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# Study of application of electromechanical actuators to deploy control surfaces according to More Electrical Aircraft concept.

**UPC Final Degree Project**

***Report***

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# Abstract

Transportation represents a 20% of green-house effect gases emissions worldwide. There is a trend to move towards greener powered systems, such as electric ones. More Electric Aircraft (MEA) is a new concept into the aircraft industry which is trying to increase the usage of electrically powered systems in aeroplanes. This project reviews the available technologies for MEA in special regard to flight control surfaces actuators. Conventional hydraulic actuators in aircraft systems are low efficient and require high maintenance tasks and a heavy infrastructure. Powered-by-wire systems are being broadly applied into aeroplanes to improve the maintainability, reliability and also manoeuvrability of the next generation of aeroplanes. Throughout the project different fields involved in the effective development of new electromechanical actuators are inspected. First, the generation and power conversion of electric power into a plane. Second, the technologies concerning the building block of an electromechanical actuator in three main aspects: electric layout, mechanical layout and potential failures. Third, the monitoring devices and redundant architecture that would satisfy the continued airworthiness regulations. Finally, a mock-up for simulating and demonstrating purposes is designed and the most proper pattern nowadays for an electromechanical actuator for flight control surfaces is presented together with further work of MEA for the successful implementation of electromechanical actuators into the next generation of aircraft.

## Keywords

More electric aircraft, power converters, electromechanical actuators, monitoring systems, efficiency, power-by-wire

# Sinopsis

La industria del transporte es responsable en todo el mundo de un 20% de las emisiones de gases de efecto invernadero. Hay una tendencia importante de empezar a usar sistemas que estén alimentados de potencias provenientes de fuentes más ecológicas, como los sistemas eléctricos. MEA es un concepto innovador de la industria aeronáutica que intenta incrementar el número de sistemas alimentados por electricidad dentro de los aviones. Este proyecto lleva a cabo una revisión exhaustiva de las tecnologías que existen hoy en día para el MEA y especialmente aquellas relacionadas con los actuadores de las superficies de control. Los actuadores hidráulicos convencionales usados en los aviones son poco eficientes y requieren de un mantenimiento exhaustivo y una infraestructura pesada. Los nuevos aviones están empezando a instalar de forma generalizada sistemas alimentados a través de cable (power-bywire) que permiten mejorar de forma considerable el mantenimiento, la fiabilidad y también la maniobrabilidad. A lo largo del proyecto se inspeccionan los distintos campos que están involucrados en el diseño de actuadores electromecánicos. Primeramente, los sistemas de generación y conversión de potencia eléctrica de los aviones. En segundo lugar, las tecnologías asociadas al bloque constructivo del actuador electromecánico en tres aspectos principales: diseño eléctrico, diseño mecánico y potenciales fallos. En tercer lugar, se examinan los dispositivos de monitorización y las arquitecturas de sistemas redundantes que permitirán satisfacer las normativas de aeronavegabilidad continuada. Finalmente, se diseña una maqueta con fines de simulación y demostrativos y se presenta, junto con las tareas futuras de MEA para la instalación efectiva y exitosa de actuadores electromecánicos para superficies de control de vuelo en la futura generación de aviones, el esbozo del actuador electromecánico que hoy en día es el más válido para ser aplicado en las superficies de control de vuelo.

## Palabras clave

More electric aircraft, convertidores de potencia, actuadores electromecánicos, sistemas de monitorización, eficiencia, power-by-wire

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Title of the Thesis : *Study of application of electromechanical actuators to deploy control surfaces according to More Electrical Aircraft concept.*

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# Nomenclature

| | |
|---|---|
| $\alpha$ | Angle of attack $[rad]$ |
| $\alpha_{eff}$ | Effective angle of attack $[rad]$ |
| $\alpha_s$ | Stalling angle of attack $[rad]$ |
| $\beta$ | Sideslip angle $[rad]$ |
| $\delta$ | Control surface deflection $[°]$ |
| $\delta_t$ | Tab deflection $[°]$ |
| $\omega$ | Angular speed $rad/s$ |
| $\rho_0$ | Air density at Sea Level $[kg/m^3]$ |
| $\tau$ | Torque $[Nm]$ |
| $a$ | Slope of the aeroplane normal force coefficient $[N/rad]$ |
| $c$ | Mean aerodynamic chord $[m]$ |
| $C_H$ | Hinge-moment coefficient |
| $g_0$ | Gravity constant $[m/s^2]$ |
| $H$ | Hinge moment $[Nm]$ |
| $P$ | Power $[W]$ |
| $S$ | Planeform area $[m^2]$ |
| $V$ | Speed $[m/s]$ |
| $V_{s_0}$ | Stalling speed in landing configuration $[m/s]$ |
| $V_{s_1}$ | Stalling speed with wing flaps retracted $[m/s]$ |
| $V_s$ | Stalling speed $[m/s]$ |
| $w$ | Average wing loading $[N/m^2]$ |

# Acronyms

| | |
|---|---|
| **AEA** | All-Electric Aircraft |
| **AESA** | Agencia Estatal de Seguridad Aérea |
| **BLDC** | Brushless-Direct Current |
| **CP** | Certification Process |
| **CS** | Certificate Specifications |
| **CSD** | Constant Speed Drive |
| **DOA** | Design Organisation Approvals |
| **DOH** | Design Organisation Handbook |
| **EAS** | Equivalent Airspeed |
| **EASA** | European Union Aviation Safety Agency |
| **ECS** | Environmental Control System |
| **ECU** | Electronic Control Unit |
| **EHA** | Electro Hydraulical Actuator |
| **EMA** | Electro Mechanical Actuator |
| **EWIS** | Electrical Wiring Interconnection System |
| **FAA** | Federal Aviation Administration |
| **FAR** | Federal Administration Regulations |
| **HDO** | Head of Design Organisation |
| **HISM** | Head of Independent System Monitoring |
| **HMI** | Human-Machine Interface |
| **HOA** | Head of Office of Airworthiness |

