ſ	Fail plane	10.6	30.4	%	10.4	28.0	%



Table 6.13 Area summary for PH (m^2)

Fig. 6.13 Output trajectories for PH

the gas turbines are therefore switched-off, and the propulsive and non-propulsive power needs are all supplied by the batteries. With $r_{T2_{TO}}$ and $r_{T2_{CL}}$ equal to one, turboshaft engines provide their maximum power during take-off and climb. The take-off rating for batteries being close to zero, the propulsion of the aircraft during take-off is almost solely ensured by the gas turbines. Along the first quarter of the climb phase batteries are lightly recharged and then provide power boost during the rest of the phase. The maximum hybrid ratio during climb, calculated from Figure 6.14b and defined as the ratio of battery power to the sum of turboshaft power and battery power, is 15.6 %. In cruise, the optimiser selected a turboshaft rating $r_{T2_{CR}}$ implicating neither discharge nor recharge of batteries. In descent, since the propeller input power is always positive, the optimum trajectory does not involve any energy recovering. During the air phase of the landing segment, the battery power rating is negative in accordance with the value of $r_{T2_{LG}}$ which actually has low influence on the total fuel consumption because of the short duration of this phase. Batteries are sized according to energy, and the C-rates are always lower than maximum values since the battery power rating never reaches + or -100 %. Note that the maximum output power is only 260 kW per battery. Also, the state of charge of batteries (Figure 6.14) at the end of the mission is equal to the minimum allowed value of 20 %.

Despite its higher mass and poorer aerodynamic efficiency, the PH configuration shows fuel savings versus the reference aircraft by drawing energy from batteries during taxi and descent phases rather than from fuel. The hybrid climb contributes in downsizing the turboshaft engines compared to an aircraft that would only implement electric taxi and electric descent. It should also be remembered that the objective function is the fuel consumption along the mission. The optimum solution, equipped with only 1,500 kg of batteries and showing a maximum hybrid ratio of 16 %, thus emphasises the difficulty of the implementation of hybrid-electric or all-electric architectures (Chapter 2). The latter is addressed in more detail in Section 6.6.

The optimum energy management strategy for EIS2030+ is fairly similar to the EIS2025 case as depicted in Figure 6.15. For a roughly equivalent total battery mass, the energy storage capacity is larger thanks to the new technology assumptions, and the maximum hybrid ratio in climb is now 23.8 %. With more energy drawn from batteries and higher component efficiencies, the PH configuration for EIS2030+ enables to save 5 % more fuel than with the shorter-term assumptions. However, recharging or swapping batteries at the airport can be a strong operating constraint especially in isolated facilities used by regional aircraft. For this reason, the PH-R optimisation case of the next section was also studied.



Fig. 6.14 PH - EIS2025: energy management strategy



Fig. 6.15 HP - EIS2030+: energy management strategy and powers

6.4.2 Parallel-hybrid with battery recharge PH-R

Based on the definition of the PH optimisation case, the constraint R_{RechB} is added to the optimisation problem in order to force batteries to be recharged along the nominal mission. This constraint, written in Equation 6.11, requires the state of charge at the end of the mission (time t_f) to be at least equal to the state of charge at the beginning (time 0).

$$R_{RechB}: \quad SOC(t_f) \ge SOC(0) \tag{6.11}$$

This optimisation run involves the same number of design variables as the PH case (i.e. 36). As can be seen from the weight breakdown of Table 6.14, the battery mass is about 300 kg, that is to say 5 times less than for the PH architecture. The weight penalty is also lower because of the smaller electrical components.

Regarding the areas of lifting surfaces, the fin is still much bigger than for the REF aircraft because a propeller diameter of about 5 m was selected by the optimiser (like for the PH case). The increase of the other surfaces are related to the added masses versus the reference. Despite the battery recharge constraint, the PH-R configuration is more fuel efficient than REF as shown in Table 6.14.

Figure 6.14 depicts the energy management strategy along the nominal mission for EIS2025. As can be seen from Figure 6.16a, the same management of the turboshaft power ratings as in the PH case is used. Hence, even for PH-R, only batteries supply propulsive and non-propulsive power needs during the Taxi-out and the Taxi-in

	EIS2025	$\Delta\%$ vs. REF	EIS2030+	$\Delta\%$ vs. REF
Total structure	5734	3.4~%	5734	3.4~%
Equipments	5435	0.6~%	5435	0.6~%
Propeller (x2)	564	60.8~%	563	60.4~%
Gearbox $(x2)$	342	29.6~%	342	29.8~%
Turboshaft $(x2)$	583	-2.7 %	584	-2.5 %
Nacelle $(x2)$	42	17.4~%	42	17.0~%
Mot./gen. $(x2)$	55		53	
Converter $(x2)$	22		26	
Cables $(x2)$	1		5	
Battery $(x2)$	272		350	
Total propulsion	2366	50.0~%	2346	48.7~%
MZFW	20185	5.3~%	20165	5.2 %
Fuel burn F_{400}	787	-2.6 %	769	-4.8 %

Table 6.14 Weight summary for PH-R (kg)

	EIS2025	$\Delta\%$ vs. REF	EIS2030+	$\Delta\%$ vs. REF
Wings	50.9	$5.5 \ \%$	50.9	5.4~%
Fin	14.6	31.2~%	14.7	31.8~%
Tail plane	9.8	20.7~%	9.8	20.9~%

Table 6.15 Area summary for PH-R (m^2)

segments. However, in descent, one can notice that the propeller input power becomes negative along the the second half of this phase, indicating that the aircraft recovers energy through propellers. Between 50 % and 75 % of the descent duration, the propellers are used to help batteries supplying non-propulsive loads. After 75 % of the descent time, the propellers supply all the non-propulsive loads and even lightly recharge batteries. Such strategy at the end of the descent actually makes sense in comparison to the reference aircraft, for which braking drag is generated to dissipate energy before the landing (Figure 5.7).

In order to satisfy R_{RechB} , the optimiser selected a turboshaft rating $r_{T2_{CR}}$ enabling the battery to be recharged during this phase.

In climb, the maximum hybrid ratio is only 2.8 %. Regarding the electrical power level, a maximum value of 170 kW per battery is used during the flight. For comparison, let us recall that the power consumption of each non-propulsive load component (i.e. NPL) is 70 kW. The PH-R architecture can therefore be characterised as micro-hybrid



Fig. 6.16 PH-R - EIS2025: energy management strategy

propulsion system. But by taking the advantage of the high efficiency of the electrical chain in taxi and descent, and by recharging the battery in cruise where the gas turbine efficiency is good, the PH-R aircraft eventually saves 2.6 % of fuel versus the reference aircraft. The fuel saving opportunity is of course much less than for the PH configuration, but the PH-R aircraft only needs to be refuelled for the next trip.

The optimisation results for EIS2030+ reveal a similar energy management strategy, as depicted in Figure 6.17. In this case, batteries are never recharged by the propellers during the descent. However, propellers still recover a small amount of energy from the airflow to supply part of the non-propulsive loads during the second half of the



Fig. 6.17 PH-R - EIS2030+: energy management strategy and powers

descent. The larger fuel saving (Table 6.14) compared with EIS2025 comes from the better efficiencies of electrical components and from the higher maximum hybrid ratio in climb going up to 5.9 %.

6.5 Comparison of fuel and energy efficiencies

The results of the different optimisation cases presented before are compared in terms of MTOW and fuel consumption for the 400 nm mission versus the REF aircraft in the left-hand side graph of Figure 6.18.

The two architectures TE4 and PT8 that have been considered for the study of the potential aerodynamic improvements enabled by electrified propulsion systems are less fuel efficient than the optimised reference aircraft, including for the 2030+ technology assumptions. Furthermore, let us remind that optimistic models are used for these optimisation cases: the single electrical machine model for TE4 and PT8 which is not compatible for all the machines of these architectures (Chapter 2), but also the high-lift propeller model for PT8. While the TE4 aircraft fully fits the bounds of the experimental data that are used by ESDU (Appendix C) for testing the accuracy of the blown wing model, these results suggest that the blown wing concept may not be adapted to this aircraft application defined by the top level aircraft requirements (TLARs). A more aggressive target on the landing speed and/or the take-off field length would probably result in more achievable fuel burn reductions with the blown wing concept.

Inversely, the architectures PH and PH-R that have been used for the analysis of new energy management strategies all show fuel savings. As expected, their fuel efficiency is increased with electrical technology improvements. Contrary to the turboelectric architecture TE4, the parallel-hybrid architectures PH and PH-R involve lower electrical powers requiring smaller technology advances beyond the state of the art to be implemented. However, the main savings comes from the possibility to stop the engines during idle phases, requiring such technology to ensure fast relight in case of emergency and to be developed for aeroplanes (Section 1.3.4). Besides, the depicted benefits will somehow be reduced by thermal constraints that have been neglected in this thesis. Taking into account the energy reserves that were neglected in this work should not change the relative fuel savings for these architectures, since fuel is likely to be used such as for the reference.

Because a mono-objective function was chosen (fuel burn), the PH shows a higher MTOW than TE4 even under EIS2030+ assumptions. Seeing that this architecture actually replaces some of the fuel energy by energy stored in batteries, it is also interesting to compare it in term of energy consumption. The energy consumption shown on the left chart of Figure 6.18 is calculated by considering the energy drawn from batteries (i.e. the energy lost during the recharge at the airport is not included) and assumes that 1 kg of fuel burn is equivalent to 43 MJ of energy consumption (typical fuel lower heating value). Based on this definition, the PH configuration is still more energy efficient than the reference because of the better efficiency of the power chain when batteries are used. With this metric giving the same cost to 1 MJ from fuel and from batteries, the energy benefit is now less than 10 % for EIS2030+. In terms of operating costs, the variation of fuel price versus electricity in the next decades can also change the picture.

Finally, if the comparison of the different architectures in Figure 6.18 is based on the maximum range mission, it should be done for the different trips of the flight distribution; and in particular for the most occurring mission (200 nm). The PH configuration of Section 6.4.1, though optimised according to a single nominal mission because of computing power limitations and time constraint, would already show higher relative benefits on a smaller range thanks to its secondary energy source sized for the maximum range. But for this architecture using two types of energy sources, including more nominal missions in the design process would fully make sense in order to optimise the battery capacity, its energy management strategy, and the structural weights with regard to the flight distribution. Allowing the battery capacity to be modified for the different flights even provides an additional degree of freedom. The



Fig. 6.18 Summary of results versus reference aircraft

resulting parallel-hybrid aircraft would most probably be more fuel efficient along its operating life than the PH aircraft optimised in this chapter.

6.6 Technology target setting: all-electric AE

Even though the objective function of the previous optimisation cases was clearly in favour of a full electric flight, the optimum parallel-hybrid aircraft still use significant fuel quantities. The identification of minimum battery technology levels for the all-electric aircraft AE is addressed in this section.

6.6.1 Airframe and component models

The AE aircraft is modelled thanks to the components presented in Chapter 5 and Chapter 6. Like in the above optimisation cases, the wing span is fixed to 27 m, and the same design variables relating to the airframe are considered. The single difference at component level deals with the inputs and outputs of the battery model.

Whereas the battery specific energy \overline{E} was previously fixed by the EIS, it is now considered as design variable. Also, the battery length L_b that was used to control the battery size is removed from the set of design variables since the battery mass is directly calculated from the MTOW that is fixed, as explained in the next section. The maximum power P_{max} is still calculated from Equation 6.7 (i.e. same maximum C-rates) and the battery capacity E_{max} is simply evaluated from the product of the battery mass and \overline{E} .

All the other electrical components are still defined by the previous models based on the EIS assumptions.

6.6.2 Missions, objective function and constraints

The following optimisation runs are performed according to the same types of missions as the ones constituting S1 (Chapter 5) but the range of the nominal mission is varied between the different optimisation cases: 100 nm, 200 nm, 400 nm and 800 nm. The minimum battery technology level is searched by choosing the battery specific energy \overline{E} as objective function. But because it is always possible to design a heavy aircraft, on top of the trade on maximum range, three different levels of MTOW are considered: 25 t, 30 t and 35 t.

For a given MTOW (which is equal to the MZFW for an all-electric aircraft), the total battery mass $M_{b_{tot}}$ is given by Equation 6.12, where $(OEW - M_{b_{tot}})$ is the sum of all the masses of the aircraft components excluding batteries (e.g. wings, fin, motor, gearbox, etc...). Note that this sum depends on the design variables driving the size of these components.

$$M_{b_{tot}} = MTOW - payload - (OEW - M_{b_{tot}}) \tag{6.12}$$

Finally, the different AE aircraft are optimised according to the same set of constraints as the parallel-hybrid PH (Section 6.4.1).

6.6.3 Propulsion

The orientation of the all-electric propulsion system in XMDO is represented in Figure 6.19. In this architecture, gear reducers are in-line gearboxes. The propulsion model does not include any non-propulsive power off-take (NPL component) since the following optimisations were performed before the implementation of the new generic model in XMDO (Section 4.5.2). For the same reason the minimum power rating for batteries is zero, and, therefore, does not allow energy recovering. The local power rating r_{B2} of Battery₂ is controlled by the Newton-Raphson for solving the equations of motion. The relation $r_{B1} = r_{B2}$ is used in nominal operation (also during taxi), and r_{B1} is set to zero in propulsion failure scenarios. When r_{B1} is zero, Propeller₁ is feathered. Otherwise, the same assumptions on control laws (local power ratings and flap setting) along the different missions of S1 as for the reference aircraft (Section 5.3.1) are used.



Fig. 6.19 Orientation of the all-electric architecture AE

6.6.4 Optimisation results

With the combined trade-off studies on design range ($\times 4$), MTOW ($\times 3$) and EIS ($\times 2$), 24 optimisation runs were launched. A single run involves 27 design variables: Aircraft: 10, Global: 6, OEI ceiling: 1, nominal mission: 10; (variable groups in reference to Table 5.4).

The design variables of the Aircraft group and the take-off rating (Global group) for the four ranges at 25 t MTOW under EIS2030+ assumptions are provided in Table 6.16. It can be seen that all the design variables relating to a component dimension (except for the propeller diameter) are identical from one design range to the other. This can be explained by the fact that the MTOW being constant (and so are the MZFW and MLW), these components are sized under the same conditions by the same critical scenarios. Indeed, the wing area and the horizontal stabiliser area are both sized by the landing at MLW, and the fin area is constrained by the take-off with one motor inoperative. Finally, unlike turboshaft engines, the maximum available power of the electric chain does not fall with altitude. Hence, all the components of the propulsion system (except batteries) are now sized by the power required to take-off at MTOW. Like for the other optimisation cases of the thesis, the maximum power constraint for the propeller is never active. Gear ratios are therefore different between the optimisation runs, and they range from 1.4 to 1.8 for EIS2025 and from 2.3 to 2.9 for EIS2030+.

With identical geometrical parameters for almost all the aircraft components between the mission ranges under a given EIS, the total battery weight calculated according to Equation 6.12 is only a function of MTOW at first order, as shown in Figure 6.20a. In addition, areas of lifting surfaces are driven by the critical scenarios mentioned previously, and all the airframe components (fuselage, equipments) are only functions of MTOW. The airframe mass therefore only depends on the MTOW, and the mass left to the propulsion system including batteries is independent of the EIS assumptions as depicted in Figure 6.20a. Since the required propulsive power are only functions of MTOW at first order, the lighter components brought by the EIS2030+ enable the aircraft to be equipped with heavier batteries.

Name	Notation	Unit	Min.	Max.	100 nm	$200~\mathrm{nm}$	$400~\mathrm{nm}$	800 nm
Wing chord	c_{r_W}	m	0.80	7.00	2.95	2.95	2.95	2.95
HTP chord	c_{r_H}	m	0.70	3.00	2.12	2.12	2.12	2.12
VTP chord	c_{r_V}	m	0.80	8.00	4.00	4.03	4.00	4.01
Propeller diam.	D_p	m	3.00	5.50	4.66	4.91	4.74	4.95
Gearbox diam.	D_{qbx}	m	0.55	0.90	0.48	0.48	0.48	0.49
Motor diam.	D_{mot}	m	0.2	1.5	0.83	0.85	0.82	0.84
Motor len.	L_{mot}	m	0.14	2.4	0.79	0.74	0.79	0.75
Converter len.	L_{conv}	m	0.01	1.5	0.33	0.31	0.32	0.32
Cable diam.	D_c	mm	1	100	20.26	20.25	19.9	20.21
Batt. Spec. Energy	\overline{E}	Wh/kg	100	2500	307	583	1062	2139
Take-off rating	r_{TO}	-	0.10	1.00	0.77	0.39	0.23	0.11

Table 6.16 Design variables: main solutions for 25 t MTOW and EIS2030+

The battery specific energies found by the optimisation algorithms are reported in Figure 6.21. Because the total battery mass turned out to be constant at a given MTOW, the aircraft energy consumption can be directly derived from each iso-MTOW curve. When increasing the MTOW, heavier batteries can be carried by the aircaft to perfom the mission requiring lower battery specific energies. The smaller specific energy reduction between the largest MTOWs than between 25 t and 30 t comes from the aerodynamic efficiency getting strongly affected by the constant span assumption. Indeed, as depicted in Figure 6.24, the 35 t MTOW involves large lifting areas and,


Fig. 6.20 Total battery mass and propulsion system mass (incl. batteries)

above all, a small wing aspect ratio compared to the other MTOWs (Figures 6.22 and 6.22).

Based on the 30 t MTOW, a 100 nm full-electric flight appears to be achievable with the battery technology prediction of EIS2025. On the longer-term, the battery technology of EIS2030+ would enable the all-electric aircraft to fly over 200 nm.

Yet, these results must be carefully considered. First, contrary to the other hybrid aircraft cases of the thesis, the energy consumption of non-propulsive systems is neglected. Also, the consideration of thermal constraints can clearly penalise the all-electric architecture involving high electric power levels. If battery mass is often considered as a show stopper for full-electric flights, battery volume is at least as restrictive as it. Finally, while fuel is an excellent energy storage for reserves on hybrid aircraft equipped with gas turbines, the energy capacity of batteries of all-electric aircraft should include them. With current regulatory rules, the range of the diversion mission that a regional aircraft must be able to cover is in the order of 100 nm, reducing by half the achievable range for EIS2030+. All of these emphasises the challenging implementation of all-electric propulsion systems for 70-seat propeller aircraft, even in the long-term.



Fig. 6.21 Best objective function value for the 24 optimisation cases



Fig. 6.22 Geometry of the all-electric aircraft: 400 nm design range, 25 t MTOW and EIS 2030+



Fig. 6.23 Geometry of the all-electric aircraft: 400 nm design range, 30 t MTOW and EIS 2030+



Fig. 6.24 Geometry of the all-electric aircraft: 400 nm design range, 35 t MTOW and EIS 2030+

Conclusion

The design space exploration for the 70-seat hybrid regional aircraft of this thesis started with the identification of energy saving opportunities through performance analyses of a conventional aircraft of the same category. Estimated potentials of energy saving are obtained from first order calculations by neglecting the mass and drag penalties added by the electrical components.