**IDG** Constant Frequency Integrated Generator

**IPS** Ice-Protection System

**LDS** Lineal Displacement Sensor

**LVDT** Linear Variable Differential Transformer

**MEA** More Electric Aircraft

**MOC** Means of Compliance

**MOET** More Open Electrical Technologies

**MOP** Memoria de Organización de Producción

**PBW** Powered-by-Wire

**PCU** Power Control Unit

**PLC** Programmable Logic Controller

**PMSM** Permanent Magnet Synchronous Motor

**POA** Production Organisation Approvals

**PRSM** Planetary Roller Screw Mechanism

**RAMS** Reliability, Availability, Maintainability and Safety

**RDS** Rotary Displacement Sensor

**SR** Switched Reluctance

**TC** Type Certificate

**TVC** Thrust Vector Control

**VFD** Variable Frequency Drive

**VSCF** Variable Speed Constant Frequency

**VSVF** Variable Speed Variable Frequency

# Scope

The scope of this Thesis includes the following deliverables:

- Code to compute Hinge Moment derivatives
- Discussion about the regulations to be satisfied
- Description of future steps about MEA and electromechanical actuators
- Documentation to construct a demo case for the proposed solution
- Technical sheets of the systems developed
- Budget of the project

Furthermore, the scope statements are presented below:

- This project studies the applicability of electrical actuators for ailerons on aircraft. Other control surfaces actuators are not included.
- This project addresses aircraft flying on subsonic regime or below. Huge compressibility effects due to high Mach number will not be addressed.
- This project focus on the development of an initial electromechanical solution for ailerons actuators. No complex control systems will be studied and implemented.
- A sizing method will be defined: this sizing procedure for electromechanical actuators will use empirical data from bibliographical references. Computational fluid dynamics and wind-tunnel empirical analysis will not be carried out.
- An electromechanical actuator solution for control surfaces will be evaluated. However, not a single study will be carried out in order to quantify the positive effect of this solution in the reduction of fuel consumption. Furthermore, the reduction in maintenance costs will neither be estimated.
- The implementation of electric servomotors will be addressed. However, no mechanical study will be developed in order to increase the torque delivered by the servomotor.

# Chapter 1

# Introduction

It is a fact that climate change is a major problem that must be overcome today. Transport industry plays an important role when thinking of reducing the carbon footprint of human activities. Can still a change be possible? It is a must for transportation industry to redefine its basis in order to become more ecofriendly but, how can aeronautics get involved into this global renovation? More Electric Aircraft concept is going to be essential in the years to come.

*Chap. 1* is intended to summarize the motivation of this final degree project, to describe its aim an purpose, to present its background and to review the methodology to be followed.

## 1.1 Background

Climate change is one of the major issues worldwide nowadays [20]. The carbon footprint generated by human activities is causing severe damage to both biosphere and atmospheric phenomena. It is predicted that by 2050 the energy budget and emissions of greenhouse effect gases of the transportation industry will increase by 80% [51]. Global politics are being set to overcome this terrific tendency: it is a must to rethink the economic system, specially in regard to the energy sources used by transport [2].

Without a substantial cut in emissions from transportation none of the Global Politics, the aim of which is to reach a reduction in $CO_2$ emissions of at least 50% by 2050 [51], would not be achieved. Air transportation is widely used by everyone: it is cheap and gets people quickly from one place to another. However, transport sector represents a 20% of the total green-house effect gases emissions worldwide [56]. For this reason, the development and implementation of new technologies that provide energy-efficient solutions is extremely important [75].

Within transportation, the need to reduce the carbon footprint has led to a greater electrification of vehicles due to the inherently higher efficiencies of electrical systems [53, 75]. Aircraft industry has also been influenced by this tendency: many ambitious projects are being conducted towards More Electric Aircraft[1], redefining the energy sources used by planes [57]. MEA may be considered as a philosophy which is trying to evolve, step by step, into an all-electric aircraft world [14].

Hence, the objective of this project is to discuss and study the availability of electromechanical actuators' application on aircraft's control surfaces, with the aim to advance into MEA. Powered-by-Wire (PBW) systems improve the actuation performance of aircraft in respect to typical hydraulic power systems, as they transport the power in 'wires' between the devices instead of hitherto hydraulic pipelines, reducing the total weight of aircraft at the same time [53]. Although this project focus on flight control surfaces actuators, PBW systems technology extends the applications of electrically powered actuators for other systems such as landing gear, Thrust Vector Control (TVC) and engine actuation system [15, 36].

Some studies have been executed successfully –some of them are mentioned along *Chap. 3*– in the research of electromechanical solutions for control surfaces actuators. These new types of actuators are intended to increase the energy efficiency of the whole plane by reducing the required power during flight [63, 75]. Notwithstanding that many of the aforementioned solutions only consider secondary flight control surfaces actuators, could an electromechanical actuator be feasible for primary flight control surfaces?

## 1.2 Problem

MEA is going to be essential to face climate change for the years to come, as aforementioned along *Sec. 1.1*. The electrification of the industrial sector has led to an increase in factories performance and energy efficiency [44]. Moreover, transportation by traction vehicles is highly incentivized by the Administration to evolve towards increasingly electric technologies, which will help to drastically reduce the carbon footprint of day-to-day commuting [44, 51].

Nonetheless, most of aeroplanes systems are still powered by energy sources other than electricity [75]. The high demands of the aeronautical industry regarding reliability, redundancy of systems and effectiveness, as well as the tough certification processes of the aeronautical authorities –including European Union Aviation Safety Agency (EASA) and

[1]The definition of a MEA can be derived as an aircraft where the majority of the systems or a higher percentage of systems compared to conventional aircraft are electrically powered [63]

Federal Aviation Administration (FAA)–, give rise to this slowness in the evolution of electrical systems for aircraft. It is a matter of fact that there is a direct relationship between the accomplishment of airworthiness requirements and the time spent in the Certification Process of any aeronautical system or component [22]. In addition, the development of PBW systems for aeronautics is mainly financed by private equity [75]: Administration is still not giving enough resources for its advancement, although tough environmental politics have already been set up [51].

Therefore, how can the energy efficiency of aeroplanes be increased by rethinking the type of actuators? How can the energy generation and distribution into an aircraft be modified to increase the whole efficiency of the plane? What kind of solutions can be considered? How can those meet airworthiness requirements? May aeronautical Authorities redefine or update the aeronautical Regulations –Certificate Specifications (CS) for EASA and Parts for FAA [25]– to encourage the use of other power sources?