The analysis of a typical power profile showed that potential energy saving brought by transient energy storage is much smaller than for ground-based transportation. In addition, while kinetic energy recovering is used on hybrid cars, it was highlighted that recovering braking energy at landing has low interest because of the small fraction of the total energy consumption it represents. If the gravitational potential energy of the aircraft at the end of the cruise is much larger than the braking energy at landing, the simplified study showed that recovering it does not bring benefit on the overall energy balance in normal operation. Yet, it was decided to study the *Energy recovering* concept more in depth in the thesis.

Energy saving opportunities actually come from new propulsion power managements enabled by secondary energy sources (e.g. batteries), and from aerodynamic improvements enabled by distributed propulsion. New propulsion power managements can improve the propulsion system efficiency during idle phases such as taxi and descent (*Start & Stop*), and relax design requirements for the gas turbines (*Engine* downsizing). Potential aerodynamic improvements come from the reduction of lifting areas: the wings for the Blown wing concept, and the vertical stabiliser for the Differential thrust. A simple analysis in cruise demonstrated the poor interest of boundary layer ingestion for such aircraft application. Regarding the concepts that were selected to be studied in more detail in the thesis, at least as many energy savings were expected from aerodynamic improvements (Blown wing, Differential thrust) as from new power managements (Energy recovering, Start & Stop, Engine downsizing). In order to correctly evaluate the previous energy saving opportunities, six aircraft configurations were defined. These aircraft were all based on a similar airframe configuration and featured different propulsion systems. One of the six aircraft is a conventional twin turboprop, and served as a reference. Since the potential energy savings coming from new power managements are independent at first order to those coming from the aerodynamic improvements, it was decided to study them separately. Hence, the turboelectric with distributed propulsion and the partial-turboelectric with high-lift propellers were chosen for the study of aerodynamic improvements (*Blown wing, Differential thrust*), and the parallel-hybrid for new power managements (*Energy recovering, Start & Stop, Engine downsizing*). The last aircraft is an all-electric configuration and was used in the last part of the thesis for the evaluation of the required specific energy of batteries for full-electric flight.

Then, the constrained multidisciplinary optimisation problem that must be solved when designing the selected aircraft configurations was stated. From the classical design process for conventional aircraft, the required modifications to perform the design of hybrid-electric aircraft involving multiple types of energy source were identified. In particular, a new mathematical formulation of the so-called Mass-Performance Loop based on multi-mission evaluations was proposed.

The implementation of the design process for the different aircraft was first carried out in a commercial MDO platform. In addition, a high-speed performance evaluation tool for conventional aircraft was made compatible with the use of batteries and multiple power flow combinations, and was integrated into the MDO platform. The lack of flexibility, but also the high computation time required by the modified aircraft performance tool, pushed the author to consider an other MDO tool set. The XMDO tool was finally selected as it already incorporated most of the required modifications, including a simplified multi-mission Mass-Performance Loop. This tool is an in-house Airbus MDO platform dedicated to the optimisation of new aircraft concepts with low level of knowledge (i.e. without a priori) of sizing missions and occurrences of sizing constraints. The implementation of the multi-mission Mass-Performance Loop exposed in the thesis was straight forward thanks to the IDF-based architecture of this tool. Moreover, XMDO enabled to optimise at the same time the airframe, the propulsion system, the control laws (e.g. energy management) and the trajectories, which was one of the hardest achievable objectives of the thesis. Although XMDO provided strong bases, numerous developments and tests were done to carry out the aircraft studies. For examples, several estimation models were built for the aircraft components: turboshaft, propeller, gearbox, electrical motor, blown wing, etc. The simulation models for take-off and landing were also improved (failure scenarios, one-g stall speed definition, balanced field length calculation, etc), different trajectory formulations were tested and a new parametric control law for flap configuration was defined. In order to include the possibility to optimise aircraft configurations using more than one type of energy source, such as the parallel-hybrid, a new generic formulation for solving the propulsion system equilibrium was implemented. Lastly, several optimisation algorithms and penalty functions were tested.

The optimisation runs for the reference aircraft with XMDO, according to fuel burn minimisation, enabled to validate the new design method through the analyses of the optimum solutions and their sizing scenarios. In addition, they confirmed the rationale of the classical OAD process, involving a single-mission Mass-Performance Loop, when the aircraft uses a unique type of energy. The reference aircraft shows a large fuel burn reduction of 22 % on a 200 nm mission versus ATR72, coming from relaxed TLARs (maximum payload, maximum range), neglected fuel reserves, higher cruise altitudes, improved technology levels (turboshaft engines, aerodynamics), and taking benefit of blown wing effect on one-g stall speed calculation.

Afterwards, all the hybrid-electric aircraft were optimised under the same aircraft design requirements and objective function as the reference. Besides, the same parametric models as for the reference were used, and two levels of technology were defined for the electrical components regarding the service entry date of the aircraft (2025 or 2030+). Despite optimistic models, the two architectures (turboelectric and partial turboelectric) that were selected for the study of aerodynamic improvements are less fuel efficient than the reference, including for the longer-term technology level. Oppositely to the expectations arisen from the simple analyses of the first part of the thesis, both the blown wing and the differential thrust (i.e. fin area reduction) concepts do not bring fuel savings versus the reference. While these concepts effectively create aerodynamic improvements, the aerodynamic gains are always overweighted by the mass penalties from the additional electrical components. On top of these penalties, the turboelectric architecture suffers from efficiency penalties of power transmissions throughout the flight. It should also be noted that the reference aircraft already offers excellent aerodynamic performances, making their improvement even more difficult.

The parallel-hybrid configuration, defined for the investigation of new power managements, always shows fuel benefits: -5 % with in-flight battery recharge and -14 % with battery swapping or battery recharge at the airport, under 2030+ assumptions. In addition, the optimum energy management strategies confirmed the interest of the electric taxi, the electric descent and the electric power boost in climb (i.e. engine downsizing)—all three identified in the first chapter—for this aircraft application. With in-flight battery recharge, a small amount of energy from the airflow is even harvested during the descent thanks to the propellers; with recovered powers in the order of non-propulsive power needs. The fraction of the fuel consumption on idle phases being even more important relatively to the trip fuel as the range is reduced, more relative fuel savings are expected along shorter ranges. Therefore, on a 200 nm mission, by combining all the effects of aircraft redesign and propulsion hybridisation, fuel saving may be as high as -40% (-18 %, respectively) with battery swapping or battery recharge at the airport under 2030 + assumptions versus the ATR72 (the reference aircraft REF, respectively). This figure is even more attractive that the electrical powers involved in the parallel-hybrid configurations of the thesis are in the order of hundreds of kW, favouring their technical implementation. Also, if the parallel-hybrid aircraft are designed to fly 400 nm efficiently, batteries could be replaced by fuel to cover longer ranges for temporary needs.

The optimisations of the all-electric aircraft, performed under optimistic assumptions (no energy reserves, no thermal constraints, no non-propulsive load), reveal that the required specific energy for batteries far exceed the 2025 (2030+, respectively) technology prediction for a design range larger than 100 nm (200 nm, respectively). These results emphasises the challenging implementation of all-electric propulsion systems for a 70-seat propeller aircraft, even in the long-term. In addition, they show the importance of attacking aircraft hybridisation challenges on intercity aircraft and/or smaller ones such as commuters.

Of course, all the results of the thesis depend on the models, the assumptions and the optimisation methods that were considered. Yet, being not based on a priori sizing scenarios, XMDO enabled to enhance the design space exploration for the different hybrid-electric aircraft configurations. Prospects for future work are many and varied, and commented next. Because the estimation of energy reserves requires additional segments or missions to be evaluated in the design process, they were neglected in order to limit the computation time. Except for the all-electric, fuel is likely to be used as energy storage for reserves. Although their influence on the relative fuel savings is expected to be small, energy reserves should be evaluated for the correct estimation of aircraft performance.

While the multi-mission Mass-Performance Loop was presented as an important modification of the aircraft design process when using different types of energy source, the parallel-hybrid aircraft was optimised according to a single nominal mission because of computing power limitations. Hence, using large computing resources and parallel programming appears necessary to take the benefit of the multi-mission sizing for the parallel-hybrid aircraft, which would most probably be more fuel efficient along its operating life than the one optimised in this thesis. In addition, more degrees of freedom could then be considered such as the battery capacity that could be modified for the different flights for example. Including more nominal missions in the design process for the other selected aircraft would also enable to compare them for each mission of the flight distribution, though it would change neither the size of the aircraft nor that of its propulsion system.

Trajectories and control laws were kept as simple as possible in order to limit the number of design variables, and so, here again, for computing power reasons. The design space exploration can be further enlarged by increasing the number of control points along each segment of the mission. However, let us note that, except for take-off and landing, no particular operating constraint (e.g. flight levels, margins at service ceiling) was taken into account. Therefore, considering current operating rules in the optimisation problem would also limit the possible trajectories.

In this thesis, thermal management was neglected in the interest of reducing the number of components of the propulsion chain modelling. While thermal constraints can become real design drivers with electrified propulsion systems, more refined studies would at least require to account for the effects of cooling drag forces and cooling system weights at aircraft level.

For more system focused studies, the propulsion system modelling that currently exchanges powers between the components could be modified to exchange data sets such as (torque,speed) or (voltage,current). This would make sense if the complexity of the component models is upgraded, and in particular that of the electrical component models.

As commented before, all the results depends on the models and assumptions that were used. If optimising the aircraft with two technology levels provides a first order sensitivity analysis regarding the electrical component assumptions, the optimisations should actually be performed with uncertainty propagations to give additional meaning to the results.

While the parallel-hybrid architecture involves lower electrical powers than a turboelectric or a series-hybrid, thus requiring smaller technology advances beyond the state of the art to be put into service, the main savings comes from the possibility to stop the engines during idle phases. If a start-and-stop system can be easily implemented on internal combustion engines that are mainly used by ground-based vehicles, its feasibility for gas turbines equipping aeroplanes is still to be proven. Indeed, such system must meet strong requirements with regard to reliability and to restarting speed in case of emergency. Lessons can most probably be learnt from the hybridisation of multi-engine helicopters for which engine manufacturers are developing standby operating modes and electric-assisted restarting, though the engines are never fully switched-off.

About the blown wing concept, the results of this thesis suggest that it may not be adapted to the aircraft application defined by the top level aircraft requirements (TLARs) considered in the thesis. By considering the problem the other way around, it would be interesting to use MDO methods to define the right application for this concept. This could be done by setting some of the TLARs as design variables, those related to the take-off field length and the landing speed for examples, and add a constraint expressing that the fuel burn of the aircraft fitting a blown wing must be smaller than that of the reference. Of course, this require the two aircraft to be optimised under the same requirements, and, therefore, at the same time. Such approach could actually be used to prove the viability of any of the concepts identified in the first chapter of this manuscript. To go even further, it should be interesting to study the sensibility of these concepts to the aircraft size (i.e. seat-capacity).

Finally, it should be recalled that all the concepts for fuel saving listed in the first part of the thesis were not studied. Wing tip propellers, which may enable to improve both the aerodynamic efficiency and the propulsive efficiency of the aircraft, were not considered because of the lack of analytical models. The hydrogen-powered aircraft was not analysed due to time constraint. These are two technology bricks that are worth being studied more in depth in the case of a regional aircraft.

References

- Aainsqatsi, K. (2018). Schematic diagram illustrating the operation of a 2-spool, high-bypass turbofan engine, with lp spool in green and hp spool in purple. [online; accessed 2018-05-26] https://commons.wikimedia.org/wiki/File:Turbofan_operation. svg?uselang=en.
- [2] Abeyounis, W. K., Patterson, J. C., Stough, H. P., Wunschel, A. J., and Curran, P. D. (1990). Wingtip vortex turbine investigation for vortex energy recovery. Technical report, SAE Technical Paper.
- [3] Airbus (2002). *Getting to Grips with Aircraft Performance*. Airbus Customer Services.
- [4] aircraft, A. (2018). Atr72-600. [online; accessed 2018-06-26] http://www.atraircraft. com/datas/download_center/27/fiche72_27.pdf.
- [5] Akli, C. R. (2008). Conception systémique d'une locomotive hybride autonome : application à la locomotive hybride de démonstration et d'investigations en énergétique lhydie développée par la sncf.
- [6] Akli, C. R., Roboam, X., Sareni, B., and Jeunesse, A. (2007). Energy management and sizing of a hybrid locomotive. In *Power Electronics and Applications*, 2007 *European Conference on*, pages 1–10. IEEE.
- [7] Anderson Jr, J. D. (2010). *Fundamentals of aerodynamics*. Tata McGraw-Hill Education.
- [8] André, M. (2004). The artemis european driving cycles for measuring car pollutant emissions. *Science of the total Environment*, 334:73–84.
- [9] Antcliff, K. R. and Capristan, F. M. (2017). Conceptual design of the parallel electric-gas architecture with synergistic utilization scheme (pegasus) concept. In 18th AIAA/ISSMO Multidisciplinary Analysis and Optimization Conference, page 4001.
- [10] Antcliff, K. R., Guynn, M. D., Marien, T., Wells, D. P., Schneider, S. J., and Tong, M. J. (2016). Mission analysis and aircraft sizing of a hybrid-electric regional aircraft. In 54th AIAA Aerospace Sciences Meeting, page 1028.
- [11] Authorities, J. A. (1996). Jar-ops. Flight operations.
- [12] Badufle, C. (2007). Conceptual aircraft design: towards multiobjective, robust and uncertain optimisation. PhD thesis, Université Paul Sabatier-Toulouse III.

- [13] Balabanov, V., Charpentier, C., Ghosh, D., Quinn, G., Vanderplaats, G., and Venter, G. (2002). Visualdoc: a software system for general purpose integration and design optimization. In 9th AIAA/ISSMO symposium on multidisciplinary analysis and optimization, page 5513.
- [14] Birman, J. (2013). Uncertainty quantification and propagation in Conceptual Aircraft Design: from deterministic optimization to chance constrained optimization. PhD thesis, Ph. D. thesis. University of Toulouse III-Paul Sabatier.
- [15] Black, D., Menthe, R., and Wainauski, H. (1978). Aerodynamic Design and Performance Testing of an Advanced 30 Swept, Eight Bladed Propeller at Mach Numbers from 0.. 2 to 0.85. National Aeronautics and Space Administration.
- [16] Boeing (2000). Aero magazine no. 12. [online; accessed 2018-04-17] http://www.boeing.com/commercial/aeromagazine/aero_12/aoa.pdf.
- [17] Boeing (2001). Wingtip devices. [online; accessed 2018-01-23] http://www.boeing. com/commercial/aeromagazine/aero_17/wingtip_devices.html.
- [18] Boldea, I. (2015). Synchronous generators. CRC Press.
- [19] Borer, N. K., Patterson, M. D., Viken, J. K., Moore, M. D., Clarke, S., Redifer, M., Christie, R., Stoll, A., Dubois, A., Bevirt, J., et al. (2016). Design and performance of the nasa sceptor distributed electric propulsion flight demonstrator. *AIAA Aviation*, pages 13–17.
- [20] Bradley, M. K., Droney, C. K., and Allen, T. J. (2015). Subsonic ultra green aircraft research. phase ii-volume i; truss braced wing design exploration.
- [21] Brown, G. (2011). Weights and efficiencies of electric components of a turboelectric aircraft propulsion system. In 49th AIAA aerospace sciences meeting including the new horizons forum and aerospace exposition, page 225.
- [22] Brown, G. V., Kascak, A. F., Ebihara, B., Johnson, D., Choi, B., Siebert, M., and Buccieri, C. (2005). Nasa glenn research center program in high power density motors for aeropropulsion. Technical report, US Army Research Laboratory Aberdeen Proving Ground United States.
- [23] Budinger, M., Liscouët, J., Hospital, F., and Maré, J. (2012). Estimation models for the preliminary design of electromechanical actuators. *Proceedings of the Institution* of Mechanical Engineers, Part G: Journal of Aerospace Engineering, 226(3):243–259.
- [24] Carlton, J. (2012). Marine propellers and propulsion. Butterworth-Heinemann.
- [25] Corporation, J. S. (2018). Electrical steel sheets. [online; accessed 2018-05-10] http://www.jfe-steel.co.jp/en/products/electrical/catalog/fle-001.pdf.
- [26] Czinczenheim, J. (1977). A deflected slipstream aircraft of the sixties/breguet 941/and its contribution to the solution of some stol problems. In V/STOL Conference, page 565.
- [27] Dassault, S. (2012). Isight simulia. Dassault Systèmes.

- [28] Defoort, S., Serre, L., Grenon, R., Varnier, J., Carrier, G., Scherrer, D., and Narmada (2012). Zehst: environmental challenges for hypersonic passenger transport. In 18th AIAA/3AF International Space Planes and Hypersonic Systems and Technologies Conference, page 5873.
- [29] Delahaye, D., Puechmorel, S., Tsiotras, P., and Féron, E. (2014). Mathematical models for aircraft trajectory design: A survey. In *Air Traffic Management and Systems*, pages 205–247. Springer.
- [30] Diederich, F. W. (1952). A simple approximate method for calculating spanwise lift distributions and aerodynamic influence coefficients.
- [31] DOE, U. (2017). Targets for onboard hydrogen storage systems for light-duty vehicles. [online; accessed 2018-01-26] https://energy.gov/sites/prod/files/2017/05/f34/fcto_myrdd_table_onboard_h2_storage_systems_doe_targets_ldv_1.pdf.
- [32] Dubois, A., van der Geest, M., Bevirt, J., Christie, R., Borer, N. K., and Clarke, S. C. (2016). Design of an electric propulsion system for sceptor's outboard nacelle. In 16th AIAA Aviation Technology, Integration, and Operations Conference, page 3925.
- [33] EASA (2018). Type certificates. [online; accessed 2018-06-05] https://www.easa. europa.eu/document-library/type-certificates.
- [34] EASA, C. (2009). Certification specifications for large aeroplanes.
- [35] Enterprises, S. et al. (2012). Scilab: Free and open source software for numerical computation. *Scilab Enterprises, Orsay, France*, page 3.
- [36] ESDU, I. (2015). Esdu 88031: Lift and longitudinal forces on propeller/nacelle/wing/flap systems.
- [37] Etel (2018). Etel torque-motors. [online; accessed 2018-05-28] https://www.etel.ch/en/torque-motors/tmb/specifications/.
- [38] Falck, R. D., Chin, J. C., Schnulo, S. L., Burt, J. M., and Gray, J. S. (2017). Trajectory optimization of electric aircraft subject to subsystem thermal constraints. In 18th AIAA/ISSMO Multidisciplinary Analysis and Optimization Conference, Denver, CO.
- [39] Falck, R. D. and Gray, J. S. (2016). Parallel aircraft trajectory optimization with analytic derivatives. In 17th AIAA/ISSMO Multidisciplinary Analysis and Optimization Conference, part of AIAA Aviation 2016 (Washington, DC).
- [40] Felder, J. L. (2015). Nasa electric propulsion system studies.
- [41] Felder, J. L., Brown, G. V., DaeKim, H., and Chu, J. (2011). Turboelectric distributed propulsion in a hybrid wing body aircraft.
- [42] Fitzgerald, A. E., Kingsley, C., Umans, S. D., and James, B. (2003). *Electric machinery*, volume 5. McGraw-Hill New York.

- [43] Fusalba, F. (2015). Key drivers for aerospace batteries. today and future aircraft electrically powered. [online; accessed 2018-01-25] https://sunjet-project.eu/sites/ default/files/CEA-Fusalba.pdf.
- [44] Gallard, F., Meaux, M., Montagnac, M., and Mohammadi, B. (2013). Aerodynamic aircraft design for mission performance by multipoint optimization. In 21st AIAA Computational Fluid Dynamics Conference, page 2582.
- [45] Gallard, F., Mohammadi, B., Montagnac, M., and Meaux, M. (2015). An adaptive multipoint formulation for robust parametric optimization. *Journal of Optimization Theory and Applications*, 167(2):693–715.
- [46] Garner, H. (1968). Some remarks on vortex drag and its spanwise distribution in incompressible flow. *The Aeronautical Journal*, 72(691):623–625.
- [47] Gerada, D., Mebarki, A., Brown, N. L., Gerada, C., Cavagnino, A., and Boglietti, A. (2014). High-speed electrical machines: Technologies, trends, and developments. *IEEE Transactions on Industrial Electronics*, 61(6):2946–2959.
- [48] Gerhardt, H. (1999). Wingtip vortex device for induced drag reduction and vortex cancellation. US Patent 5,934,612.
- [49] Globalsecurity (2018). Al0993 figure 10.1: Cutaway view of the t700-ge-700. [online; accessed 2018-05-26] https://www.globalsecurity.org/military/library/policy/army/ accp/al0993/fig10_1.htm.
- [50] Gratzer, L. B. (1992). Spiroid-tipped wing. US Patent 5,102,068.
- [51] Gray, J. S., Moore, K. T., and Naylor, B. A. (2010). Openmdao: An open-source framework for multidisciplinary analysis and optimization. In 13th AIAA/ISSMO Multidisciplinary Analysis and Optimization Conference, Fort Worth, TX, AIAA, AIAA-2010-9101, Fort Worth, Texas. AIAA.
- [52] Hansen, N. and Kern, S. (2004). Evaluating the cma evolution strategy on multimodal test functions. In *International Conference on Parallel Problem Solving* from Nature, pages 282–291. Springer.
- [53] Hendricks, E. and Tong, M. (2012). Performance and weight estimates for an advanced open rotor engine. In 48th AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit, page 3911.
- [54] Hermetz, J., Ridel, M., and Doll, C. (2016). Distributed electric propulsion for small business aircraft a concept-plane for key-technologies investigations. In *ICAS* 2016.
- [55] Hosking, E., Kenny, D., McCormick, R., Moustapha, S., Sampath, P., and Smailys, A. (1998). The pwloo engine: 20 years of gas turbine technology evolution. In Proceedings of the RTO AVT Symposium on Design Principles and Methods for Aircraft Gas Turbine Engines, Toulouse, France, pages 11–15. Citeseer.

- [56] Isikveren, A. T., Seitz, A., Vratny, P. C., Pornet, C., Plötner, K. O., and Hornung, M. (2012). Conceptual studies of universally-electric systems architectures suitable for transport aircraft. In *Deutscher Luft-und Raumfahrt Kongress*. DLRK Berlin.
- [57] ISO 13691:2003-02 (2003). Petroleum and natural gas industries High-speed special-purpose gear units. Standard, International Organization for Standardization.
- [58] Jayaram, S., Myklebust, A., and Gelhausen, P. (1992). Acsynt-a standards-based system for parametric, computer aided conceptual design of aircraft. In *Aerospace Design Conference*, page 1268.
- [59] Johnson, S. G. (2014). The nlopt nonlinear-optimization package.
- [60] Jones, E., Oliphant, T., and Peterson, P. (2014). {SciPy}: open source scientific tools for {Python}.
- [61] Juvé, L., Fosse, J., Joubert, E., and Fouquet, N. (2016). Airbus group electrical aircraft program, the e-fan project. In 52nd AIAA/SAE/ASEE Joint Propulsion Conference, page 4613.
- [62] Kroo, I. and Shevell, R. (2001). Aircraft design: Synthesis and analysis. Desktop Aeronautics Inc., Textbook Version 0.99.
- [63] Kroo, I. and Takai, M. (1988). A quasi-procedural, knowledge based system for aircraft synthesis. In AIAA Aircraft Design Conf., AIAA Paper, number 88-6502.
- [64] Kuhn, R. E. (1959). Semiempirical procedure for estimating lift and drag characteristics of propeller-wing-flap configurations for vertical-and short-take-off-and-landing airplanes.
- [65] La Roche, U. and Palffy, S. (1996). Wing-grid, a novel device for reduction of induced drag on wings. In *ICAS PROCEEDINGS*, volume 20, pages 2303–2309.
- [66] Laurenceau, J. and Meaux, M. (2008). Comparison of gradient and response surface based optimization frameworks using adjoint method. In 49th AIAA/ASME/ASCE/AHS/ASC Structures, Structural Dynamics, and Materials Conference, 16th AIAA/ASME/AHS Adaptive Structures Conference, 10th AIAA Non-Deterministic Approaches Conference, 9th AIAA Gossamer Spacecraft Forum, 4th AIAA Multidisciplinary Design Optimization Specialists Conference, page 1889.
- [67] Lents, C. E., Hardin, L. W., Rheaume, J., and Kohlman, L. (2016). Parallel hybrid gas-electric geared turbofan engine conceptual design and benefits analysis. In 52nd AIAA/SAE/ASEE Joint Propulsion Conference, page 4610.
- [68] Loth, J. L. (2006). Vorticity cancellation at trailing edge for induced drag elimination. US Patent 7,134,631.
- [69] Malone, B. and Papay, M. (1999). Modelcenter: an integration environment for simulation based design. In *Simulation Interoperability Workshop*.
- [70] Marconi, P. and Thiriet, R. (2013). Method of optimizing the specific fuel consumption of a twin engine helicopter and twin engine architecture with control system for implementing it. US Patent App. 13/879,829.

- [71] Marek, C. J., Smith, T. D., and Kundu, K. (2005). Low emission hydrogen combustors for gas turbines using lean direct injection.
- [72] Martins, J. R. and Lambe, A. B. (2013). Multidisciplinary design optimization: a survey of architectures. AIAA journal, 51(9):2049–2075.
- [73] Martins, J. R., Sturdza, P., and Alonso, J. J. (2003). The complex-step derivative approximation. ACM Transactions on Mathematical Software (TOMS), 29(3):245– 262.
- [74] Martins, J. R. R. A. (2012). A short course on multidisciplinary design optimization. University of Michigan.
- [75] Maskell, E. and Spence, D. (1959). A theory of the jet flap in three dimensions. Proc. R. Soc. Lond. A, 251(1266):407–425.
- [76] Massey, B. S. (1971). Units, dimensional analysis and physical similarity.
- [77] Mazzuca, T. and Torre, M. (2008). The fremm architecture: a first step towards innovation. In *Power Electronics, Electrical Drives, Automation and Motion, 2008.* SPEEDAM 2008. International Symposium on, pages 574–579. IEEE.
- [78] McCormick, B. W. (1967). Aerodynamics of V/STOL flight. Courier Corporation.
- [79] McCullers, L. (1984). Aircraft configuration optimization including optimized flight profiles.
- [80] McLean, D. (2005). Wingtip devices: What they do and how they do it. In *Boeing* Performance and Flight Operations Engineering Conference.
- [81] Meaux, M., Cormery, M., and Voizard, G. (2004). Viscous aerodynamic shape optimization based on the discrete adjoint state for 3d industrial configurations. In 4th European Congress on Computational Methods in Applied Sciences and Engineering (ECCOMAS), Jyväskylä, Finland, July, pages 24–28.
- [82] Meher-Homji, C. B. and Prisell, E. (1999). Pioneering turbojet developments of dr. hans von ohain—from the hes 1 to the hes 011. In ASME 1999 International Gas Turbine and Aeroengine Congress and Exhibition, pages V001T01A005–V001T01A005. American Society of Mechanical Engineers.
- [83] Meier, N. (2005). Jet engine specification database. [online; accessed 2018-01-18] http://www.jet-engine.net/index.html.
- [84] Moore, K., Naylor, B., and Gray, J. (2008). The development of an open-source framework for multidisciplinary analysis and optimization. In 10th AIAA/ISSMO Multidisciplinary Analysis and Optimization Conference, Victoria, Canada. AIAA. AIAA 2008-6069.
- [85] Nam, T., Soban, D., and Mavris, D. (2005). Power based sizing method for aircraft consuming unconventional energy. In 43rd AIAA Aerospace Sciences Meeting and Exhibit, page 818.

- [86] Nguyen-Dinh, M., Peter, J., Sauvage, R., Meaux, M., and Désidéri, J.-A. (2014). Mesh quality assessment based on aerodynamic functional output total derivatives. *European Journal of Mechanics-B/Fluids*, 45:51–71.
- [87] Noesis, S. (2005). Optimus optimisation software tool. *Noesis Solutions*.
- [88] Ozcan, M. F., Chakraborty, I., and Mavris, D. N. (2016). Impact of subsystem secondary power requirements on gas turbine sizing and performance. In 16th AIAA Aviation Technology, Integration, and Operations Conference, page 3146.
- [89] Pacelab, A. (2012). Pacelab apd 3.0.0. PACE Aerospace Engineering and Information Technology GmbH.
- [90] Patterson, J. (1990). Wingtip vortex turbine. US Patent 4,917,332.
- [91] Patterson, M. D. (2016). Conceptual Design of High-Lift Propeller Systems for Small Electric Aircraft. PhD thesis, Georgia Institute of Technology.
- [92] Patterson, M. D., Derlaga, J. M., and Borer, N. K. (2016). High-lift propeller system configuration selection for nasa's sceptor distributed electric propulsion flight demonstrator. In 16th AIAA Aviation Technology, Integration, and Operations Conference, page 3922.
- [93] Patterson Jr, J. C. and Bartlett, G. R. (1985). Effect of a wing-tip mounted pusher turboprop on the aerodynamic characteristics of a semi-span wing. AIAA Paper, (85-1347).
- [94] Patterson Jr, J. C. and Bartlett, G. R. (1987). Evaluation of installed performance of a wing-tip-mounted pusher turboprop on a semispan wing.
- [95] Patterson Jr, J. C. and Flechner, S. G. (1985). Exploratory wind-tunnel investigation of a wingtip-mounted vortex turbine for vortex energy recovery.
- [96] Patterson Jr, J. C. and Jordan Jr, F. L. (1977). Thrust-augmented vortex attenuation.
- [97] Perullo, C., Trawick, D., Armstrong, M., Tai, J. C., and Mavris, D. N. (2017). Cycle selection and sizing of a single-aisle transport with the electrically variable engine (tm)(eve) for fleet level fuel optimization. In 55th AIAA Aerospace Sciences Meeting, page 1923.
- [98] Petrov, A. (2012). Aerodynamics of stol airplanes with powered high-lift systems. In Proceedings of the ICAS 2012 Congress, Brisbane, Australia.
- [99] Pornet, C., Gologan, C., Vratny, P. C., Seitz, A., Schmitz, O., Isikveren, A. T., and Hornung, M. (2014). Methodology for sizing and performance assessment of hybrid energy aircraft. *Journal of Aircraft*, 52(1):341–352.
- [100] Powell, M. J. (2009). The bobyqa algorithm for bound constrained optimization without derivatives. Cambridge NA Report NA2009/06, University of Cambridge, Cambridge, pages 26–46.

- [101] Prautzsch, H., Boehm, W., and Paluszny, M. (2013). Bézier and B-spline techniques. Springer Science & Business Media.
- [102] Prigent, S. (2015). Innovative and integrated approach for environmentally efficient aircraft design and operations. PhD thesis, ISAE-Institut Supérieur de l'Aéronautique et de l'Espace.
- [103] Raymer, D. P. (1996). An update on rds-professional. Technical report, SAE Technical Paper.
- [104] Raymer, D. P. (1999). Aircraft design: a conceptual approach, american institute of aeronautics and astronautics. *Inc.*, *Reston*, *VA*, 21.
- [105] Riley, D. R. (1951). Wind-tunnel investigation and analysis of the effects of end plates on the aerodynamic characteristics of an unswept wing.
- [106] Rodriguez, D. L. (2001). A multidisciplinary optimization method for designing boundary layer ingesting inlets. PhD thesis, Stanford University.
- [107] Roe, M. H., Renselaer, D. J., and Quam, R. A. (1973). Stol tactical aircraft investigation, externally blown flap. volume 2. design compendium. Technical report, ROCKWELL INTERNATIONAL LOS ANGELES CA LOS ANGELES AIRCRAFT DIV.
- [108] Roskam, J. (1985). Airplane design. DARcorporation.
- [109] Roskam, J. (1998). Airplane flight dynamics and automatic flight controls. DARcorporation.
- [110] Roskam, J. (2000). Airplane design part vi: Preliminary calculation of aerodynamic. Thrust and Power Characteristics, DARcorporation, Lawrence, Kansas, USA.
- [111] Roskam, J. and Lan, C.-T. E. (1997). Airplane aerodynamics and performance. DARcorporation.
- [112] Roskam, J., Malaek, S., and Anemaat, W. (1990). Aaa (advanced aircraft analysis): A user-friendly approach to preliminary aircraft design. *ICAS Paper*, pages 90–92.
- [113] Rostek, P. (2015). Hybrid electric propulsion. In A European Initiative for Technology Development, Electric & Hybrid Aerospace Technology Symposium.
- [114] Roth, B. and Giffin III, R. (2010). Fuel cell hybrid propulsion challenges and opportunities for commercial aviation. AIAA Paper, 6537:2010.
- [115] Rowan, T. (1990). The subplex method for unconstrained optimization. PhD thesis, Ph. D. thesis, Department of Computer Sciences, Univ. of Texas.
- [116] Sareni, B., Krahenbuhl, L., and Nicolas, A. (2000). Efficient genetic algorithms for solving hard constrained optimization problems. *IEEE Transactions on Magnetics*, 36(4):1027–1030.

- [117] Schaefer, C. G. J. (2017). Xv-24a lightningstrike, presentation to the ahs international 73rd annual forum and technology display, approved for public release.
- [118] Schiltgen, B. T. and Freeman, J. (2016). Aeropropulsive interaction and thermal system integration within the eco-150: A turboelectric distributed propulsion airliner with conventional electric machines. In 16th AIAA Aviation Technology, Integration, and Operations Conference, page 4064.
- [119] S.Gray, J., Mader, C. A., Kenway, G. K. W., and Martins, J. R. R. A. (2017). Modeling boundary layer ingestion using a coupled aeropropulsive analysis. *AIAA Journal of Aircraft*.
- [120] Siemens (2015). Electric propulsion components with high power densities for aviation. [online; accessed 2018-05-10] https://nari.arc.nasa.gov/sites/default/files/ attachments/Korbinian-TVFW-Aug2015.pdf.
- [121] Siemens, A. (2012). Electrical machine. DE Application DE102013205418A1.
- [122] Siemens, A. (2016). Aerobatic airplane "extra 330le" with world-record electric motor from siemens. [online; accessed 2018-05-10] https://www.siemens.com/press/pool/ de/events/2016/corporate/2016-12-innovation/inno2016-aerobatic-airplane-e.pdf.
- [123] Simos, D. (1990). Project interactive analysis and optimization-piano version 3.8. Lissys Limited, 2001.
- [124] Snyder, M. H., Zumwalt, G., et al. (1969). Effects of wingtip-mounted propellers on wing lift and induced drag. J. Aircraft, 6(5):392–397.
- [125] Standard, H. (1963). Generalized method of propeller performance estimation 1961-1963. Technical report, Hamilton Standard.
- [126] Stephenson, J. D. (1949). The effects of aerodynamic brakes upon the speed characteristics of airplanes.
- [127] Stoll, A. M. (2015). Comparison of cfd and experimental results of the leaptech distributed electric propulsion blown wing. In 15th AIAA aviation technology, integration, and operations conference, page 3188.
- [128] Stoll, A. M. and Veble Mikic, G. (2016). Design studies of thin-haul commuter aircraft with distributed electric propulsion. In 16th AIAA Aviation Technology, Integration, and Operations Conference, page 3765.
- [129] Storn, R. and Price, K. (1997). Differential evolution-a simple and efficient heuristic for global optimization over continuous spaces. *Journal of global optimization*, 11(4):341–359.
- [130] Svanberg, K. (1987). The method of moving asymptotes—a new method for structural optimization. International journal for numerical methods in engineering, 24(2):359–373.
- [131] Tedford, N. P. and Martins, J. R. (2010). Benchmarking multidisciplinary design optimization algorithms. Optimization and Engineering, 11(1):159–183.

- [132] ThreeDHorse (2018). 3d pratt & withney pw100 turboprop model. [online; accessed 2018-05-20] https://www.3dhorse.com/collections/3d-aircraft/products/ 3d-pratt-whitney-pw100-turboprop-model.
- [133] Torenbeek, E. (2013). Synthesis of subsonic airplane design: an introduction to the preliminary design of subsonic general aviation and transport aircraft, with emphasis on layout, aerodynamic design, propulsion and performance. Springer Science & Business Media.
- [134] Turpin, C., Astier, S., Roboam, X., Sareni, B., and Piquet, H. (2012). Systemic design methodologies for electrical energy systems: Analysis, synthesis and management, chapter 6.
- [135] Uranga, A., Drela, M., Greitzer, E. M., Titchener, N. A., Lieu, M. K., Siu, N. M., Huang, A. C., Gatlin, G. M., and Hannon, J. A. (2014). Preliminary experimental assessment of the boundary layer ingestion benefit for the d8 aircraft. In 52nd Aerospace Sciences Meeting, American Institute of Aeronautics and Astronautics, National Harbor, MD, pages 1–25.
- [136] Veres, J. P. and Thurman, D. R. (2010). Conceptual design of a two spool compressor for the nasa large civil tilt rotor engine.
- [137] Von Karman, T. (1946). On laminar and turbulent friction.
- [138] Walsh, P. P. and Fletcher, P. (2004). Gas turbine performance. John Wiley & Sons.
- [139] Walt, S. v. d., Colbert, S. C., and Varoquaux, G. (2011). The numpy array: a structure for efficient numerical computation. *Computing in Science & Engineering*, 13(2):22–30.
- [140] Welstead, J. and Felder, J. L. (2016). Conceptual design of a single-aisle turboelectric commercial transport with fuselage boundary layer ingestion. In 54th AIAA Aerospace Sciences Meeting, pages 2016–1027.
- [141] Wheeler, P. W., Clare, J. C., Trentin, A., and Bozhko, S. (2013). An overview of the more electrical aircraft. *Proceedings of the Institution of Mechanical Engineers*, *Part G: Journal of Aerospace Engineering*, 227(4):578–585.
- [142] Willis, R. (1963). Lightest weight gears. Product engineering, 5:64–75.