## 1.3 Purpose

The thesis discusses innovative solutions to redefine the power sinks and sources of air-transportation taking into account both, Airworthiness Directives and engineering feasibility. It illustrates how electric systems can be applied in aircraft power distribution and actuation systems as a means to increase the efficiency of air-transportation. Furthermore, the dissertation presents some of the most important points that may be considered to comply with airworthiness regulations when redefining the energy sources of an aeroplane.

## 1.4 Goal

The goal of this project is to discuss how electromechanical actuators can be implemented in airplanes control surfaces considering both, feasibility from an engineering point of view and compliance with current airworthiness regulations.

Technical documentation to build up a demonstration case for simulation purposes will be developed, as well as its associated technical sheets. Moreover, a preliminary budget to construct this system will also be presented. In the end, a consistent conclusion regarding whether is possible or not to start using these new systems for flight control surfaces will be exposed together with further work to be developed.

## 1.5 Benefits, Ethics and Sustainability

Sustainability is the primary objective of this degree project: the aim is to cut out aircraft transportation carbon footprint by increasing aircraft efficiency. All the transportation industry is indeed moving towards the use of electricity as the main power source. However, as the power density of jet fuel is markedly higher than the one offered by power electronics, being the gross weight of extreme importance when flying, the aircraft industry is focusing in enlarging the whole efficiency of aircraft systems [63].

This Thesis will benefit aeronautics industry by offering a way to reduce fixed maintenance costs [79] and increase the efficiency of the whole plane, thus reducing the amount of fuel needed for a flight [30]. As a result, MEA will benefit from new development towards achieving the objective of an All-Electric Aircraft (AEA). Furthermore, global environment will also benefit from these new technologies, so a reduction on aircraft fuel consumption will suppose an important decrease of transport's carbon footprint [56], as seen along *Sec. 1.1*.

Despite these evident technological benefits some ethical problems can arise, especially those related to the scientific development of humanity. Science has the capacity to reshape the planet and carry us into the cosmos, but is any of that worth the risks? Scientific development has not been the advantage of everyone, particularly for the Earth itself [11]. The question is: should we reduce aircraft transportation carbon footprint by improving its efficiency, or may us downscale globalization and use airplanes only for essential causes? It is sometimes better to take one step back to take two steps forward, as Winston Churchill ones said: *"It is arguable whether the human race have been gainers by the march of science beyond the steam engine...Give me the horse"*.

## 1.6 Methodology

*Fig. 1.6.1* describes the methodology that will be followed to execute the project and accomplish the aforementioned goals and scope statements.

At the beginning, a deep bibliographical research will be carried out with the aim to find useful

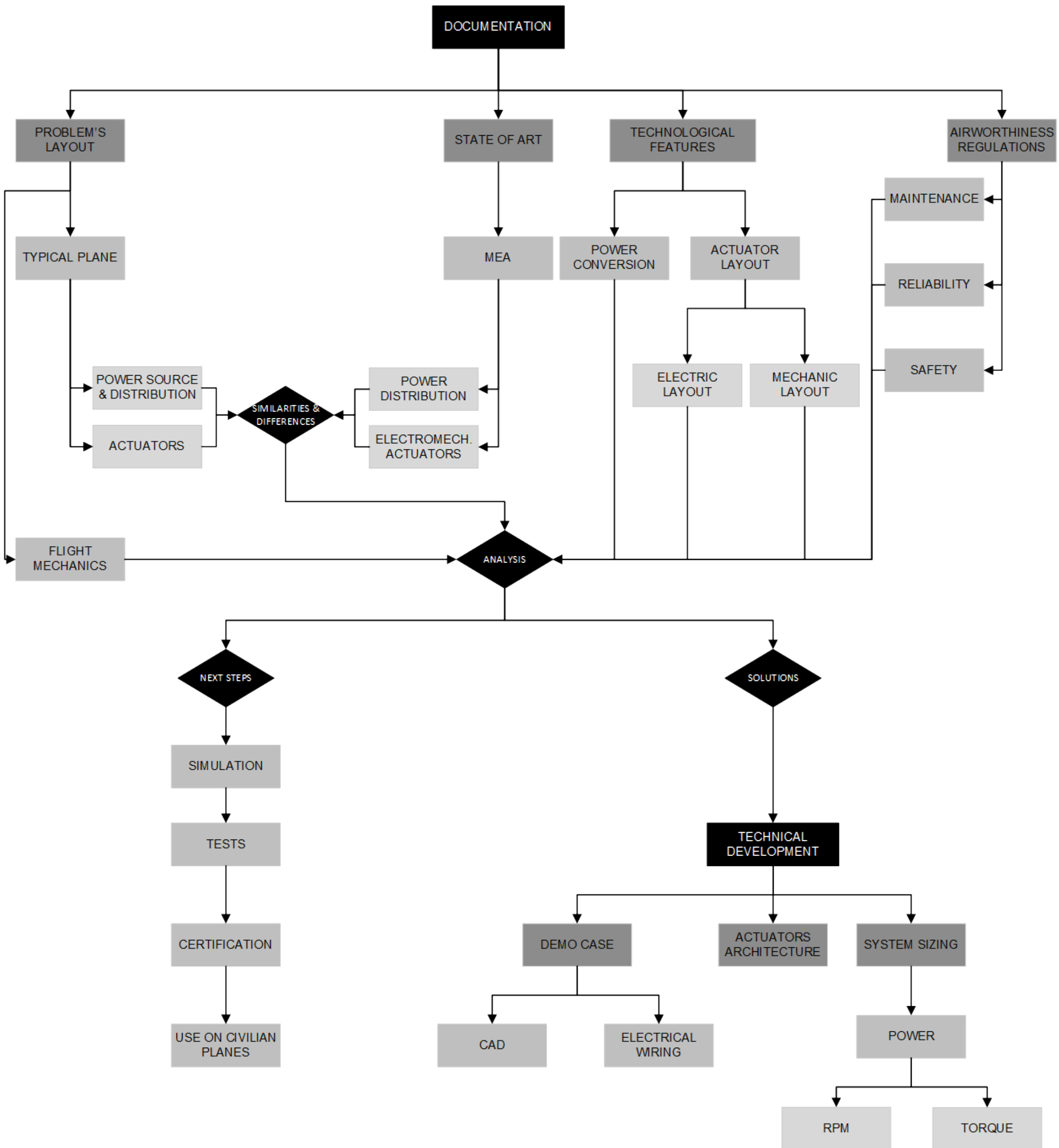



Figure 1.6.1: Schematic of the methodology to be followed during the project

information to later on decide deductively which solutions are the most suitable for the project. Airplanes systems and functionalities will be studied as well as technological features in the regard of electrical systems that are already employed into aircraft industry or that have just been implemented into an industrial environment. Airworthiness regulations and the state of art of noth MEA and EMA will play a fundamental role during the decision process.