Appendix A

XMDO: optimisation features

A fully parametric optimisation case definition

A case study is described with the help of configuration files that can be seen as templates used by XMDO to generate the objects handled by the code. These objects, described in Section 4.2, are recalled in Figure A.1. At least three top level configuration files are needed to define an optimisation case: for the **Vehicle**, the **PerfoEval** and the **PerfoOptimWrapper** objects. The concept of configuration file is also extended to vehicle component, mission and segment providing high flexibility for defining an optimisation case.

As shown in Figure A.2, the configuration file of the vehicle can encapsulate configuration files of components, facilitating the vehicle modification (e.g. propulsion system) from one study to the other. In the same way, the configuration file of the **PerfoEval** object encapsulates configuration files of missions, which in turns, encapsulates configuration files of segments. The configuration file of the vehicle lists the components constituting the vehicle as well as the model name associated to each of them. The configuration file of a mission contains a reference to the vehicle evaluated over the mission, a description of the different segments composing the mission, the time-ordered sequence of segments and the name list of constraints calculated after mission evaluation. The configuration files of the evaluated missions. Finally, the configuration file used for the description of the **PerfoEval** object includes a reference to the configuration file of the **PerfoEval** object includes a reference to the configuration file of the **PerfoEval** object includes a reference to the configuration file of the **PerfoEval** object of interest, the names of the objective function and optimisation algorithm, and other parameters related to optimisation method tunings.



Fig. A.1 Main objects handled by XMDO

A parameter file can also be associated with each configuration file. Once an object has been built from the configuration file by XMDO, the default attributes of the object are overloaded with the parameter file. As illustrated in Figure A.2, it is therefore possible to generate two nominal missions of the flight distribution (say *Mission1* and *Mission2*) from the same configuration file (*MissionGen*), providing an identical structure (e.g. segment types and sequence) to both missions, and to overload the default parameters (e.g. range) from two different parameter files. In the parameter file, a given parameter can be fixed to a constant value, defined as a function of other parameters (e.g. propeller span-wise position according to its diameter) or set as a design variable. It is also important to note that any parameter file can be accessed by the lower level parameter files, which enables to define common parameters between those files (e.g. same flap deflection angle for all the take-off segments). The **PerfoOptimWrapper** object considers all the parameters defined as design variables in its **PerfoEval** object to perform the optimisation.



Fig. A.2 Illustration of the definition of a case study with configuration and parameter files

Optimisation methods

Since the design space exploration of unconventional vehicles may involve large numbers of design variables, XMDO was primarily developed to be used with gradient-based algorithms that are usually more efficient in such cases. However such methods require accurate and efficient gradient calculations for effective optimisation. In XMDO the derivatives for the construction of gradients or jacobians can be estimated with the Finite Differences or Complex Step methods. While the Finite Differences method is very easy to implement and does not require to modify the source code (i.e. even works with "black-box" models), it suffers from low accuracy due to the step-size sensitivity of the derivative approximation. Equation A.1 recalls the mathematical expression of a forward difference for evaluating the derivative of function f at variable x, where his the finite difference step-size and $\mathcal{O}(h)$ is the truncation error.

$$f'(x) = \frac{f(x+h) - f(x)}{h} + \mathcal{O}(h) \tag{A.1}$$

One may want to use a very small h to reduce the truncation error. But making h too small results in subtractive cancellation (i.e. f(x+h) - f(x)) due to the finite precision arithmetic and, consequently, yields very large approximation errors. The Complex-Step method, which uses complex variables to compute the derivatives of

real functions, enables to cope with the previous subtractive cancellation issue. The complex-step derivative approximation of f written in Equation A.2 can be obtained by taking the imaginary parts of both sides of a Taylor series expansion of f and dividing it by h.

$$f'(x) = \frac{Im[f(x+ih)]}{h} + \mathcal{O}(h^2)$$
(A.2)

Since Equation A.2 does not involve any subtraction operation, the only source of numerical error in this approximation is the truncation error $(\mathcal{O}(h^2))$ that is smaller than in Finite Differences $(\mathcal{O}(h))$. The comparison made by Martins et al. [73] shows the robustness of the Complex-Step approximation with decreasing step-size versus Finite Differences on a simple function (Figure A.3).



Fig. A.3 Relative error of the derivative versus decreasing step size [73]

Nonetheless, the implementation of the Complex-Step method requires the source code to be adapted. In the particular case of XMDO, all the functions and operators in the models must be made compatible with complex numbers. Other methods for computing derivatives that are not implemented in the current version of XMDO can be found out in Martins' course [74] providing a complete overview of the Multidisciplinary Design Optimisation discipline.

As already pointed out in Chapter 3 and similarly to many engineering design optimisation problems, XMDO must solve a constrained optimisation problem. Not to mention that the constraints involves in vehicle design are most often non-linear. In addition, the selected optimisation algorithm in XMDO must be able to handle bounds on the design variables (i.e. constraints on the design variables) in particular to avoid failures of the design process due to improper vehicle geometry description or trajectory definitions. All of these restricts the selection of the optimisation algorithms. For the purpose of this section, the statement of a general optimisation problem is recalled in Equation A.3, where f is the objective function, x the vector of design variables, g the vector of inequality constraints and h the vector of equality constraints.

minimise
$$f(x)$$

with respect to $x \in \mathbb{R}^n$
subject to $g_j(x) \le 0, \quad j = 1, \dots, l$
 $h_j(x) = 0, \quad j = 1, \dots, m$
(A.3)

The first optimisation studies of XMDO have been performed with the NLopt [59] and Scipy [60] packages, two of the numerous available optimisation librairies in Python. Of the local gradient-based optimisation algorithms provided by the NLopt library, only MMA (Method of Moving Asymptotes) and SLSQP (Sequential Least-Squares Quadratic Programming) support arbitrary nonlinear inequality constraints, and only SLSQP supports nonlinear equality constraints. One of the main advantages of these two algorithms is that they do not require any transformation of the inputs (i.e. objective function, design variables and constraints are provided as is to the algorithm). SLSQP is a sequential least squares programming algorithm which optimises successive second-order (quadratic/least-squares) approximations of the objective function, with first-order (linearised) approximations of the constraints. MMA also uses successive approximations of the problem: for each step of the iterative process, a strictly convex approximating subproblem is generated and solved. The generation of these subproblems is controlled by the so-called moving asymptotes, which both stabilise and speed up the convergence of the general process [130]. While SLSQP most always failed in solving the problem, MMA performed better but at a low rate of convergence, pushing the developer to consider other algorithms of Scipy. However these other gradient-based algorithms are only able to solve unconstrained optimisation problems requiring the problem to be transformed. Using penalty methods is one of the usual ways of doing this. A penalty method replaces a constrained optimisation problem by a sequence of unconstrained problems that are formed by adding to the initial

objective function a penalty function ϕ for constraint violation multiplied by a penalty parameter ρ , defining the new objective function π of Equation A.4.

$$\pi(x,\rho) = f(x) + \rho\phi(x) \tag{A.4}$$

Exterior and interior penalties are the two main types of penalisation methods and differ from each other by the formulation of ϕ [74]. In the first case, the penalty function ϕ increases as far as x deviates from the feasible domain and the solution $x^*(\rho)$ of the modified problem approaches the solution x^* of the initial problem as ρ tends to infinity. In the latter case exploration points are forced to remain interior to the feasible domain and ϕ increases as far as x approaches the boundary of an inequality constraint. For interior penalties, $x^*(\rho)$ approaches x^* as ρ tends to zero. But starting with a too high or too low value of ρ for exterior and interior penalties, respectively, raises ill-conditioning issues making the use of gradient-based algorithms inappropriate. Running such algorithms with penalty methods usually require an iterative change of ρ with successive solving of unconstrained problems. For exterior penalties, the iterative process ensures that the solution $x^*(\rho)$ of the modified problem falls into the feasible domain of the initial problem. In the case of internal penalties, it stops when the rate of improvements of $x^*(\rho)$ and $f(x^*(\rho))$ stay within the desired tolerances. Since external penalties requires π to be continuous over the exploration domain, interior penalties are therefore more suitable when the objective function f is not defined in the non-feasible domain, but require a feasible solution as a starting point which can not always be provided for hard constrained optimisation problems. In the case of XMDO, the models have been made compatible with the evaluation of fin the non-feasible domain under the condition that the design variables stay within their bounds. As several gradient-based algorithms include variable bounds handling for solving unconstrained optimisation problem, external penalties are implemented in XMDO. Of these algorithms tested by the developer, L-BFGS-B of Scipy was one of the most successful in solving XMDO studies. L-BFGS-B (Limited-memory BFGS) is based on the Broyden–Fletcher–Goldfarb–Shanno (BFGS) algorithm which is generally considered to be the most effective quasi-Newton methods [74]. According to the developer, the use of L-BFGS-B on XMDO studies showed a much higher convergence rate than MMA but also a better exploration of the design space thanks to the external penalty method. This observation is most probably linked to the particular shape of the feasible design space that is not a one block subset of the design space but rather a set of subsets for the studied optimisation cases. The use of external penalty methods then helps exploring the design space by allowing the design point to be between

feasible regions. However such feasible design space may also characterise the presence of multimodality in the design space which does not favour the use of gradient-based algorithms.

Appendix B

Basics of B-splines

The definition of a B-spline curve involves a set $\{P_0, P_1, ..., P_n\}$ of n+1 control points of an euclidian space \mathbb{E} , a knot vector $U = (u_0, u_1, ..., u_m)$ of m+1 knots of \mathbb{R} provided in increasing order, and a degree p that must satisfy m = n + p + 1. The B-spline curve C that defines a function from $[u_0, u_m]$ to \mathbb{E} is a linear combination of B-spline basis functions of the same degree as expressed in Equation G.9. The *i*-th B-spline basis function of degree p, written as $B_{i,p}(u)$, is defined by Equation B.2. Note that p must be at least 1 to ensure the continuity of C (piecewise linear function), and can be no more than n.

$$C(u) = \sum_{i=0}^{n} B_{i,p}(u) P_i$$
 (B.1)

$$B_{i,0}(u) = \begin{cases} 0 & \text{if } u_i \le u < u_{i+1} \\ 1 & \text{otherwise} \end{cases}$$

$$B_{i,p}(u) = \frac{u - u_i}{u_{i+p} - u_i} B_{i,p-1}(u) + \frac{u_{i+p+1} - u}{u_{i+p+1} - u_{i+1}} B_{i+1,p-1}(u)$$
(B.2)

The derivative of a B-spline curve of degree p, shown in Equation B.3, is another B-spline curve of degree p-1 with a new set of n control points $\{Q_0, Q_1, ..., Q_{n-1}\}$ defined by Equation B.4. It is an interesting property for the purpose of trajectory or control laws modelling since first or higher order derivatives of a B-spline curve can be evaluated easily.

$$\frac{\mathrm{d}}{\mathrm{d}u}C(u) = C'(u) = \sum_{i=0}^{n-1} B_{i+1,p-1}(u)Q_i$$
(B.3)

$$Q_i = \frac{p}{u_{i+p+1} - u_{i+1}} (P_{i+1} - P_i)$$
(B.4)

A clamped B-spline curve passes through its first and last control points and is tangent to the first and last legs of its control polyline such as illustrated in Figure B.1. This curve is obtained by giving a multiplicity p+1 to the first knot and the last knot (i.e. $u_0 = u_1 = ... = u_p$ and $u_{m-p} = u_{m-p+1} = ... = u_m$). The behaviour of a clamped B-spline curve is then perfectly known at the extremities (i.e. $u = u_0$ and $u = u_m$). Also, note that a clamped B-spline curve with n = p and 2(p+1) knots reduces to a Bézier curve [101].



Fig. B.1 Example of a clamped B-spline curve

The knot vector U of a clamped B-spline curve can also be characterised according to its uniformity if n > p. The knot vector is said uniform if the knots between u_p and u_m are uniformly spaced. If it is not the case, the knot vector is said non-uniform. The control of a B-spline curve of uniform knot vector is mainly achieved through the control points. While the uniform knot distribution provides enough flexibility in most cases, moving the knots of a non-uniform knot vector may be required to give additional degrees of freedom to control the curve.

Appendix C

Aerodynamic model

This appendix deals with the models that have been developed for the prediction of aerodynamic forces acting on the aircraft components. Most of the next equations are based on classical methods that can be found in any aircraft design handbook such as [133], [111], [104] or [62].

Power-off lift

This section focuses on the lift force generated by a lifting surface in the absence of propeller slipstream interaction. Only the force generated by the wings, the horizontal stabiliser and the vertical stabiliser are taken into account.

Wings

The power-off lift $L_{W_{p-off,0}}$ created by the wing in clean configuration (i.e. flaps retracted) is estimated thanks to Equation C.1. In this expression, S_W is the wing planform surface area, α is the angle of attack defined relatively to the aircraft body x-axis, and α_0 is the zero-lift angle of attack. The lift curve slope $C_{l_{\alpha,W}}$ of the wing, written in Equation C.2, is evaluated from the 2D lift curve slope $C_{l_{\alpha,2D}}$ (Equation C.3) of an aerofoil, the wing aspect ratio λ (i.e. b_W^2/S_W) and the span efficiency factor e_b (Equation C.4).

$$L_{W_{p-off,0}} = \frac{1}{2} \rho V^2 S_W C_{l_{\alpha,W}}[\alpha - \alpha_0]$$
 (C.1)

$$C_{l_{\alpha,W}} = \frac{C_{l_{\alpha,2D}}}{1 + 57.3C_{l_{\alpha,2D}} / (\pi \lambda e_b)}$$
(C.2)

$$C_{l_{\alpha,2D}} = \frac{2\pi}{\sqrt{1-M^2}}$$
 (C.3)

$$e_b = \frac{2}{2 - \lambda + \sqrt{4 + \lambda^2}} \tag{C.4}$$

The power-off lift $L_{W_{p-off,\delta}}$ created by the wing with flaps deflected of an angle δ is evaluated from the analytical method proposed by Roskam [110]. According to this method, the wing lift can be written such as in Equation C.5, where both the lift curve slope correction factor K_{δ} and the lift offset ΔL_{δ} depend on the wing geometry, the flap geometry and the flap deflection angle. Additionally, ΔL_{δ} evolves with flight conditions, such as $L_{W_{p-off,0}}$. These parameters are evaluated thanks to the charts for single slotted flaps provided in [110].

$$L_{W_{p-off,\delta}} = K_{\delta} L_{W_{p-off,0}} + \Delta L_{\delta} \tag{C.5}$$

In this thesis the lift curve slope is supposed to be linear up to the maximum angle of attack α_{max} , which is the maximum allowable angle of attack in flight. Although the stall angle of attack evolves with the Mach number in reality, α_{max} is assumed constant here.

Horizontal stabiliser

In trimmed flight, the horizontal stabiliser usually generates lift that can be positive or negative depending on the position of the wing aerodynamic centre of pressure versus the aircraft centre of gravity (CG). However, the tail plane lift is supposed to be zero in this model for two reasons:

- the pitching moment of the wing is neglected.
- the aircraft CG is supposed to be aligned with the quarter of the wing MAC, which is the simplest estimation of the position of the wing centre of pressure.

Although the tail plane contribution to lift is neglected, a sizing criterion is required. The classical method consists in using a tail volume coefficient V_H [109] defined by Equation C.6, where S_H is the horizontal tail area, MAC_W is the wing MAC, and d_H is the distance from the aircraft CG to the 25% of the tail MAC. This volume coefficient is derived from the equilibrium of the pitching moment for the worst CG position and the worst flight conditions requiring high lift demand to the tail plane. The sizing scenario usually occurs in approach phase.

$$V_H = \frac{d_H S_H}{MAC_W S_W} \tag{C.6}$$

The tail volume coefficient method requires the derivative aircraft to feature the same CG range as the reference aircraft, but also the same level of wing lift coefficient. Since the lift coefficient of blown wings is expected to be much larger than that of classical wings, the modified tail volume coefficient \overline{V}_H (Equation C.7) is introduced.

$$\overline{V}_H = \frac{d_H S_H}{MAC_W L_W / (0.5\rho V^2)} \tag{C.7}$$

By using the wing lift L_W in the expression of \overline{V}_H , the modified tail volume coefficient actually limits the maximum lift coefficient for the tail plane accordingly. \overline{V}_H is therefore calculated at each evaluation point of the mission where the aircraft is airborne, and the tail plane is correctly sized if Equation C.8 is satisfied, whit $\overline{V}_{H_{min}}$ being estimated from existing aircraft.

$$\forall t, \quad \overline{V}_H(t) \ge \overline{V}_{H_{min}} \tag{C.8}$$

Vertical stabiliser

In this model, the lift force generated by the vertical stabiliser that is normal to the planform is evaluated according to Equation C.9, expressing the equilibrium of the yaw moment based on the contribution of thrust forces only. In this formula, T_C is the thrust (positive or negative) generated by component C and z_C is the spanwise coordinate of the force application point.

$$L_V d_V = \sum_C T_C z_C \tag{C.9}$$

The two volume coefficients defined in Equation C.10 and C.11 are commonly used in preliminary aircraft design studies [109] for the vertical tail plane sizing. In these equations, S_V is the vertical tail area and d_V is the distance from the aircraft CG to the quarter of the fin MAC. The volume coefficient V_{V_s} relates to stability criteria (e.g. Dutch Roll) while V_{V_c} expresses controllability criterion with regards to thrust asymmetry.

$$V_{V_s} = \frac{d_V S_V}{b_W S_W} \tag{C.10}$$

$$V_{V_c} = \frac{d_V S_V}{\left|\sum_C T_C z_C\right|} \tag{C.11}$$

A reference value of 0.02, derived from glider aircraft, is chosen for V_{V_s} . The reference value $V_{V_{c_{min}}}$ for V_{V_c} is evaluated from existing turboprop aircraft. Finally, in this model, the vertical tail plane is correctly sized if Equations C.12 and C.13 are satisfied, with V_{V_c} being calculated at each evaluation point where the aircraft is airborne.

$$V_{V_s} \ge 0.02 \tag{C.12}$$

$$\forall t, \quad V_{V_c}(t) \ge V_{V_{c_{min}}} \tag{C.13}$$

Power-off drag

This section addresses the calculation of the drag force D_C acting on a component C in the absence of propeller slipstream interaction. In this model the drag force is divided into:

- D_{C_f} : the friction drag generated by skin friction between the molecules of the air and the solid surface
- D_{C_p} : the viscous pressure drag (or form drag) arising from the varied pressure distribution which depends on the body shape
- D_{C_a} : the additional drag (or excrescence drag) created by surface imperfections (e.g. panel joints, gaps around doors and control surfaces, etc...).
- D_{C_i} : the induced drag (or lift-induced drag) resulting from the creation of lift on a three-dimensional lifting body (e.g. wing, tail plane, etc...).

A particular focus is also given on the wing drag estimation model including the flap deflection, which is the direct application of the analytical method proposed by Roskam [110]. In order to distinguish the wing case from the other components in this section, the wing power-off drag with flap retracted and with flap deflected are noted $D_{W_{p-off,0}}$ and $D_{W_{p-off,0}}$, respectively.

Friction drag

 D_{C_f} is calculated according to Equation C.14, where ρ is the air density, V the freestream velocity, f_p the experimentally-derived friction coefficient (Equation C.15) for fully turbulent flat plates [62][7], and $S_{C_{wet}}$ the wetted area of the component.

$$D_{C_f} = \frac{1}{2}\rho V^2 f_p S_{C_{wet}} \tag{C.14}$$

$$f_p = \frac{0.455}{\log^{2.58} Re} \tag{C.15}$$

The Reynolds number Re of Equation C.15 is evaluated thanks to Equation C.16, where μ is the air viscosity and l_C is a characteristic linear dimension of the component.

$$Re = \frac{\rho V l_C}{\mu} \tag{C.16}$$

For bodies (e.g. fuselage, nacelle), the wetted area $S_{C_{wet}}$ is computed by adding the contribution of the nose cone (length l_{nose}), constant section (diameter D) and tail cone (length l_{tail}). In order to account for curved shapes against the conical modelling of bodies, the wetted areas of the nose and tail sections are supposed to be $0.75\pi D l_{nose}$ and $0.72\pi D l_{tail}$, respectively, as suggested in [62]. Also, the characteristic length l_C is the overall length of the body. For aerodynamic surfaces (e.g. wing, tail, fin) the wetted area $S_{C_{wet}}$ is calculated with Equation C.17 (modified from [62]), where t_{rav} is the average relative thickness of the surface and S_C is the planform area. For these components, l_C is the MAC of the surface.

$$S_{C_{wet}} = 2(1 + 0.2t_{r_{av}})S_C \tag{C.17}$$

Note that the model uses gross wetted areas, meaning that the overlapping areas at component intersections (wing-fuselage, fin-fuselage rear end, etc...) are note removed from $S_{C_{wet}}$. On the other hand, the resulting drag estimation should not be too conservative since surfaces such as fairings (wing root, landing gear) are not considered.

Wing friction drag with flaps deflected

By assuming that the wetted area does not change with flap deflection, the model uses Equation C.18, where $D_{W_{f_{p-off,0}}}$ is the friction drag component of $D_{W_{p-off,0}}$ obtained from Equation C.14.

$$D_{W_{f_{p-off,\delta}}} = D_{W_{f_{p-off,0}}} \tag{C.18}$$

Viscous pressure drag

 D_{C_p} is calculated as a function of D_{C_f} with a so called form factor k_C (Equation C.19). Typical values for k_C range from 0.1 to 0.6 depending on the shape of the component.

$$D_{C_p} = k_C D_{C_f} \tag{C.19}$$

Wing viscous pressure drag with flaps deflected

The viscous pressure drag $D_{W_{p_{p-off,\delta}}}$ for the wing with flaps deflected is calculated from Equation C.20, where $D_{W_{p_{p-off,0}}}$ is defined by Equation C.19, and $\Delta D_{W_{p_{p-off,\delta}}}$ is calculated according to the charts of [110] for single slotted flaps. The calculation of this latter parameter depends on wing geometry, flap geometry, flap deflection angle and flight conditions.

$$D_{W_{p_{p-off,\delta}}} = D_{W_{p_{p-off,0}}} + \Delta D_{W_{p_{p-off,\delta}}}$$
(C.20)

Additional drag

 D_{C_a} is expressed as in Equation C.21, where r_C is a constant characterizing the surface imperfections. In the models of the thesis r_C varies from 0 to 0.45.

$$D_{C_p} = r_C (D_{C_f} + D_{C_p}) \tag{C.21}$$

Wing additional drag with flaps deflected

A similar formula written in Equation C.22 is used for the wings with flaps deflected. Note that r_W is assumed δ -independent.

$$D_{W_{p_{p-off,\delta}}} = r_W(D_{W_{f_{p-off,\delta}}} + D_{W_{p_{p-off,\delta}}})$$
(C.22)

Induced drag

Because no lift force is generated by the horizontal stabiliser in this model, only the induced-drag estimation formulas for the vertical tail plane and the wings are shown.

Vertical stabiliser

The induced drag D_{V_i} generated by the vertical stabiliser lift L_V is given by Equation C.23, derived from the simplest expression of drag due to lift assuming an Oswald coefficient equal to unity.

$$D_{V_i} = \frac{2}{\rho S_V V^2} \frac{L_V^2}{\pi \lambda_V} \tag{C.23}$$

Wings

The lift induced-drag of the wings is estimated by adding up a base induced-drag term and a penalty term coming from the modification of the lift distribution with flap deflection, as written in Equation C.24.

$$D_{W_{i_{p-off,\delta}}} = \underbrace{(1+\Delta_i)\frac{2}{\rho S_W V^2} \frac{L^2_{W_{p-off,\delta}}}{\pi \lambda_W}}_{\text{base induced-drag [133]}} + \underbrace{\frac{2}{\rho S_W V^2} [K_i (L_{W_{p-off,\delta}} - L_{W_{p-off,0}})]^2}_{\text{lift-distribution change penalty [110]}}$$
(C.24)

The base term is calculated from the method of Garner [46], that is also discussed with other methods by Torenbeek [133]. According to Garner, Δ_i depends on the spanwsise position of the center of pressure which is analytically evaluated from the Diederich's method [30][133] in this thesis. The lift induced-drag penalty term with flaps deflected is still calculated from the method of [110] for single slotted flaps. The factor K_i is a function of wing and flap geometries. In clean configuration (i.e. $\delta=0^\circ$), Equation C.24 then simply reduces to the base term.

Summary

All the equations required for the generation of aircraft drag polars under power-off condition have been presented in this section. It should be noted that neither the ground effect nor the landing gear drag are included in this model.

Although the factors k_C and r_C that have been used in the thesis are not written in the manuscript for confidentiality reasons, all the four references quoted at the beginning of the appendix provide estimation guidelines.

Finally, since the thesis focuses on the evaluation of blown wing concepts, modifications of the previous equations are required to take into account the wing-propeller slipstream interactions. The selected method is presented in the next section, and uses
the power-off lift and power-off drag forces as inputs. Equation C.25 recalls the general formula for the wing power-off drag, which holds for clean or low-speed configurations.

$$D_{W_{p-off}} = D_{W_{f_{p-off}}} + D_{W_{p_{p-off}}} + D_{W_{a_{p-off}}} + D_{W_{i_{p-off}}}$$
(C.25)

Wing power-on aerodynamic forces: blown wing effect

During the thesis several analytical models have been looked at and tested for the prediction of power-on aerodynamic forces. Among the most famous models, one can first mention the jet flap theory introduced by Maskell and Spence [75], that was used and derived in several publications. For examples, Roe et al. [107] extend the theory to externally blown flap and McCormick [78] combines the jet flap theory with flow momentum theory to assess the aerodynamic characteristics of a propeller/wing combination. In 1959, NASA published a methodology based on flow momentum theory for estimating the lift and drag characteristics of a propeller-wing-flap configuration [64]. More recently, this method was updated by ESDU and made available in [36]. Finally, Patterson [91] also published an analytical model based on CFD results for a small aircraft equipped with high-lift propellers.

The method provided by ESDU 88031 has been selected as it is one of the most advanced analytical method that can be applied in every phase of flight. While the initial method assumes identical wing-mounted propellers, a generalisation is proposed below to cover the propeller installations of the thesis.

Generalisation of ESDU 88031

The final formula given by the ESDU method for the prediction of lift and longitudinal forces on propeller/nacelle/wing/flap systems uses dimensionless parameters and can only be applied if all the propellers are identical and produce the same thrust. The proposed generalisation is based on the same definitions and momentum theory equations, but uses forces instead. Similarly to the initial method, it is assumed that each propeller streamtube behaves independently of the other ones.

By using the same notations with an additional subscript i for the parameters related to propeller #i, the following equations for force and momentum resolved in the X- and Z- directions (Figure C.1) are implemented.

• The surface area A_w is defined by Equation C.26.

$$A_w = \pi b^2 / 4 - \sum_i A_{s_i}$$
 (C.26)

• The mass-flow through A_w then becomes Equation C.27.

$$m_w = \rho V_0 b^2 / 4 - \sum_i m_{s_i} \tag{C.27}$$

• Forces for the wing streamtube follows the same expressions as in the ESDU method. In these formulas, recalled by Equation C.28, k_w and θ_w are evaluated according to the power-off forces (i.e. $L_{W_{p-off}}$ and $D_{W_{p-off}}$) in the same flap configuration.

$$X_w = m_w V_0[(1 - k_w)\cos\theta_w - 1] \quad Z_w = m_w V_0(1 - k_w)\sin\theta_w$$
(C.28)

• Forces for the set of propellers alone are evaluated from Equation C.29; where T_i is the value returned by the propeller performance model if positive, and zero otherwise.

$$X_p = \sum_i T_i \cos \alpha_{p_i} \quad Z_p = \sum_i T_i \sin \alpha_{p_i} \tag{C.29}$$

• Forces for the propeller streamtubes as a whole then follow Equation C.30. Similarly to the initial method, V_{s_i} is calculated from the actuator disk theory (recalled in [36]), and k_{s_i} and θ_{s_i} are evaluated thanks to the ESDU analytical formulas involving experimental-derived data but also the power-off forces in clean configuration (i.e. $L_{W_{p-off,0}}$ and $D_{W_{p-off,0}}$).

$$X_{p+s} = \sum_{i} m_{s_i} [(1 - k_{s_i}) V_{s_i} \cos \theta_{s_i} - V_0] \quad Z_{p+s} = \sum_{i} m_{s_i} (1 - k_{s_i}) V_{s_i} \sin \theta_{s_i} \quad (C.30)$$

• Finally, by using the directions of the aerodynamic reference frame of the thesis, the power-on forces acting on the wings are defined through Equation C.31.

$$L_{W_{p-on}} = Z_w + Z_{p+s} - Z_p \quad D_{W_{p-on}} = -(X_w + X_{p+s} - X_p) \tag{C.31}$$

As can be seen from the previous equations, the proposed generalised method exactly reduces to the ESDU method with identical propellers generating the same thrust (i.e. $\forall i, \alpha_{p_i} = \alpha_p, T_i = T$ and $D_i = D$). Also, if the ESDU method is compared with test data in [36] and shows predictions of the lift and longitudinal force within 10% and 15%, respectively, it should be noted that these test data do not involve more than four propellers. Hence, such level of accuracy with more propellers can neither be guaranteed with the initial method nor with the generalised one. Still, both the physically-based construction and the analytical formulation make the proposed generalised method one of the best candidate for the consideration of the blown wing effect in every phase of flight in this thesis.





Appendix D

Turboshaft model

This appendix deals with the models that have been developed for the prediction of engine mass, maximum available power and specific fuel consumption according to the engine geometry. These models are based on several reference engines that were designed with an in-house engine preliminary design tool using published component maps. After presenting the design assumptions and the design outputs for the reference engines, the surrogate models constituting the turboshaft model of this thesis are detailed.

Reference Engines

The in-house engine preliminary design tool was used to design six different turboshaft engines with mechanical power ranging from 1,000 hp to 6,000 hp at 20,000 ft, Mach 0.45 and ISA conditions. Because engine configuration is also an important degree of freedom for engine optimisation on such wide range of power, two configurations were considered:

• AC: a two-spool architecture where the high-pressure spool comprises an axialcentrifugal compressor and a high pressure turbine driving the compressor stages. The combination of axial and centrifugal compressors offers a good compromise between high pressure ratio, high efficiency and limited length. A higher level of efficiency can theoretically be reached by a full axial compressor of same pressure ratio but the blade height of the last stages would be so small that flow turbulence imparted by required clearance between blade tips and casing would degrade its performance, not to mention the mechanical design and manufacturing limitations. The low pressure-shaft consists of a low pressure turbine driving the load. Such two-spool architecture with axial-centrifugal compressor is implemented for examples on the General Electric CFE738 turbofan, and on the General Electric T700 turboshaft illustrated in Figure D.1. A 3D rendering of such architecture from the preliminary design tool is shown in Figure D.3.



Fig. D.1 Cutaway view of the General Electric T700 turboshaft [49]

• **BAC**: a boosted two-spool architecture which includes a low-pressure compressor mounted on the low-pressure shaft and driven, like the load, by the low-pressure turbine. This low-pressure compressor, also known as booster, supercharges the high-pressure compressor helping to raise the overall pressure ratio of the engine cycle, and thus, its efficiency. Contrary to the boosted two-spool architecture depicted in Figure D.2, the studied **BAC** configuration still features an axialcentrifugal compressor on the high pressure spool.

For both architectures, the high-pressure spool also incorporates handling bleed valves that are used only in case of overloading, in particular during idle operations. Finally, these engines were designed without accounting for any bleed or mechanical off-takes: in the thesis, it is assumed that non-propulsive power can be drawn from the gas turbine output shaft (i.e. low-pressure shaft) for example.

Three engines of different power capability for each configuration were designed, as listed in Table D.1. The design assumptions and the design process are described next.

Each engine is optimised on a single design point defined at 20,000 ft, Mach 0.45 and ISA conditions. The design variables are:

- W_F : the fuel mass flow rate
- W_0 : the intake air mass flow rate (Figure D.3)
- PR_{HPC_a} : the pressure ratio of the axial part of the high-pressure compressor
- PR_{HPC_c} : the pressure ratio of the centrifugal part of the high-pressure compressor



Fig. D.2 Schematic drawing of a boosted two-spool turbofan [1]

- PR_{LPC} : the pressure ratio of the booster (for BAC configuration only)
- T_4 : the temperature at the combustion chamber exit (Figure D.3)
- N_{HP} : the rotation speed of the high-pressure spool
- N_{LP} : the rotation speed of the low-pressure spool

In this optimisation problem, several constraints must be satisfied:

• The pressure ratios cannot exceed maximum values based on the number of stages and/or compressor type (i.e. axial or centrifugal), as expressed by Equations D.1, D.2 and D.3.

$$PR_{HPC_a} \le PR_{HPC_a,max} \tag{D.1}$$

$$PR_{HPC_c} \le PR_{HPC_c,max} \tag{D.2}$$

$$PR_{LPC} \le PR_{LPC,max} \tag{D.3}$$

• Since the external and internal radii of the different sections are calculated by the tool from the above inputs, the last stage blade span h_b (Figure D.3) of the axial part of the high-pressure compressor must be higher than a minimum achievable value $h_{b,min}$. Veres and Thurman [136] points out an acceptable design minimum of 11.2 mm, which was chosen for $h_{b,min}$. Then, Equation D.4 must be satisfied

at the optimum design.

$$h_b \ge h_{b,min}$$
 (D.4)

• The rotation speeds of the shafts must not lead to blade tip over-speeds. The tip speeds V_{LPT} and V_{HPC} , defined in Figure D.3, are selected to limit the rotation speeds of the low-pressure and high-pressure spools, respectively. Hence, these speeds must satisfy Equations D.5 and D.6, where $V_{LPT,max}$ and $V_{HPC,max}$ are based on design practices.

$$V_{LPT} \le V_{LPT,max} \tag{D.5}$$

$$V_{HPC} \le V_{HPC,max} \tag{D.6}$$

• Because of material limitations the temperature T_4 cannot exceed a maximum value T_{4max} . In this study, the maximum allowable temperature in continuous operation for future gas turbines is supposed to be 1,850 K, leading to Equation D.7.

$$T_4 \le T_{4max} \tag{D.7}$$

• Since each engine is designed to provide a given mechanical power SHP_{design} at the design conditions, the set of optimum design variables must allow the engine to effectively produce this power. The output power SHP returns by the engine cycle solving at the design point must then meet Equation D.8.

$$SHP = SHP_{design}$$
 (D.8)

• In this study, it was also stated that the optimum turboshaft design is achieved when the net thrust F_n produced by the exhaust gas at the design conditions is zero. Therefore, Equation D.9 must be satisfied at the design point.

$$F_n = 0 \tag{D.9}$$

Because the engine will not only operate at the design conditions, two off-design points referring to take-off conditions are also included in the design process: ISA, sea level, static, and ISA+15, sea level, Mach 0.25. For these two set of conditions, the engine cycle is solved for $T_4 = T_{4_{max}}$ based on the geometry resulting from the design point analysis. Fuel flow, rotation speeds, pressure ratios, thrust and shaft power are then outputs of such off-design studies. The two additional operating points are used to ensure that Equations D.1-D.3 and Equations D.5-D.6 still hold in these off-design conditions.



Fig. D.3 Definition of some parameters used in the engine optimisation problem based on the 3D view of an AC turboshaft modelled in the in-house preliminary design tool

The objective function of this optimisation problem is to minimise W_F at the design point which also relates to the minimisation of the specific fuel consumption SFCdefined as the ratio of fuel mass flow rate to output shaft power. Some of the most important results are grouped in Table D.1 and the definitions for the length and diameters can be found in Figure D.3.

Name	Conf.	$\begin{array}{c} SHP_{design} \\ (hp) \end{array}$	SFC_{design} (lb/hr/hp)	$N_{LP_{design}}$ (rpm)	$\frac{SHP_{SLS}{}^{a}}{(hp)}$	L (m)	$\begin{array}{c} D_{max} \\ (\mathrm{m}) \end{array}$	D_{in} (m)
AC1000		1,000	0.418	18,842	2,246	1.05	0.60	0.22
AC2000	AC	2,000	0.366	15,369	4,216	1.27	0.64	0.26
AC4000		4,000	0.351	11,086	8,577	1.74	0.84	0.36
BAC2000		2,000	0.377	$11,\!400$	4,788	1.51	0.66	0.30
BAC3000	BAC	$3,\!000$	0.340	11,843	8,636	1.85	0.68	0.40
BAC6000		6,000	0.302	12,106	$14,\!304$	2.20	0.77	0.48

Table D.1 Comparison of the reference turboshaft engines

 $^a\mathrm{ISA}$ sea level static

Based on an efficiency criteria, it seems interesting to switch from the AC configuration to the BAC for a gas turbine size of approximately 2,150 hp at design point, as can be seen from Figure D.4. Since efficiency optimisation was selected for the design of the reference engines, but also because the generation of the complete performance charts (i.e. off-design) was time-limited, only three of the best candidates were selected for the generation of the estimation models: AC1000, AC2000 and BAC6000. For these three engines, off-design performance data were produced by varying the temperature T_4 and solving the engine cycle equilibrium for multiple flight conditions defined by an altitude between sea level and 30,000 ft, and a Mach number ranging from 0. to 0.6.



Fig. D.4 Specific fuel consumptions of reference engines at design point

Geometry

The external envelope of the gas turbine is modelled by a cylinder of diameter D_{gt} and length L_{gt} , as illustrated in Figure D.5. In order to account for a potential radial air intake, such as in Figure D.5, L_{gt} is taken equal to the turboshaft length L plus the inlet diameter D_{in} . Also, to include space for the installation of fuel systems, accessory gearbox or other equipments in the turboshaft external envelope, D_{gt} is chosen equal to $1.174D_{max}$.

In order to reduce the number of inputs for the geometry model, L_{gt} is defined as a function of D_{gt} . The geometry model also returns the power P_{SLS} , that is equivalent to SHP_{SLS} in Table D.1 but expressed in SI unit. Finally, the optimum rotation speed $N_{LP_{design}}$ of the low-pressure shaft at design point is also given by the geometry model in rpm. The estimation models for L_{gt} , P_{SLS} and $N_{LP_{design}}$ were built from the selected reference engines (i.e. AC1000, AC2000 and BAC6000) and are written in Equations D.10, D.11 and D.12.