After all this documentation, quantitative and qualitative analysis and decisions are going to be made with the aim to define next steps for MEA and electromechanical actuators for the aircraft industry. Then, the construction of a demonstration case will be hold with simulating purposes: this will ease the quantification of the performance of EMA. In the end, conclusions about the feasibility of EMA for airplanes control surfaces will be stated together with the next steps to be followed for their future implementation and approval.

## 1.7 Stakeholders

The stakeholders of the present degree project can be segregated into two different groups:

- *Internal stakeholders:*
  This group includes Control Techniques company; PDI, PAS and students of UPC and myself. People into this group will be directly benefited by the discoveries and research carried out along the project.

  Control Techniques company can use this information to expand its business into aeronautical industry, although a prior tough airworthiness certification process will have to be completed. UPC can adopt the project as a starting point for other research projects developed internally. Myself will acquire all this new knowledge and flourish the competences printed-out into UPC's curriculum.

- *External stakeholders:*
  External stakeholders include aeronautic industry, airline companies, Governments and environmental-organizations. This thesis could help to cut out on green-house effect gases emissions of air-transportation, and thus, to accomplish the currently established climate politics goals. Furthermore, a reduction on fuel consumption and on maintenance costs is also of interest of airline companies.

## 1.8 Delimitations

The development of this study can be affected by several factors that set the boundaries of it. One of the most important is the current situation caused by Covid-19's pandemic, which will force to raise again the practical development of this thesis. Besides, as the application of EMA for flight control surfaces is innovative for planes [57], limitations will arise from the finite and restricted technical information that is currently available [53].

## 1.9 Outline

Summarising, *Study of application of electromechanical actuators to deploy control surfaces according to More Electrical Aircraft concept* will inspect the existing and potential technologies to evolve into a More Electric Aircraft. The electrification of transportation will be of extreme importance to deal with global warming. Consequently, air-transportation will also need to redefine its power sources and hence, increase the proportion of electrically powered systems. This study will focus on the electrification of flight control surface actuators specially in regard to its maintainability and efficiency in order to conclude whether or not is a suitable and realistic technology for the next generation of aeroplanes.

# Chapter 2

# Typical Plane Systems

Along this chapter typical plane functions and systems are summarised with the aim to draw lines on the boundary conditions of the problem being studied all along this project. Furthermore, the flight envelope for the maximum hinge moments of the control surface is analyzed for an specific plane –Boeing 737-800 in this case[1]– together with the quantification of the actuation requirements of flight control surfaces. This information will be valuable later on to discuss and determine deductively an appropriate solution for electric actuated flight control surfaces.


Figure 2.0.1: Boeing 737-800 [38]

[1]The reason of this election is because it is one of the most used short and medium range commercial planes nowadays (4,989 deliveries [10]) together with Airbus A320 (5,696 deliveries [3]). In addition, Boeing 737-800 is the only available short and medium range commercial plane on the simulation software used along *Chap. 9*

## 2.1 Conventional aircraft power layout

In a conventional architecture of the power distribution system on a typical plane, as shown in *Fig. 2.1.1*, fuel is converted into power by the gas turbine engines: most of this power is used as propulsive power or thrust to move the aircraft, whereas the reminder is transformed into useful non-propulsive power for aircraft systems [57].