Fig. D.5 Geometry assumptions based on a 3D rendering of a PW100 engine [132]

$$L_{gt} = 14.063(D_{gt}/1.174)^2 - 10.938(D_{gt}/1.174) + 2.770$$
 (D.10)

$$P_{SLS} = [1.237(D_{gt}/1.174)^2 - 1.166(D_{gt}/1.174) + 0.271] \times 10^8$$
 (D.11)

$$N_{LP_{design}} = [184.849(D_{gt}/1.174)^2 - 311.100(D_{gt}/1.174) + 139.190] \times 10^3$$
(D.12)

Mass

With regards to the difficulty for predicting accurately the mass of a turboshaft, the mass model uses a simple specific power density assumption. Based on the ISA sea level static power P_{SLS} of the turboshaft engine, a power density of 9.9 kW/kg (i.e. 6hp/lb) relating to future advances in gas turbine technology is used for the mass estimation.

Specific fuel consumption

The following model was developed to capture the engine size effect on its maximum efficiency but also the efficiency degradation with part-load operation. This surrogate model is based on the performance data of the AC1000, AC2000 and BAC6000 turboshaft engines under ISA conditions. Figure D.6 plots them for all the flight conditions and power levels that were simulated with the engine preliminary design tool. Note that idle operations were studied into details for the AC2000 only which explains why idle points are not shown for the two other engines. Nonetheless, based on

Figure D.6, one can predict the possibility to build a transformation method between the performance data of the three engines.



Fig. D.6 SFC values of the engine performance charts (design and off-design)

The first step of the surrogate model development consisted in using a reference engine rating, which is an engine operating mode at a constant temperature T_4 , to build the dimensionless parameters X_{SFC} and X_P for the specific fuel consumption SFC and the shaft power P, respectively. The reference rating of this model, called MCL, is the rating for which $T_4 = T_{4_{design}}$. These two dimensionless parameters are defined in Equations D.13 and D.14, where M is the Mach number and z the altitude.

$$X_{SFC}(z,M) = \frac{SFC(z,M)}{SFC_{MCL}(z,M)}$$
(D.13)

$$X_P(z,M) = \frac{P(z,M)}{P_{MCL}(z,M)}$$
(D.14)

The dimensionless parameters associated to the flight conditions evaluated for the reference engines are depicted in Figure D.7. For a given X_P , there exist different values of X_{SFC} depending on the flight conditions (i.e. M and z).

To cope with this, the referred parameters defined in [138] and based on the Buckingham PI theorem [76] were used. Referred parameters involve δ and θ defined in Equations D.15 and D.16, where P_t and T_t are the total inlet pressure and temperature, respectively, and can be evaluated for M and z with a standard atmosphere model.



Fig. D.7 Dimensionless parameters calculated from engine performance charts

The referred parameters for X_{SFC} and X_P , written X_{SFC_R} and X_{P_R} , are defined in Equations D.17 and D.18.

$$\delta = \frac{P_t}{101,325 \text{ Pa}} \tag{D.15}$$

$$\theta = \frac{T_t}{288.15 \text{ K}} \tag{D.16}$$

$$X_{SFC_R}(z, M) = \frac{X_{SFC}(z, M)}{\delta\sqrt{\theta}}$$
(D.17)

$$X_{P_R}(z,M) = X_P(z,M)\theta \tag{D.18}$$

Figure D.8 plots the referred parameters for the three reference engines. A regression formula was built according to these points and is given in Table D.2. Two additional response surfaces, provided in the same table, were generated for P_{MCL} and SFC_{MCL} as a function of engine size (i.e. P_{SLS}) and flight conditions (i.e. δ and θ). The accuracy levels of the response surfaces are satisfactory for preliminary design studies. The model made of the three regression formulas of Table D.2 then enables:

• to evaluate the power P_{MCL} that can be delivered by the turboshaft as a function of its size and flight conditions. In the thesis P_{MCL} is supposed to be the maximum power that can be delivered by the turboshaft (i.e. engine ratings with $T_{4_{design}} < T_4 \leq T_{4_{max}}$ are not used in the simulations).

- to evaluate the specific fuel consumption SFC_{MCL} at P_{MCL} as a function of turboshaft size and flight conditions.
- to evaluate the specific fuel consumption SFC at any power $P \leq P_{MCL}$ as a function of turboshaft size, flight conditions, and shaft power.

Note that SFC(z, M, P) was additionally limited to $4 \times SFC_{MCL}(z, M)$ to avoid unrealistic fuel consumption at very low power.

$\widetilde{X}_{appg} = 10^{A \times B}$	A	$\begin{array}{c} 10^{-2} \times [2.0, -27.5, 29.3, -71.5, -52.4, 331.2, \ldots \\ 380.4, -445.5, -910.1, -500.3, -91.8] \end{array}$
$ASFC_R = 10$	B^{T}	$\forall i \in \llbracket 0; 10 \rrbracket, B_i = \log^i X_{P_R}$
	error	$\begin{array}{rll} {\rm min:} \ -9.0\% & {\rm max:} \ 10.0\% \\ {\rm mean:} \ 0.0 \ \% & {\rm std:} \ 2.52\times 10^{-3} \end{array}$
~	A	$\begin{array}{c} 10^{-1} \times \left[\text{-}114.4, 44.0, \text{-}61.2, 318.5, \text{-}2.6, \dots \right. \\ \left. \text{-}34.4, \text{-}539.8, 8.4, \text{-}41.9, 255.9 \right] \end{array}$
$P_{MCL} = 10^{A \times B}$	B^\intercal	$ \begin{bmatrix} 1, \log P_{SLS}, \log \delta, \log \theta, \log^2 P_{SLS}, \log^2 \delta, \log^2 \theta, \dots \\ \log P_{SLS} \log \delta, \log P_{SLS} \log \theta, \log \delta \log \theta \end{bmatrix} $
	error	$\begin{array}{rll} {\rm min:} \ -0.8\% & {\rm max:} \ 1.0\% \\ {\rm mean:} \ 0.0 \ \% & {\rm std:} \ 736.85 \ {\rm W} \end{array}$
	A	$\begin{array}{c} 10^{-1} \times [52.9, -11.3, 50.2, -231.1, 0.7, \ldots \\ 42.9, 808.4, -5.2, 24.5, -365.6] \end{array}$
$SFC_{MCL} = 10^{-8} \times 10^{A \times B}$	B^\intercal	$ \begin{bmatrix} 1, \log P_{SLS}, \log \delta, \log \theta, \log^2 P_{SLS}, \log^2 \delta, \log^2 \theta, \dots \\ \log P_{SLS} \log \delta, \log P_{SLS} \log \theta, \log \delta \log \theta \end{bmatrix} $
	error	min: -1.0% max: 1.2% mean: 0.0 % std: $1.86 \times 10^{-11} \text{kg/s/W}$

Table D.2 Response surfaces in SI units for the specific fuel consumption model



Fig. D.8 Referred parameters calculated from engine performance charts

Appendix E

Propeller model

This appendix describes the models that have been developed and implemented for the prediction of propeller mass, maximum power capability and performance according to the propeller geometry and the propeller type (variable-pitch or high-lift). The following symbols are used:

- *D*: propeller diameter in m
- S: propeller disk area in m² (i.e. $\pi D^2/4$)
- x: number of blades
- β : blade pitch angle setting
- n: propeller rotation frequency in Hz
- V: aircraft air speed in m/s
- ρ : air density in kg/m³
- P: shaft power in W (P > 0: shaft-driven propeller; P < 0: propeller-driven shaft)
- T: force generated by propeller in N (T > 0: thrust; T < 0: drag)
- J: advance ratio (see performance model)
- C_p : power coefficient (see performance model)
- C_t : thrust coefficient (see performance model)
- η_p : propulsive efficiency defined by Equation E.1
- η_r : power recovering efficiency defined by Equation E.2

$$\eta_p = \begin{cases} \frac{TV}{P} & \text{if } P, T > 0\\ 0, & \text{otherwise} \end{cases}$$
(E.1)

$$\eta_r = \begin{cases} \frac{1}{\eta_p}, & \text{if } P, T < 0\\ 0, & \text{otherwise} \end{cases}$$
(E.2)

Variable-pitch propeller

Geometry

The propeller geometry is defined by the number of blades x and the diameter D. The propeller rotation speed is commonly limited by transonic flow at the blade tips: over a critical speed the propeller efficiency falls due to shock waves which sharply increase noise at the same time. By considering the same critical tip speed as that of ATR72 propeller (HS568F), the maximum propeller rpm is given by Equation E.3.

$$N_{max} = \frac{1200 \times 3.93}{D} \tag{E.3}$$

Also, based on the same maximum blade loading, the maximum power P_{max} that can be absorbed by the propeller is estimated from Equation E.4 (SI units).

$$P_{max} = \left(\frac{D}{3.93}\right)^2 \frac{x}{6} 2.25 \times 10^6 \tag{E.4}$$

Note that the reference values for the HS568F propeller are derived from its type certificate [33].

Mass

The propeller mass is supposed to be proportional to the number of blades and the disk area. With the HS568F propeller as reference [33], the propeller mas M is evaluated from Equation E.5 (Si units).

$$M = 180 \left(\frac{D}{3.93}\right)^2 \frac{x}{6}$$
(E.5)

Performance

This section describes the different models that have been developed and tested for the purpose of propeller performance estimation.

Performance Map

One of the most accurate and fast methods for the assessment of static (i.e. V = 0) and in-flight propeller performances consists in using performance maps of wind-tunnel tested propellers. These maps are plotted according to the dimensionless parameters J, C_p and C_t , that can be obtained from the Buckingham PI theorem [76]. These three parameters are defined by Equations E.6 to E.8. Also, the propulsive efficiency η_p can be rewritten such as in Equation E.9.

$$J = \frac{V}{nD} \tag{E.6}$$

$$C_p = \frac{P}{\rho n^3 D^5} \tag{E.7}$$

$$C_t = \frac{T}{\rho n^2 D^4} \tag{E.8}$$

$$\eta_p = \begin{cases} \frac{TV}{P} = J \frac{C_t}{C_p}, & \text{if } P, T > 0\\ 0, & \text{otherwise} \end{cases}$$
(E.9)

In the generalised performance calculation method of [125], a series of performance maps of variable-pitch propellers is provided. Each of them defines accurately the propeller performance for a particular set of shape parameters: number of blades, blade activity factor and blade integrated design lift coefficient. The blade activity factor characterizes the blade planform while the integrated design lift coefficient is representative of the aerofoil sections and their twist distribution law. The mathematical formulations of these parameters can be found in [125]. While these two parameters can play an important role in the detailed design of the propeller, they have been supposed constant in the preliminary design phase of this thesis. From [125] and based on propeller blade geometry of existing turboprop aircraft, the performance data of the 4-bladed propeller with 80 activity factor and 0.5 integrated design lift coefficient were selected as reference. The static and dynamic data are given in Figure E.1. One can notice that these data only deal with the propulsion case (i.e. $C_p > 0$).

Thanks to the use of the dimensionless parameters J, C_p and C_t , the performance maps of Figure E.1 are applicable to any propeller derived from the reference propeller under geometric similarity (i.e. iso-number of blades and homothetic transformation of all dimensions). Since current turboprop aircraft usually feature more than 6 blades to reduce blade loading and noise, the performance estimation method of this section has been extended to an x-bladed propeller. By assuming that each propeller blade contributes independently to the total shaft power absorption and thrust production, the reference data can still be used to evaluate the performance of the x-bladed propeller of similar diameter and blade shape parameters thanks to Equations E.10 and E.11. In the remainder of this first section, the subscript 4-blades has been intentionally skipped for simplicity, and all the equations refer to the 4-bladed reference propeller.

$$C_{p_{4-blades}} = C_{p_{x-blades}} \frac{4}{x} \tag{E.10}$$

$$C_{t_{x-blades}} = C_{t_{4-blades}} \frac{x}{4} \tag{E.11}$$

Let us recall that with regard to the selected orientation of the propulsion system modelling in the thesis, geometrical data, free stream conditions, propeller speed of rotation and shaft power are the propeller model inputs while the generated force is the output. However, as can be seen from Figure E.1b, the low C_p region and the small J region (i.e. 0 < J < 0.2) are not provided because of the difficulty to measure accurately propeller performance in these conditions. Since the use of a Newton-Raphson solver during aircraft performance evaluation in XMDO requires the continuity of the propeller model even outside classical operating domains, these regions must be filled properly.

The performance continuity relatively to the advance ratio in the small J region has simply been enforced by interpolating performance data at iso C_p between J = 0(i.e. static) and J = 0.2 (i.e. dynamic).

In the low C_p region, it should be pointed out that for a given J there actually exists a particular value of $C_p > 0$ for which $C_t = 0$. In this propeller operating mode, called transparency mode, the propeller produces neither thrust nor drag and the power absorbed by the propeller is equal to the power dissipated by friction and pressure forces. Since the knowledge of this limit is of great importance to optimise the propeller operation in descent for example, it has been estimated by plotting the C_t coefficients associated to the reference dynamic performance map. As shown in Figure E.2, each β -line (i.e. solid line) was extrapolated to define the advance ratio J_0 for which $C_t = 0$. From the different sets of β and J_0 , each β -line of Figure E.1b was also extrapolated until it reaches J_0 , which provided the associated $C_{p_{T=0}}$ along the vertical axis. The regression formula based on these points shows mean, minimum and maximum errors of 0.0 %, -3.1 %, and 1.9%, respectively, and is provided in Equation E.12.

$$C_{p_{T-0}} = (146.8 + 13.9J^{3.2}) \times 10^{-4}$$
(E.12)

Under given flight conditions, one may want to operate the propeller in a transparency mode in the most efficient way, that is to say by driving the shaft with minimum power. To do so, the propeller rotation frequency must be optimised in order to minimise the quantity $C_{p_{T=0}}n^3$. Solving this problem numerically yields to the advance ratio $J_{opt_{T=0}} \approx 4.90$. The associated rotation frequency can then be calculated thanks to the definition of J (Equation E.6). But depending on the flight conditions, the use of $J_{opt_{T=0}}$ value can lead to blade tip speeds higher than the maximum allowable value. According to Equation E.3, one can rewrite $nD \leq 80$ Hzm. The minimum power $P_{min_{T=0}}$ required for driving a propeller in transparency mode can then be estimated with Equation E.13.

$$P_{min_{T=0}} = \rho \left(\frac{V}{J}\right)^3 D^2 C_{p_{T=0}}(J), \quad \text{with } J = \begin{cases} J_{opt_{T=0}} & \text{if } V/J_{opt_{T=0}} \le 80\\ V/80 & \text{otherwise} \end{cases}$$
(E.13)

In order to avoid interpolation issues, two response surfaces based on non-extrapolated data of the reference static and dynamic performance maps were primarily generated. These response surfaces were built in log scale and their general expression is given by Equation E.14, where A and X are row and column vectors. Table E.1 provides A and X, as well as the regression errors. The accuracy levels of the response surfaces are highly satisfactory for such preliminary design studies, not to mention that minimum and maximum errors are located on a minimum number of data as shown in Figure E.3, which may also come from by the manual digitisation of the reference data.

$$C_t = 10^{A \times X} \tag{E.14}$$

For a given propeller (i.e. D) and given flight conditions (i.e. J, ρ and V), the Newton-Raphson solver of XMDO would vary the propeller shaft power, and hence C_p , to match a particular thrust force, and so a particular C_{treq} , imposed by the aircraft equations of motion. As can be seen from Figure E.4 which shows the evolution of C_t with C_p for different advance ratio J, there may exist no solution if $C_{treq} > C_{tmax}$ or even two solutions if $C_{treq} < C_{tmax}$, making the use of the response surface incompatible with the Newton-Raphson solver of XMDO. To cope with the existence and uniqueness of the solution, several transformations of the surface response have been tested. Since the decreasing part of the surface response is associated to worst propeller efficiencies than the increasing one, it has been replaced by a increasing quadratic surface with a C^1 connexion at C_{tmax} . The proper sizing of the propeller could then be checked

	A	$10^{-2} \times [44.4, 569.5, 1154.2, 1150.4, 550.9, 104.6]$					
Static	X^\intercal	$\forall i \in \llbracket 0; 5 \rrbracket, \ X_i = \log^i C_p$					
	error	min: -0.7 % max: 1.3% mean: 0.0% std ^{<i>a</i>} : 0.10×10^{-3}					
	A	$\begin{array}{c} 10^{-2} \times \left[-54.0, 29.9, 2.6, -60.0, 118.3, -59.3, \ldots \right. \\ \left44.5, 31.2, 3.9, -17.7, 2.8, -10.9, 4.2, -3.4, -6.4\right] \end{array}$					
Dynamic	X^\intercal	$ \begin{bmatrix} 1, \log J, \log C_p, \log^2 J, \log J \log C_p, \log^2 C_p, \dots \\ \log^3 J, \log^2 J \log C_p, \log J \log^2 C_p, \log^3 C_p, \log^4 J, \dots \\ \log^3 J \log C_p, \log^2 J \log^2 C_p, \log J \log^3 C_p, \log^4 C_p \end{bmatrix} $					
	error	$\begin{array}{rll} {\rm min:} \ -3.0\% & {\rm max:} \ 5.2\% \\ {\rm mean:} \ 0.0 \ \% & {\rm std:} \ 4.47\times 10^{-3} \end{array}$					

Table E.1 Response surfaces for the reference static and dynamic C_t coefficient

 a Standard deviation

with the constraint $C_{t_{req}}(t) < C_{t_{max}}(t)$ for all time t of the mission. However, such transformation still raised issues when the Newton-Raphson approached $C_{t_{max}}$, the point of maximum efficiency at a given J that shows a zero-derivative according to C_p . For these reasons, other simpler models have been considered in the thesis and are described in the next sections.

Actuator Disk

The well-known actuator disk theory, also referred to as Froudes' momentum theory, is probably the most widely used method to model propeller performance in conceptual design studies. More information about it can be found in [111] for example. The simple relations derived from this theory are shown in Equations E.15 and E.16 for the static and dynamic cases, respectively. Since the original method provides an estimate of the ideal operation of the propeller, it is usual to include friction losses and blade tip losses through a correction factor η_c .

$$T = (2\rho S P^2 \eta_c^2)^{1/3} \tag{E.15}$$

$$P = \frac{0.5VT}{\eta_c} \left[\left(\frac{T}{0.5\rho SV^2} + 1 \right)^{0.5} + 1 \right]$$
(E.16)

It is important to note that Equation E.16 works for both the propulsive mode (P > 0) and the wind turbine mode (P < 0) but is not able to represent the transparency

mode discussed in the previous section. In addition, since P must be the input of the propeller performance model in the thesis, the calculation of T in the dynamic case requires Equation E.16 to be solved, which can be done in a fast and reliable way with a gradient-based solver. Finally, if the propeller efficiency estimated from Equation E.16 in high speed phases is generally close to the one obtained from the performance map of a variable-pitch propeller, the use of the actuator disk theory during low speed phases tends to over-predict it, as shown in Figure E.5. In order to improve the accuracy of propeller performance estimation in low speed phases, a correction of Equation E.16 has been used in the model presented in the next section.

Selected model

This section focuses on an other model that has been developed during the thesis and that was finally selected to carry out the hybrid aircraft studies. This model combines the assets of the two methods exposed previously.

Nominal operation: n > 0

Let $T_{AD}(P)$ be the force returned by the evaluation of Equation E.15 or the resolution of Equation E.16 when $P \ge 0$. In this propulsive mode, T_{AD} is calculated with $\eta_c = 0.91$ in accordance with the maximum efficiency shown in Figure E.1. Also, let $D_{AD}(P)$ be the force found by solving Equation E.16 in wind turbine mode (i.e. P < 0). In this latter case, η_c is taken equal to 0.45 because a propeller cannot be as efficient to produce thrust as to recover power if it was primarily designed for propulsion. The correction applied to $T_{AD}(P)$ to improve the thrust estimation at low speed is given in Equation E.17, where $T_{M_s}(P)$ is the static thrust that can produce the propeller driven at maximum tip speed (i.e. nD = 80 Hzm) under the same air density, and calculated according to the response surface of the reference static performance map of Table E.1 rescaled with respect to the number of blades. This correction has been built with the objective to approach the thrust returned by the performance map method. The different models are compared in Figure E.5.

$$T_{AD_{mod}}(P) = T_{M_s}(P) \arctan\left(1.18 \frac{T_{AD}(P)}{T_{M_s}(P)} + 1.40 \left[\frac{T_{AD}(P)}{T_{M_s}(P)}\right]^2\right) (753.96 - 0.52V) 10^{-3}$$
(E.17)

In order to include the transparency operating mode of the propeller in the model, the minimum power $P_{min_{T=0}}$ required to drive the propeller in such mode is estimated from Equation E.13 that is corrected according to the number of blades. Then, because the generated force is zero for $P = P_{min_{T=0}}$, the propeller creates drag when P = 0. With the lack of data from performance maps at such operating point, the generated force $T_{P=0}$ at zero shaft power is taken equal to $D_{AD}(-P_{min_{T=0}})$, the drag generated in wind turbine mode from the actuator disk theory when producing a shaft power equals to $P_{min_{T=0}}$.

Finally, the transitions between the different operating modes of the propeller have been implemented such as in Equation E.18.

$$T(P) = \begin{cases} T_{P=0} + D_{AD}(P) & \text{if } P < 0\\ T_{P=0}(1 - P/P_{min_{T=0}}) & \text{if } 0 \le P/P_{min_{T=0}} < 1\\ T_{AD_{mod}} \left(P - \left[\frac{1.3 - P/P_{min_{T=0}}}{0.3} \right]^3 P_{min_{T=0}} \right) & \text{if } 1 \le P/P_{min_{T=0}} < 1.3\\ T_{AD_{mod}}(P) & \text{otherwise} \end{cases}$$
(E.18)

As shown in Figure E.6 this model forms a monotonically increasing function of T with P under given flight conditions, making it suitable with the Newton-Raphson solver of XMDO during aircraft performance evaluation. In addition, it includes both the propulsive mode (P > 0) and the wind turbine mode (P < 0), the latter being useful for the study of energy recovering potentials.

<u>Failure: n=0</u>

When a variable-pitch propeller cannot be driven any more because of the failure of a gas turbine for example, the propeller blades can be rotated around their blade pitch axis to reach the feathering position and reduce the propeller drag. In this position, the chord line of the blade is approximately parallel to the on-coming airflow. The force generated by the feathered propeller is estimated thanks to Equation E.19, that is derived from [15].

$$T = -\frac{1}{2}\rho V^2 S \frac{x}{8} 0.02 \tag{E.19}$$



Fig. E.1 Performance data from [125] for the 4-bladed propeller, 80 blade activity factor and 0.5 blade integrated design lift coefficient



Fig. E.2 Thrust coefficients derived from Figure E.1b



Fig. E.3 Estimated dynamic C_t coefficients with response surface versus reference data



Fig. E.4 Illustration of response surface outputs for different advance ratio



Fig. E.5 Comparison of propeller performance models for a 6-bladed propeller of 3.93 m diameter rotating at maximum tip speed at sea level



Fig. E.6 Evolution of propeller force with shaft power for a 3.93 m diameter propeller at sea level; calculated from the selected propeller model

High-lift propeller

In this thesis, a high-lift propeller is as a fixed-pitch folding propeller.

Geometry and Mass

The geometry is still characterised by the propeller diameter and the number of blades. For simplicity, the same equations as for the variable-pitch propeller are used for the predictions of maximum rotation speed, maximum power absorption capability and mass. It should be noted that the mass is then probably overestimated since the folding mechanism is expected to be much simpler than the one controlling the blade pitch angle.

Performance

The typical operating curve of a fixed-pitch propeller can be seen as a β -line of Figure E.1. Because the selection of the correct propeller shape would require at list one additional design variable (i.e. β), the high-lift propeller performance has been simply estimated thanks to the modified actuator disk theory, as written in Equation E.20, which obviously over-predicts the high-lift propeller efficiency over a wide range of flight conditions. This has been taken into account in the analyses of the optimisation results for the hybrid aircraft configuration featuring such propellers. When the propeller is not driven, the drag created by the propeller blades folded along the nacelle is supposed to be sufficiently small compared to the total aircraft drag, and is neglected. The force generated by the folding propeller when P = 0 (i.e. nominal or failure case) is thus zero in this model.

$$T(P) = \begin{cases} T_{AD_{mod}}(P) & \text{if } P \ge 0\\ 0 & \text{otherwise} \end{cases}$$
(E.20)

Appendix F

Gearbox model

This appendix focuses on the different gear reducer models that have been developed for the prediction of torque capacity, mass and efficiency from the gearbox type and geometry. The following symbols are used:

- T_{in} : input torque of the gear assembly (N.m)
- T_{out} : output torque of the gear assembly (N.m)
- ω_{in} : input speed of the gear assembly (rad/s)
- ω_{out} : output speed of the gear assembly (rad/s)
- M_0 : overall gear ratio defined by Equation F.1
- d_i : pitch diameter of gear #i (m)
- b_i : face width of gear #i (m)
- ψ : pitch diameter to face width ratio of the smallest gear of a gear stage
- η : gearbox efficiency defined by Equation F.2
- K_H : constant which takes into account the type of application (smooth running or peak torques), the type of gears (spur, helical), the manufacturing accuracy level, the material properties, the reliability, and the safety margin

$$M_0 = \frac{T_{out}}{T_{in}} = \frac{\omega_{in}}{\omega_{out}} \tag{F.1}$$

$$\eta = \frac{T_{out}\omega_{out}}{T_{in}\omega_{in}} \tag{F.2}$$

Geometry and maximum torque

The external casing of the gearbox is modelled by a cylinder of diameter D_{gbx} and length L_{gbx} , as illustrated in Figure F.1. In order to account for the volume of bearings, shafts, pumps and other equipments for gearbox lubrication, all the gears are supposed to be

contained inside a cylinder of diameter D_g and length L_g . For all the gearboxes of this model, $D_g = 0.65D_{gbx}$ and $L_g = 0.3L_{gbx}$. In addition, two categories of gearbox which differ from each other by their serial arrangement of elementary gear sets (Figure F.2) are defined. They are called **turboprop** and **in-line**.

In order to reduce the number of inputs for the gearbox model, L_{gbx} is defined as a function D_{gbx} , M_0 and ψ . Hence, the model of this section must be able to return the maximum torque T_{inmax} that can be applied at the gearbox input shaft from D_{gbx} , M_0 , ψ and the gearbox category.



Fig. F.1 3D rendering of a PW100 engine [132]



Fig. F.2 Elementary gear sets of the gearbox model—modified from [142]

In the case of a single gear pair (i.e. offset gearbox of Figure F.2) operating as a reducer (i.e. the driving gear is the small one, also referred to as pinion), the maximum torque T_{inmax} that can be applied at the input is defined by Equation F.3, as discussed in [142] and [57], where gear #1 is the pinion¹. The different models of this appendix for the determination of the maximum torque capability are based on this equation.

¹The form factor ψ for the offset gearbox is b_1/d_1 (Figure F.2)

$$T_{in_{max}} = \psi d_1^3 \frac{M_0}{M_0 + 1} \frac{K_H}{2}$$
(F.3)

Turboprop

If different combinations of elementary gear sets can be used for the design of turboprop engine gearboxes, the double-reduction double-branch assembly (Figure F.2) is one of the simplest and most compact architecture providing high gear ratios. In addition, this assembly facilitates inherently the implementation of an offset between the input and output shafts, which enables to improve the efficiency of the air intake for the gas turbine. The double-reduction double-branch assembly is therefore chosen as the default architecture for the **turboprop** category.

The following model has been built according to the notations of Figure F.3. For simplicity, it has been assumed that the gear form factor of the two gear stages (i.e. b_1/d_1 and b_3/d_3) are identical. Nonetheless, for a given overall ratio M_0 and a given torque capability, there still exist two other variables that can change the overall dimensions of the gear system: m_i , the gear ratio for the contact between gear #1 and gear #2, and the angle α defined in Figure F.3. In order to facilitate the selection of these two parameters, they have been optimised with a reference double-reduction double-branch system of gear ratio M_0 for which ψ , K_H , and the maximum input torque $T_{in_{max}}$ are all set to unity. The diameter and length of the cylinder containing this reference gear system are called $D_{g_{ref}}$ and $L_{g_{ref}}$. After adapting Equation F.3 to the double-reduction double-branch case, M_0 was varied from 5.0 to 25.0, and m_i and α were numerically found by an optimiser minimising the volume $w \times h \times b_t$ defined in Figure F.3. The optimisation outputs are depicted in Figure F.4. Since h is always longer than w, the regression formulas written in Figure F.4 bare used for the evaluation of $D_{g_{ref}}$ and $L_{g_{ref}}$, as rewritten in Equations F.4 and F.5.

$$D_{g_{ref}} = 2.2786 M_0^{0.5255} \tag{F.4}$$

$$L_{g_{ref}} = -0.0004M_0^2 + 0.0278M_0 + 2.3517$$
 (F.5)

Based on the reference gearbox, the length of the gearbox under study can be written such as in Equation F.6.

$$L_{gbx} = L_{g_{ref}} \frac{0.65 D_{gbx}}{D_{g_{ref}}} \frac{\psi}{0.3}$$
(F.6)



Fig. F.3 Notations for the double-reduction double-branch gearbox model

The maximum input torque that can be applied to the double-reduction doublebranch gearbox of external diameter D_{gbx} and overall ratio M_0 is therefore given by Equation F.7 if L_{qbx} follows Equation F.6.

$$T_{in_{max}} = \left(\frac{D_g}{D_{g_{ref}}}\right)^3 \psi K_H \tag{F.7}$$

In-line

The planetary gearbox features the highest torque density [142] of all the elementary gear modules of Figure F.2. But because the range of gear ratio that can be covered by a single planetary gear is not wide enough, the **in-line** category was used to model one epicyclic gearbox or two epicyclic gears in series. The serial combinations that were selected as a function of M_0 and their associated models are presented next.

Simple offset: $1 < M_0 < 1.75$

Because the design of an epicyclic gear train with an overall gear ratio below 1.75 would result in very small planet gears compared to the sun gear, a simple offset gearbox was considered in this case.

For this gearbox architecture, the gearbox length is given by Equation F.8.

$$L_{gbx} = \frac{0.65 D_{gbx}}{0.3} \psi$$
 (F.8)

By assuming that $D_g = d_1 + d_2$, d_1 can be evaluated thanks to Equation F.9.



Fig. F.4 Optimisation outputs for the reference double-reduction double-branch gear system

$$d_1 = \frac{D_g}{1 + M_0} \tag{F.9}$$

Hence, based on Equation F.3, the maximum input torque that can be applied to the simple offset gearbox of external diameter D_{gbx} and overall ratio M_0 is given by Equation F.10.

$$T_{in_{max}} = \psi \left(\frac{D_g}{M_0 + 1}\right)^3 \frac{M_0}{M_0 + 1} \frac{K_H}{2}$$
(F.10)

Single epicyclic star: $1.75 \le M_0 < 2.75$

Here again, to favour the balance in gear diameters, the epicyclic gearbox is supposed to operate as a star system for $M_0 < 2.75$. In such configuration, the relation between m_i and M_0 shown in Equation F.11 holds. Also, with regards to the definition of ψ , and by assuming that the ring pitch diameter is equal to D_g , Equations F.12 and F.13 can be written for the sun gear diameter d_1 and the gearbox length, respectively.

$$m_i = \frac{M_0 - 1}{2} \tag{F.11}$$
$$d_1 = \frac{D_g}{M_0} \tag{F.12}$$

$$L_{gbx} = \frac{0.65D_{gbx}}{0.3} \frac{\psi}{M_0} \min\left(1, \frac{M_0 - 1}{2}\right)$$
(F.13)

The number of planet gears n_p of the star system is evaluated from Equation F.14 expressing the maximum number of planet gears that can fit in the assembly without interferences according to pitch diameters. In this equation, int(x) returns the integer part of x.

$$n_p = \operatorname{int}\left(\frac{\pi}{\operatorname{arcsin}([M0-1]/[M0+1])}\right)$$
(F.14)

Finally, the maximum input torque that can be applied to the single star gearbox of external diameter D_{gbx} and overall ratio M_0 is given by Equation F.15, where the factor $1+0.25\sqrt{n_p-3}$ accounts for planet load sharing and is defined as mesh load factor by ISO Standards [57].

$$T_{in_{max}} = \psi \min\left(1, \frac{M_0 - 1}{2}\right) \left[\frac{D_g}{M_0}\right]^3 \frac{M_0 - 1}{M_0 + 1} \frac{K_H}{2} \frac{1}{1 + 0.25\sqrt{n_p - 3}} n_p \tag{F.15}$$

Single epicyclic planetary: $2.75 \le M_0 < 2.75^2$

For the same gear dimensions, operating an epicyclic system as a planetary results in an overall gear ratio equals to $M_{0,s} + 1$, where $M_{0,s}$ is the gear ratio of the system in star configuration mode. Consequently, the planetary system provides higher torque densities than the star gearbox and the sizing equations can be easily derived from the previous case. The planetary architecture is considered for an overall gear ratio up to 2.75^2 since the difference in size of the sun and planet gears becomes important for that gear ratio, and the combination of two planetary systems in series provides a smaller external diameter. The formulas for the planetary gearbox are provided in Equations F.16, F.17 and F.18.

$$L_{gbx} = \frac{0.65D_{gbx}}{0.3} \frac{\psi}{M_0 - 1} \min\left(1, \frac{M_0 - 2}{2}\right)$$
(F.16)

$$n_p = \operatorname{int}\left(\frac{\pi}{\operatorname{arcsin}([M0-2]/M0)}\right) \tag{F.17}$$

$$T_{in_{max}} = \psi \min\left(1, \frac{M_0 - 2}{2}\right) \left[\frac{D_g}{M_0 - 1}\right]^3 \frac{M_0 - 2}{M_0} \frac{K_H}{2} \frac{1}{1 + 0.25\sqrt{n_p - 3}} n_p \tag{F.18}$$

Double epicyclic planetary: $7.565 \le M_0 < 35$

The double epicyclic planetary gearbox is made of two planetary gears in series. Based on Figure F.3, the length L_q is defined as the sum of b_1 and b_2 while the diameter D_q is defined as the maximum of h_1 and h_2 , that are respectively defined as the ring pitch diameters of the input and output planetary reducers. For simplicity, it has been assumed that the two planetary gears feature the same gear form factor ψ . For a given overall ratio M_0 and a given torque capability, the choice of the gear ratio of the first planetary reducer m_i can also change the overall dimensions of the gear system. Similarly to the turboprop category, this degree of freedom has been optimised with a reference double epicyclic planetary gear system for which ψ , K_H , and the maximum input torque $T_{in_{max}}$ are equal to one. The diameter and length of the cylinder containing this reference gear system are called $D_{g_{ref}}$ and $L_{g_{ref}}$. After adapting Equation F.18 to the double epicyclic planetary case, M_0 was varied from 7.565 to 35, and m_i was numerically found by an optimiser minimising the volume $\max^2(h_1, h_2) \times [b_1 + b_2]$ (Figure F.5). Figure F.6 shows the optimisation outputs and the regression formulas are used for the evaluation of $D_{g_{ref}}$ and $L_{g_{ref}}$, as rewritten in Equations F.19 and F.20.



Fig. F.5 Notations for the double planetary gearbox model

$$D_{g_{ref}} = 0.1146M_0 + 2.6743 \tag{F.19}$$

$$L_{g_{ref}} = 1e - 6M_0^5 - 0.0002M_0^4 + 0.0069M_0^3 - 0.1438M_0^2 + 1.4681M_0 - 4.0955$$
 (F.20)



Fig. F.6 Optimisation outputs for the reference double-reduction double-branch gear system

Based on the reference gearbox, the length of the gearbox under study is calculated from Equation F.21.

$$L_{gbx} = L_{g_{ref}} \frac{0.65 D_{gbx}}{D_{g_{ref}}} \frac{\psi}{0.3}$$
(F.21)

Then, the maximum input torque that can be applied to the double-planetary gearbox of external diameter D_{gbx} and overall ratio M_0 is given by Equation F.22.

$$T_{in_{max}} = \left(\frac{D_g}{D_{g_{ref}}}\right)^3 \psi K_H \tag{F.22}$$

Mass

Because a gearbox component includes gears but also bearings, shafts, casings and lubrication systems, the total gearbox mass can hardly be estimated from the gear dimensions only. In this model, the semi-empirical formula from [10] and [53], rewritten in Equation F.23 in SI units, is used whatever the gearbox type. Since the implementation of speed multipliers does not really make sense in the context of hybrid-electric aircraft propulsion, the mass of the gearbox component is set to zero if $M_0 \leq 1$.

$$M = \begin{cases} -16.7829 + 52.6167 \left(\frac{T_{out}\pi}{22380}\right)^{0.75} M_0^{0.15} & \text{if } M_0 > 1\\ 0 & \text{otherwise} \end{cases}$$
(F.23)

Efficiency

In this model the gearbox efficiency is assumed to be independent of the power flowing through the component and a constant value is assigned to η as a function of the gearbox category. The selected values are shown in Table F.1.

Category	M_0	Architecture	$\eta~(\%)$
Turboprop	$\begin{array}{l} 1 < M_0 \\ M_0 \le 1 \end{array}$	Double-reduction double-branch Direct drive	
In-line	$7.565 \le M_0$ $2.75 \le M_0 < 7.565$ $1.75 \le M_0 < 2.75$ $1 < M_0 < 1.75$ $M_0 \le 1$	Double planetary Single planetary Single star Simple offset Direct drive	98 99 99 99 100

Table F.1 Gearbox efficiency

Appendix G

Permanent magnet synchronous machine model

This appendix describes the model of the Permanent Magnet Synchronous Machine (PMSM) that was developed. It is based on the work of Budinger et al. [23] providing estimation models for the preliminary design of electro-mechanical actuators. These models are made of scaling laws, also called similarity laws or allometric models, and have the advantage of requiring only one reference component for a complete estimation of a product range. The correct generation of scaling laws involves two main assumptions:

- Material similarity: all material and physical properties must be identical between the reference and the derivative components.
- Geometric similarity: the ratio of a derivative component length to the reference component length of the corresponding spatial dimension is constant.