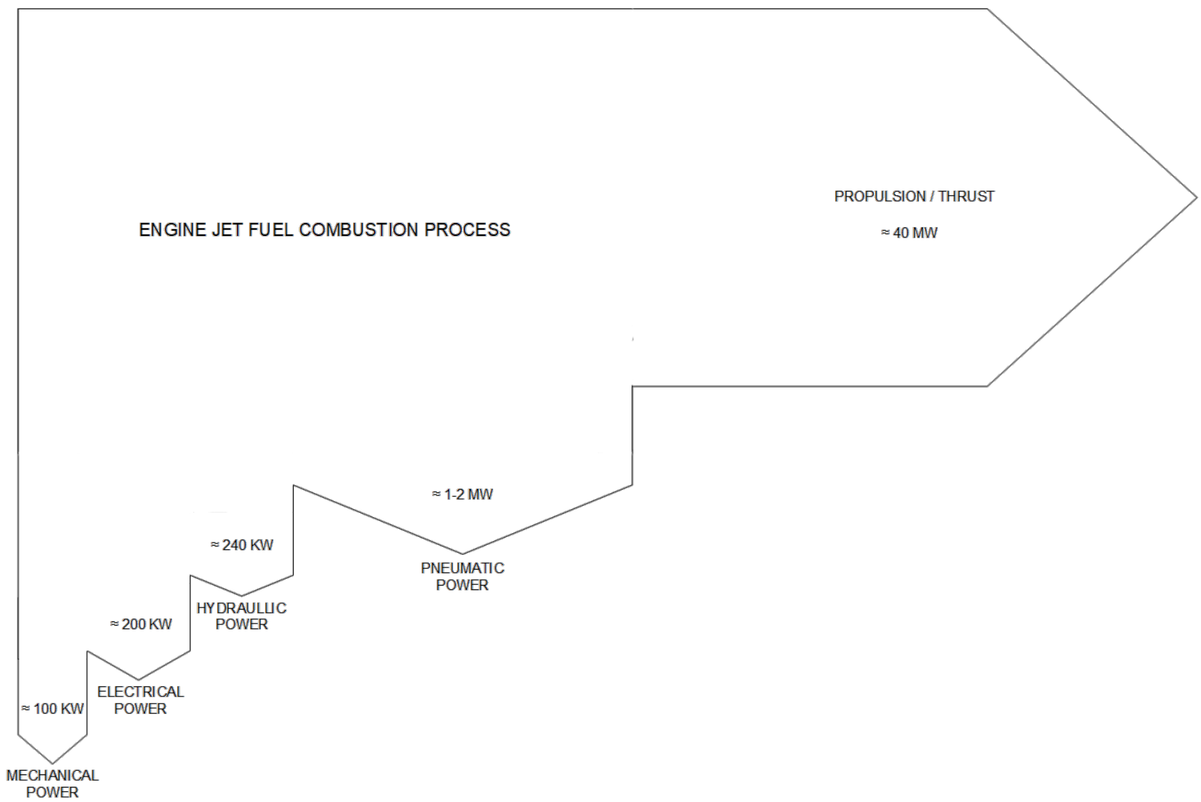



Figure 2.1.1: The power systems on a typical large civilian aircraft. Adapted from [78]

Conventional aircraft architectures used for medium or large aircraft include a combination of systems powered by different kind of energies [5]:

- *Electrical power* is obtained from the main generator as an AC power source (see *Sec. 2.3* for more information). It is later used to power avionics, cabin and aircraft lighting, galleys and other commercial loads. Electrical power is very flexible thus it does not require a heavy infrastructure. Nevertheless, its power density is conventionally lower than hydraulically powered systems and, additionally, airplane reliability can be reduced because of the higher risk of fire it generates–for instance in the case of a short circuit– [57].
- *Hydraulic power* is generated from central hydraulic pumps and then transferred to the

actuation systems, including landing gear deployment, retraction and braking; primary and secondary flight controls, among other systems (see *Sec. 2.2* for more information about the hydraulic system). In the one hand, hydraulic systems have a higher power density than electrical powered systems [52] and they are also very robust. In the other hand, hydraulic power systems need an inflexible and heavy infrastructure composed by pipelines [57], which increases the total weight of the systems. Regular maintenance tasks are also needed in order to reduce the potential leakage of dangerous and corrosive fluids [5].

- *Mechanical power* is transferred from engines to mechanically driven systems by means of mechanical gearboxes. This mechanically driven systems include, for instance, the main electrical generator and the central hydraulic pumps [5].

- *Pneumatic power* is obtained from the high-pressure compressors of the engines and is used to power Environmental Control System (ECS) and supply hot air for Ice-Protection System (IPS) systems [57]. Its main drawbacks are low efficiency and complication in detecting leaks.

Overall, the power distribution in a conventional aircraft, without considering power required for thrust or propulsion, is detailed in *Fig. 2.1.2*. As it can be seen, the main engine –turbojet or turbofan, depending on the aircraft– generates power that is later transformed into mechanical, hydraulic, electrical or pneumatic based on its specific application.

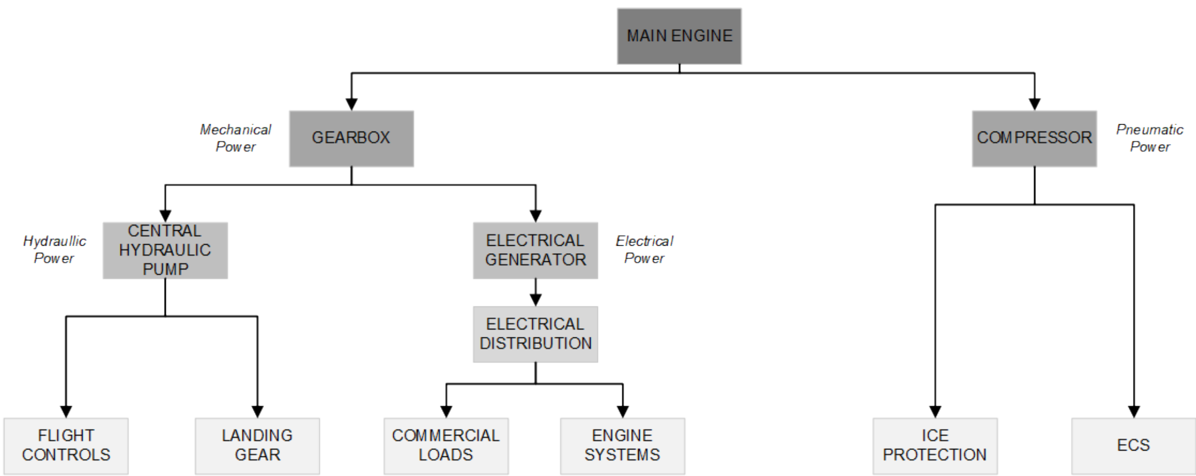



Figure 2.1.2: Schematic of conventional Power Distribution. Adapted from [57]

As these non-propulsive power systems have become more and more complex, the efficiency of the whole network –considering that they all interact with each other– has decreased. Furthermore, a simple leak of fluid in the pneumatic or hydraulic system may be difficult to locate and once located, it would not be easily accessed and repaired. This results in a grounded aircraft and undesirable flight delays [57].

## 2.2 Hydraulic system

Hydraulic power comprises one of the largest power consumption on conventional planes, as is stated in *Fig. 2.1.1*. It is generated by central hydraulic pumps, whose are normally engine-driven pumps that pressurize hydraulic fluid up to 20-35 [MPa] [14]. Since those engine-driven pumps are continuously operating, in order to get a high response rate where hydraulic actuators are used, the efficiency of the system is remarkably low [78]. In some cases –*Fig. 2.2.2* for instance– those central hydraulic pumps are electrically driven by the electrical energy generated with the aircraft generators.

*Fig. 2.2.1* presents how the hydraulic system is distributed into a narrow-wide commercial

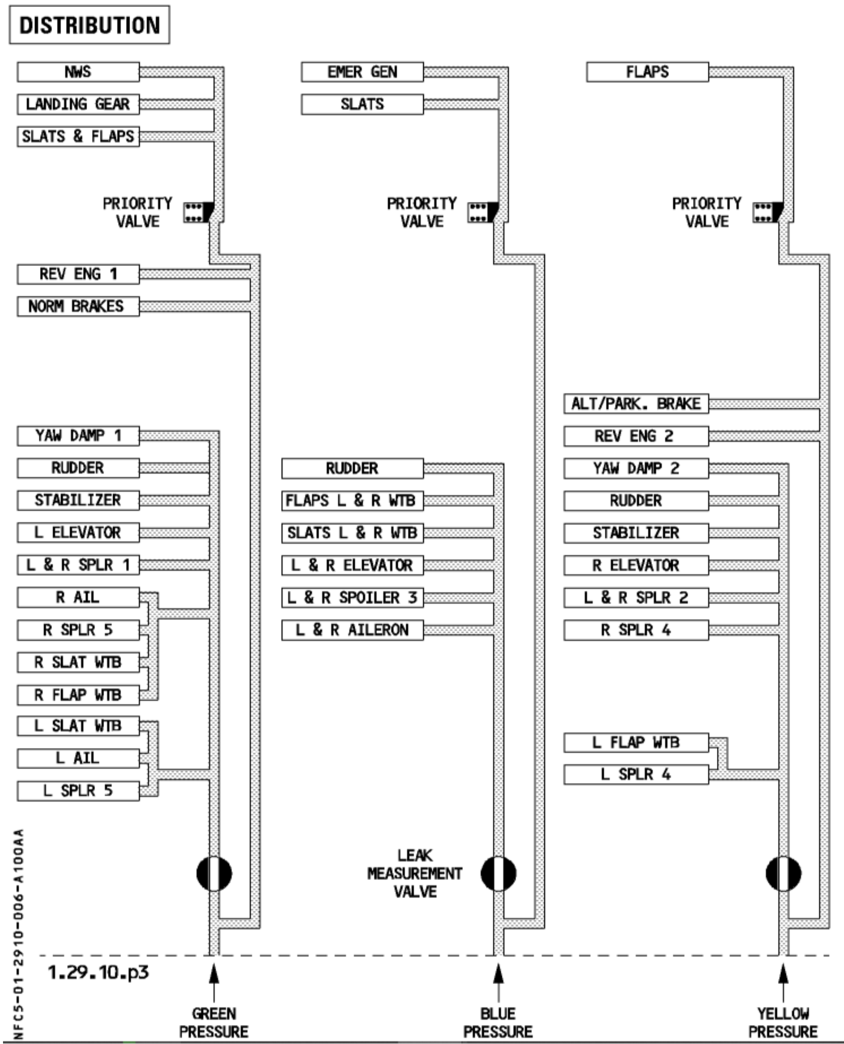



Figure 2.2.1: Distribution of fluid for Hydraulic System of a typical small-medium range plane [4]

aircraft (e.g. A320 or Boeing 737). As evidenced, the hydraulic system is composed of three subsystems: green, blue and yellow that help to increase the whole redundancy of systems to swell the safety of the plane. Both primary and secondary flight control surfaces (see *Sec. 2.4.1* and *Sec. 2.4.2*) are controlled by hydraulic actuators on conventional aircraft [64]. *Fig. 2.2.2* presents how the hydraulic actuators control the flight control surfaces: a hydraulic pump is electrically powered to pressurize hydraulic fluid that is employed to move a cylinder that is directly attached to the control surface. A control valve is handled to command the direction of the fluid and hence, the deflection of the control surface it is attached to.

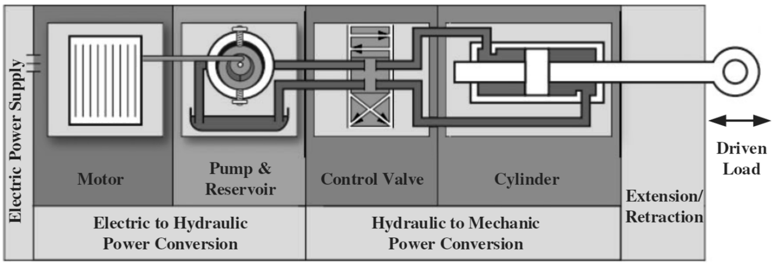



Figure 2.2.2: Hydraulic System for Aircraft composition [53]

### 2.2.1 Primary flight control surfaces hydraulic actuators

Concerning the primary flight control surfaces actuators, they are segmented redundantly onward the three different hydraulic subsystems, as *Fig. 2.2.1* displays:

- The green system is responsible for the actuation of rudder, right and left ailerons and left elevator.
- The blue system moves the rudder and both left and right ailerons and elevators.
- The yellow system pressurizes the hydraulic fluid to deploy the rudder and the right elevator.

So far as one can appreciate, a clear redundancy of systems is present because those critical primary flight control surface actuators are included in several hydraulic subsystems. This feature guarantees that, in case of failure of an individual hydraulic power subsystem, the primary flight control surfaces will be still movable and actuated by another hydraulic subsystem; hence, the aircraft will still be controllable.

### 2.2.2 Secondary flight control surfaces hydraulic actuators

With regard to typical flaps, slats, spoilers, and other secondary flight control surfaces actuation systems, those consist of a central motor located inside the fuselage which drives the individual control surfaces drive stations via a rotating torque shaft system [30]. As a consequence, only synchronous operation of these control surfaces can be achieved [69].

## 2.3 Electrical system

The electrical system is increasing its significance into aircraft systems due to an important increment on power density together with its simple installation [78]. This section describes how the conventional electrical systems used in aviation are classified, from the voltage standards to the electrical generation systems. As explained in *Sec. 2.1*, electrical systems are used in conventional aircraft to power avionics, cabin and aircraft lighting, so its percentage of power consumption compared to other types of power is slight.

### 2.3.1 Voltage standards for aircraft systems

Different voltage standards exists for electrical system on large civilian aircraft [78]. These standards are designed in order to increase electrical system's reliability and robustness [23], and are segregated in *Tab. 2.3.1*.