In this appendix, the scaling ratio x^* denotes the ratio of the parameter under consideration x for the derivative component to the equivalent parameter x_{ref} of the reference component (i.e. $x^* = x/x_{ref}$).

The PMSM model provided in [23] assumes $l^* = d^*$, where l and d are the motor length and diameter, respectively. In other words, the aspect ratio l/d of a derivative motor is the same as that of the reference component. The model presented hereafter extends the PMSM scaling laws of [23] to the case $l^* \neq d^*$. The influence of the cooling system power is also addressed.

Geometry and mass

As described in [23] and illustrated in Figure G.1, scaling laws for a PMSM can be defined for two different type of machine geometry:

- **Cylindrical**: the machine features a constant number of poles and all the radial dimensions varies homothetically.
- Annular: the number of poles is increased proportionally to the machine diameter but the ring thickness *e* is constant. The radial section of the machine can be seen as an assembly of elementary blocks.



Fig. G.1 Axial view of a PMSM [23]

According to these definitions, the machine mass M is given by Equation G.1 for the cylindrical and annular cases.

cylindrical :
$$M^* = d^{*2}l^*$$
 annular : $M^* = d^*l^*$ (G.1)

Based on the previous definitions of machine geometry, cylindrical machines are often made of a small number of poles while annular machines usually features a high number of them, making their operational areas different as illustrated in Figure G.2. At equivalent rotor speed the electrical frequency of an annular machine is therefore higher than a cylindrical one, and so are iron losses. Hence, the mechanical limit tends to be reached prior to the thermal limit for the cylindrical machine and the other way round for the annular machine. The next sections deal with the machine power losses and the different operational boundaries in more details.



Fig. G.2 Typical operational boundaries of cylindrical (top) and annular (bottom) brushless DC motors [23]

Power losses

As detailed in [23], the total power loss P_L of an electric machine can be defined as the sum of the Joule (copper) losses and the iron losses, and expressed such as in Equation G.2, where T is the electromagnetic torque, ω is the rotational speed, and α and β are the Joule and iron loss coefficients, respectively. Mechanical losses such as bearing friction losses and windage losses (air movement in the motor) also occur but are usually negligible. Note that Equation G.2 does not hold any more in case of iron saturation.

$$P_L = \alpha T^2 + \beta \omega^{1.5} \tag{G.2}$$

The electromagnetic torque of the machine can be deduced from the Laplace force expressed in Equation G.3, where n_B is the number of elementary blocks (Figure G.1), r is the distance from the rotor axis, J is the the current density and B is the magnetic field in the infinitesimal volume dV.

$$T = n_B \int_{1 \text{block}} JBr \mathrm{d}V \tag{G.3}$$

Using the scaling ratio notation, $n_B^* = 1$ and $V^* = d^{*2}l^*$ for the cylindrical machine while $n_B^* = d^*$ and $V^* = l^*$ for the annular. Hence, Equation G.4 can be written for the electromagnetic torque.

cylindrical :
$$T^* = J^* B^* d^{*3} l^*$$
 annular : $T^* = J^* B^* d^{*2} l^*$ (G.4)

For an electric machine, the magnetic flux density in the common iron sheet must not exceed given values in order to avoid saturation and associated losses. A range of electromagnetic components then must be sized with a constant maximal flux density, that is with $B^* = 1$.

Joule losses can also be expressed thanks to Ohm's law, recalled by Equation G.5, where R is the winding resistance, I the current, and ρ the electrical resistivity.

$$P_J = RI^2 = n_B \int_{1 \text{block}} \rho J^2 \mathrm{d}V \tag{G.5}$$

Since the previous expressions for n_B^* and dV^* still hold, and because material similarity involves $\rho^* = 1$, Equation G.6 expresses Joule power losses for the cylindrical and annular machines.

cylindrical :
$$P_J^* = J^{*2} d^{*2} l^*$$
 annular : $P_J^* = J^{*2} d^* l^*$ (G.6)

Combining Equations G.2, G.4 and G.6 yields to the expressions of the Joule loss coefficient shown in Equation G.7.

cylindrical :
$$\alpha^* = d^{*-4}l^{*-1}$$
 annular : $\alpha^* = d^{*-3}l^{*-1}$ (G.7)

As discussed in [23], iron losses can also be written as in Equation G.8, where f is the electrical frequency and M_I the iron mass.

$$P_I^* = f^{*1.5} M_I^* \tag{G.8}$$

Since the iron mass is proportional to the machine mass, and $f^* = \omega^*$ for the cylindrical machine (i.e. constant number of poles) but $f^* = \omega^* d^*$ for the the annular machine, combining Equations G.8, G.1 and G.2 yields to the expressions of the iron loss coefficient shown in Equation G.9.

cylindrical :
$$\beta^* = d^{*2}l^*$$
 annular : $\beta^* = d^{*2.5}l^*$ (G.9)

Thermal limit

Along the thermal limit (Figure G.2), P_L is equal to the maximum heat P_C that the cooling system is able to extract under continuous operation. Hence, the machine torque $T_{em,nom}$ at zero speed and the maximum continuous speed $\omega_{cont,max}$ at zero torque are given by Equations G.10 and G.11, respectively.

$$T_{em,nom} = \left(\frac{P_C}{\alpha}\right)^{0.5} \tag{G.10}$$

$$\omega_{cont,max} = \left(\frac{P_C}{\beta}\right)^{1/1.5} \tag{G.11}$$

Also, the mechanical power P_{M_C} along this limit is given by Equation G.12.

$$P_{M_C} = \omega T = \omega \sqrt{\frac{P_C - \beta \omega^{1.5}}{\alpha}} \tag{G.12}$$

The maximum efficiency of the machine is reached when P_{M_C} is maximum. The corresponding optimum speed $\omega_{C_{opt}}$, provided by Equation G.13 is obtained by finding the root of the derivative of P_{M_C} according to ω . The associated optimum torque $T_{C_{opt}}$ is given by Equation G.14. By multiplying $T_{C_{opt}}$ by $\omega_{C_{opt}}$ it can be seen that for a given machine (i.e. constant Joule and iron loss coefficients), the higher the power extraction of the cooling system, the higher the mechanical power P_{M_C} at the optimum point.

$$\omega_{C_{opt}} = \left(\frac{4}{7\beta}P_C\right)^{1/1.5} = \left(\frac{4}{7}\right)^{1/1.5} \omega_{cont,max} \simeq 0.69 \ \omega_{cont,max} \tag{G.13}$$

$$T_{C_{opt}} = \left(\frac{3}{7\alpha}P_C\right)^{0.5} = \left(\frac{3}{7}\right)^{0.5}T_{em,nom} \simeq 0.65 \ T_{em,nom}$$
(G.14)

For an electric motor, the machine efficiency along the thermal limit can be defined as $P_{M_C}/(P_{M_C} + P_C)$ and the resulting optimum efficiency $\eta_{C_{opt}}$ is written in Equation G.15. Thus, for a given machine, it can also be stated that the higher P_C , the higher the motor efficiency. However, let us recall that these equations only hold true while iron saturation does not occur.

$$\eta_{C_{opt}} = \left(1 + \left(\frac{3}{7\alpha}\right)^{-0.5} \left(\frac{4}{7\beta}\right)^{-1/1.5} P_C^{-1/6}\right)^{-1} \tag{G.15}$$

Some charts at the end of this appendix also show the influence of the machine size on power losses and machine efficiency.

As pointed out previously, achieving high power density requires a powerful and efficient cooling technology. The direct cooling of the windings thanks to liquid is probably the best candidate. Assuming that the machine of interest uses cooling channels in the stator, the heat exchange surface area S_{HE} evolves similarly with the motor dimensions for the cylindrical and annular machines. The relationship with scaling ratios is written in Equation G.16.

$$S_{HE}^* = d^* l^* \tag{G.16}$$

For a given cooling system, the maximum allowable temperatures of the machine components, and in particular the one of the winding insulation, actually set the operating thermal limit. Under material similarity, the maximum temperature θ_{max} of a machine component, obeys the scaling ratio formula $\theta^*_{max} = 1$. The power extraction P_C of the cooling system, can therefore be written such as in Equation G.17, under the assumptions that both the temperature and the mass flow rate of the cooling liquid are the same as for the reference machine.

$$P_C^* = S_{HE}^* = d^* l^* \tag{G.17}$$

But because changing the cooling liquid properties (e.g. thermal conductivity, temperature, mass flow rate) can lead to higher powe-to-weight ratio for the machine, the coefficient k_C is introduced in this model and defined by Equation G.18.

$$P_C^* = k_C^* d^* l^* \tag{G.18}$$

Combining Equations G.9, G.11 and G.18 then yields to the scaling law of Equation G.19 for the maximum continuous speed $\omega_{cont,max}$.

cylindrical :
$$\omega_{cont,max}^* = k_C^* {}^{1/1.5} d^{*-1/1.5}$$
 annular : $\omega_{cont,max}^* = k_C^* {}^{1/1.5} d^{*-1}$ (G.19)

Finally, the machine torque $T_{em,nom}$ at zero speed obeys the relationship of Equation G.20, obtained from the combination of Equations G.7, G.10 and G.18.

cylindrical:
$$T_{em,nom}^* = k_C^{*0.5} d^{*2.5} l^*$$
 annular: $T_{em,nom}^* = k_C^{*0.5} d^{*2} l^*$ (G.20)

Mechanical limit

The maximum rotational speed $\omega_{abs,max}$ of a PMSM can be limited by mechanical constraints induced by centrifugal forces or by axial or transverse vibrations. In the first case, the mechanical strength of the assembly prevent rotating components, such as permanent magnets, to separate from the rotor at high speed. Expressing the centrifugal mechanical constraint σ_C with scaling ratios yields to Equation G.21 for the cylindrical and annular machines.

cylindrical :
$$\sigma_C^* = d^{*2} \omega^{*2}$$
 annular : $\sigma_C^* = d^* \omega^{*2}$ (G.21)

Under material similarity, $\sigma^*_{C_{max}} = 1$, and the maximum rotational speed $\omega^*_{abs,max}$ follows the relations given by Equation G.22.

cylindrical:
$$\omega_{abs,max}^* = d^{*-1}$$
 annular: $\omega_{abs,max}^* = d^{*-0.5}$ (G.22)

Resonance frequencies can also be the limiting criteria for the maximum rotational speed, in particular when $l \gg d$. In the proposed model it is assumed that centrifugal constraints are more limiting than resonance frequencies.

Demagnetization or iron saturation

The maximum torque $T_{em,peak}$ of the transient domain of operation is limited by the demagnetization of magnets or the saturation of the magnetic circuit in particular in the stator teeth. In this model it is assumed that saturation prevails over demagnetization, which has a weak influence on the results of the thesis since only the continuous domain of operation has been used.

The magnetic saturation actually depends on both the magnetic field of the permanent magnet (supposed constant with $B^* = 1$) and the current-induced magnetic field of the stator windings. The latter can be characterised by the dimensionless number Π of Equation G.23 expressing the ratio of the current-induced magnetic field, derived from Ampère's circuital law, to the iron saturation field B_{sat} .

cylindrical :
$$\Pi = \frac{\mu_0 J d}{B_{sat}}$$
 annular : $\Pi = \frac{\mu_0 J}{B_{sat}}$ (G.23)

For a product range with equivalent maximum saturation levels (i.e. $B_{sat}^* = 1$), the saturation current density J_{sat} then follows Equation G.24.

cylindrical :
$$J_{sat}^* = d^{*-1}$$
 annular : $J_{sat}^* = 1$ (G.24)

Hence, combining Equations G.24 and G.4 yields to the expressions of Equation G.25 for $T_{em,peak}$ based on a saturation criterion, which can also be rewritten as a function of $T_{em,nom}$ by using Equation G.20, as shown in Equation G.26.

cylindrical :
$$T_{em,peak}^* = d^{*2}l^*$$
 annular : $T_{em,peak}^* = d^{*2}l^*$ (G.25)
cylindrical : $T_{em,peak}^* = k_C^{*-0.5}d^{*-0.5}T_{em,nom}^*$ annular : $T_{em,peak}^* = k_C^{*-1}T_{em,nom}^*$ (G.26)

Equation G.26 shows a risk of saturation in nominal operation with high cooling powers for both machines, and with large diameters for the cylindrical geometry assumption.

Reference motor

The Siemens SP260D motor, shown in Figure G.3, is used as the reference machine in the thesis. With a power-to-weight ratio of 5.2 kW/kg in continuous operation, this electric motor set a new world-record in 2016 by propelling the aerobatic plane EXTRA 330LE [122]. The main technical data of this liquid-cooled machine are listed in Table G.1. From this table, it can be stated that the total power loss at maximum continuous power is 13 kW.

Since Joule and iron losses are not provided, they have been evaluated in order to apply the estimation models presented before. According to the electrical steel specification sheets of [25], steel used in electrical machines generates at least 2 W/kg of iron losses under a magnetic field of 1.5 T at 50 Hz. Based on Equation G.8, iron losses can be estimated for the reference motor thanks to Equation G.27.

$$P_I = 2M_I \left(\frac{f}{50}\right)^{1.5} \tag{G.27}$$

If the SP260D features the same magnet distribution as in the Siemens patent [121], the number of poles is 18. At 2,500 rpm, the rotation speed associated to the continuous torque of Table G.1, the electrical frequency is therefore 750 Hz. By assuming that the iron mass accounts for approximately half of the total motor mass, iron losses are therefore estimated to be ~ 3-4 kW. According to Equation G.5, Joule losses can be estimated for the reference motor thanks to Equation G.28, where V_C is the copper volume.

$$P_J = V_C \rho J^2 \tag{G.28}$$

With a typical current density of 20 A/mm² for liquid-cooled machines, a copper resistivity of $1.8 \times 10^{-8} \Omega m$, a copper density of $8,900 \text{ kg/m}^3$, and by assuming that the SP260D windings are made of roughly 10 kg of copper, Joule losses are estimated to be ~ 6-10 kW. In the PMSM model of the thesis, the power losses of the reference motor under the continuous operating conditions of Table G.1 are assumed to be 9kW and 4kW for Joule and iron losses, respectively.

If most published data [122] [120] shows a maximum rotational speed of 2,500 rpm for the SP260D, this figure yields to a small peripheral speed of 50 m/s. Indeed, with today's rotor assembly technologies, the speed of the rotor surface can be even higher than 100 m/s for PMSM [47]. The 50 m/s peripheral speed can be explained by the fact that the SP260D directly drives the propeller which sets the maximum rotation speed. Based on a 100 m/s peripheral speed and the SP260D geometrical data, $\omega_{abs,max}$ is assumed to be around 5,000 rpm in this model.



Max. Continuous Power	260 kW
Torque @ 260 kW	$1{,}000~{\rm Nm}$
Efficiency $@$ 260 kW	95~%
Diameter	$416~\mathrm{mm}$
Length	$300 \mathrm{mm}$
Mass (with prop bearings)	52 kg

Table G.1 SP260D technical data [120]

Fig. G.3 CAD view of the SP260D [120]

General trends

The model of this appendix has first been applied to the case of a motor featuring the same dimensions (i.e. $d^* = 1$ and $l^* = 1$) and cooling power ($k_C = 1$) as the SP260D for the study of the machine performance along its thermal limit. As shown in Figure G.4a, operating the SP260PD at ω_{Copt} (3,750 rpm) instead of 2,500 rpm would enable to reach a power-to-weight ratio of 6.2 kW/kg. This difference with the power density published by Siemens may come from different reasons:

- the above estimation of power losses is wrong due to the strong assumptions made.
- the 1.5 power factor of Equation G.8 that comes from an approximation Budinger et al. [23] does not exactly fit to the real iron losses
- the machine design was constrained by a maximum allowable frequency in electrical steel with regards to the design methods used (e.g. validity of models, experimental measures, etc).
- the design conditions being imposed by the direct drive operation of the propeller, the resulted design optimises the motor performance at the design point which may not be the point of maximum power (i.e. design driven by other criteria and constraints).

Without more information, the previous data are still used for the reference machine of the PMSM model.

Figure G.4b plots the effect of the cooling power increase on the mechanical output power for this motor along the thermal limit. One can see that both the optimum operating speed $\omega_{C_{opt}}$ and the maximum output power increase with k_C . With a cooling power of 19.5 kW (i.e. $k_C = 1.5$), the power density of this motor operating under the same temperature as the reference motor (i.e. according to material and maximum allowable temperature similarities) reaches 9.9 kW/kg. However, let us recall that such output power level can only be achieved if the torque augmentation does not involve magnetic saturation and that the rotational speed stays below the mechanical limit $\omega_{abs,max}$. If the mechanical limit can be accounted for with Equation G.22, the evaluation of the saturation limit expressed by Equation G.26 requires $T_{em,peak}$ for the reference motor to be known, which is not the case. By considering the problem in the other way round, setting $T_{em,peak} \geq T_{em,nom}$ for the derivative motor with $k_C = 1.5$ requires $T_{em,peak}$ to be greater than $1.5T_{em,nom}$ for the reference motor according to Equation G.22, which is usually the case as shown in the motor specifications of [37]. However, the transient domain of the derivative motor would then be greatly reduced.



(a) Mechanical power and power losses with $k_C = 1$

(b) Influence of k_C on mechanical power

Fig. G.4 Application of the estimation models along the thermal limit with $d^{\ast}=1$ and $l^{\ast}=1$

An other set of calculations, depicted in Figure G.5, has also been used to compare the machine characteristics according to the cylindrical and annular geometry assumptions. The power density (Figure G.5a), the motor efficiency (Figure G.5b), and the optimum and maximum rotor speeds (Figure G.5d) are all independent of motor length since a machine of length 2l can be seen as a serial arrangement of two identical machines of length l with the proposed method. It is important to note that the maximum power density but also the maximum efficiency are independent of the motor diameter under the annular geometry assumption. This can be easily explained by the fact that power losses, heat exchange surface area and mass evolve similarly according to the repetition of the elementary blocks. On the contrary, the reason why the maximum power density and the maximum efficiency decrease with the motor diameter for the cylindrical geometry comes from the larger increase of mass and power losses with diameter than the heat exchange surface area. If electric machines are most often sized according to power density assumption in preliminary design studies of hybrid aircraft, it should be pointed out that such scaling law only holds for the annular geometry of this appendix. Finally, based on Figure G.5d, it can be seen that the limitation of the optimum operating speed by the centrifugal forces occurs for large (small, respectively) diameters under cylindrical (annular, respectively) geometry assumption.



(c) Evolution of machine mass with d and l

(d) Evolution of optimum speed $\omega_{C_{opt}}$ and maximum speed $\omega_{abs,max}$ with d

Fig. G.5 Evolution of the cylindrical and annular machine characteristics in the continuous domain of operation with d and l for $k_C = 1$