Table 2.3.1: Electrical standards on conventional aircraft [78]

| Voltage $[V]$ | Electrical Network | Uses |
|---|---|---|
| 28$V$ | DC | Low power loads and avionics on large aircraft |
| | | Whole electric system on small aircraft |
| 270$V$ (+135$V$) | DC | Military aircraft |
| | | Subsystems of various larger aircraft |
| 115$V$ | AC at $f = 400$ $[Hz]$ | Biggest loads on large civilian aircraft |

Regarding the power of the electrical system on conventional aircraft, 100 [kW] is approximately the typical installed power capacity of the electrical system on an existing medium-range aircraft –such as a Boeing 737–[78].

### 2.3.2 Electrical generation systems

Nowadays most commercial aircraft use a complex mechanism gearbox called Constant Speed Drive (CSD) to convert a variable-speed shaft from the gas turbine into a constant speed shaft [69] in order to generate a three phase 115V AC electrical power at a constant frequency of $f$=400$[Hz]$ by means of Constant Frequency Integrated Generator (IDG) [57] (see *Fig. 4.2.1* (a)). The voltage from the generator can then be controlled by an exciter and a simple control loop [78]. This gearbox is expensive to purchase and to maintain because of its complexity: the output frequency must always be constant despite the wide range of rotational speeds of the engines during any actuation in order not to damage the electrical system [76].

## 2.4 Basis of flight mechanics

Basically, those big and heavy metal or composite material structures called aircraft can fly thanks to the aerodynamic forces: lift on the vertical axis (*Eq. (2.1)*) and drag on the horizontal axis (*Eq. (2.2)*)[66]. Lift and drag are induced along the wingspan of a plane as a result of the gradient of pressure of the air flowing above and below the wing surface Since the lift is an upwards force it opposes to gravity and permits the plane fly [43].

$$L = \frac{1}{2}\rho V^2 S C_L \tag{2.1}$$

$$D = \frac{1}{2}\rho V^2 S C_D \tag{2.2}$$

The manoeuvrability of an aircraft flying through air can be modeled using three different axis: longitudinal or directional, vertical and lateral. Those axis are associated to three rotations, giving to the aircraft three degrees of freedom: roll on the longitudinal axis, yaw on the vertical axis and pitch on the lateral axis [66], as presented in *Fig. 2.4.1*.

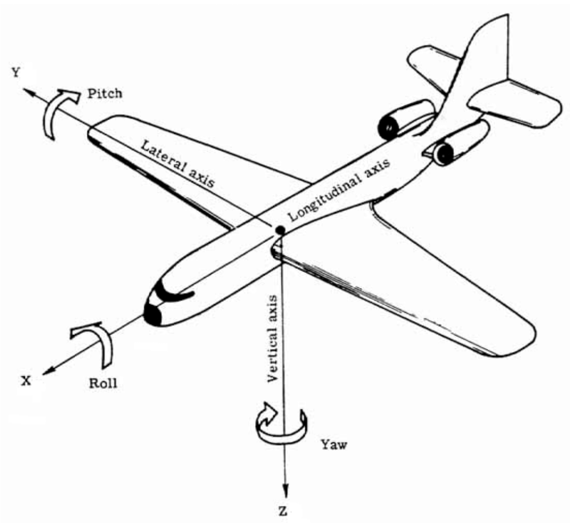



Figure 2.4.1: The three axis of rotation of an aircraft [50]

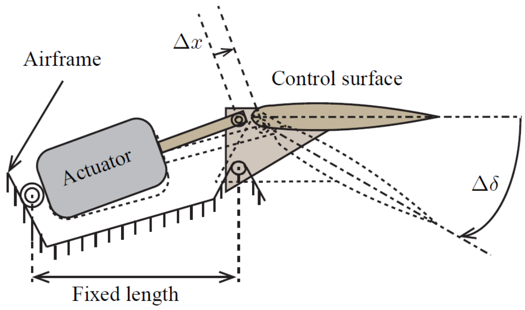



Figure 2.4.2: Actuation of a hinged control surface using a linear actuator [14]

### 2.4.1 Primary flight control surfaces

Roll, pitch and yaw rotations are mainly associated with three different control surfaces, being them the primary flight control surfaces [78]: the elevator controls the pitch, whereas the directional and lateral axis are both coupled and commanded, respectively, by the rudder and the ailerons [43]. Those three control surfaces are critical for flight, hence without any of them the aircraft loses one degree of freedom and becomes no longer controllable [78].

To move those control surfaces –movement known as surface deflection–, heavy actuators are required because the magnitude of the aerodynamic forces, and thus the torque or hinge moment, to overcome is the order of $1 \cdot 10^{-3}$ $[N]$ –i.e. [kN]– for a short and medium range commercial plane such as Boeing 737-800 [14, 64], as collected in *Tab. 2.4.1*. To generate this amount of force typically hydraulic linear actuators are used (*Fig. 2.4.2*), commanded by the Electronic Control Unit (ECU) and hydraulically powered by the central hydraulic pumps [69] (see *Sec. 2.2*).

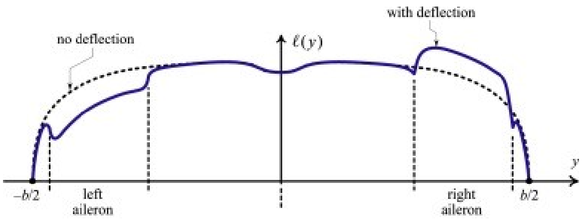



Figure 2.4.3: Aileron effect on wing spanwise lift distribution in a left turn (right aileron down) [48]

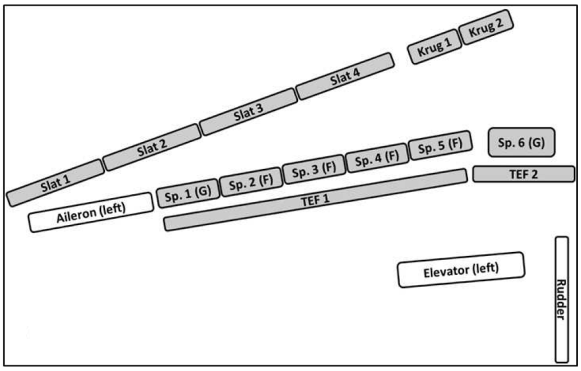



Figure 2.4.4: Primary (grey) and secondary (white) flight control surfaces distributed on a wing semispan [14]

Regarding the roll rotation –underestimating the aforementioned coupled effect between directional and lateral axis– it is directly caused by the deflection of the ailerons [66]. If the pilot wants the airplane to turn left, he will send a command that will drive the system to deflect upwards the left aileron and downwards the right aileron, originating a gradient of lift distribution between the left and the right wings [43, 66]. Consequently, since the lift on the right wing increases and the lift on the left wing decreases, the plane will roll into an anti-clockwise direction from the pilot's point of view [58], as presented in *Fig. 2.4.3*.

### 2.4.2 Secondary flight control surfaces

Flaps, slats, spoilers and other control surfaces are not critical for flight, which means that the aircraft can be flown without these minor surfaces. These surfaces are useful for the comfort and efficiency of the flight but are not essential [43]. As a consequence, they are known as secondary flight control surfaces. Hence, the actuators for these surfaces are also referred to as secondary actuators [78].

Flaps, to illustrate, are only deployed at low airspeed, when a rise on lift is required, for example during take-off. When flying, they are reefed inside the wing and are not used during cruise [64]. Besides, slats are also used to increase the lift, since they help to enlarge the stall angle of attack ($\alpha_s$) [43], but, on the contrary, they are continuously operating during flight [64].

On the other hand, spoilers are used for roll control: they have the ability to modify the lift distribution [17] as ailerons do (*Fig. 2.4.3*). They are also hydraulically powered, as stated in *Sec. 2.2.2*.

*Fig. 2.4.4* collects the typical distribution of both primary and secondary flight control surfaces on a wing semispan of a small or medium range commercial aircraft. As it can be appreciated, primary flight control surfaces are unique: only an aileron, an elevator and a rudder exist on the wing; while slats, flaps, spoilers and Krueger flaps are separated into multiple different surfaces all along the geometry of the wing.

### 2.4.3 Control surfaces actuation requirements

Control surfaces have to overcome huge aerodynamic forces when being deflected: the moment generated along the hinge line of the control surface is known, as aforementioned, as the hinge moment ($H$)–refer to *Sec. 8.1* for more information–. Airworthiness regulations for medium and large planes specify the flight conditions to be considered when sizing the primary flight control surfaces actuators (see *Sec. 7.1* for an extended explanation). If those are correctly analyzed, a flight envelope similar to *Fig. 2.4.5* can be drawn for each specific plane. This

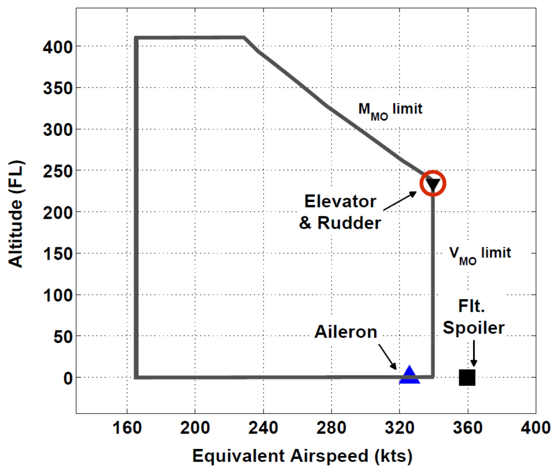



Figure 2.4.5: Boeing 737-800 flight envelope with maximum hinge moment flight condition for ailerons, elevator, rudder and flight spoilers [14]

flight envelope is valuable to precisely set up the flight conditions to size each particular control surface actuator.

In *Tab. 2.4.1*, Chakraborty et al. (2015) [14] summarize the actuation requirements –actuating load and rate of displacement or rotation– of flight control surfaces on a narrow-wide plane. To do so, Chakraborty et al. (2015) [14] established different criteria, as recapitulated below, firstly for the actuation loads:

- The flight condition that yields the maximum hinge moment coefficient for each primary control surface was obtained from Federal Administration Regulations (FAR)[2]. As a result, the flight envelope of *Fig. 2.4.5* is generated.
- The actuating loads for high lift devices highly depend on the mechanisms they are attached to [17]. For this reason, "the maximum actuation loads for the Krueger flaps and slats were derived using force and moment coefficients from published wind tunnel analyses [34, 46]" (Chakraborty et al., 2015 [14]).
- The rating of the linear ballscreaw flap actuators of the Boeing 737 set the basis of the required actuation load of the trailing edge flaps.

The criteria for the angular and linear actuation rates is recapitulated underneath:

- The maximum angular rates for primary flight control surfaces and for flight spoilers where established on the observed trend of higher rate requirements for modern aircraft, using automatic and fly-by-wire flight control systems [17].
- Deploying videos of high lift devices were used to figure out their actuation rates.

Table 2.4.1: Summary of flight control surface actuation requirements for example small narrowbody aircraft [14]

| Control Surface | Actuation Load | Rate | #/Aircraft |
|---|---|---|---|
| Ailerons | 5.2 $[kNm]$ | 60 [º/s] | 2 |
| Elevators | 7.6 $[kNm]$ | 60 [º/s] | 2 |
| Rudder | 8.2 $[kNm]$ | 60 [º/s] | 1 |
| Flight spoilers | 4.2 $[kNm]$ | 60 [º/s] | 8 |
| Ground spoilers | 3.8 $[kNm]$ | 40 [º/s] | 4 |
| Trailing-edge flaps | 51 $[kN]$ | 102 [mm/s] | 4 |
| Leading-edge slats | 6.3 $[kN]$ | 60 [mm/s] | 8 |
| Krueger flaps | 5.6 $[kNm]$ | 16 [º/s] | 4 |

*Tab. 2.4.1* and *Fig. 2.4.5* are effective tools when sizing flight control surfaces actuators. Nevertheless, hydraulic actuators are widely used in the aeronautical industry, consequently, the choice of size and power can be quickly deduced by similarity [58, 64]. *Sec. 8.1.1* proposes a semiempirical sizing methodology based on hinge moment derivatives that will be helpful

[2]FAA regulations have an equivalent on EASA regulations: CS, that will be later on discussed along *Sec. 7.1*.

when calculating actuation hinge moments for electromechanical actuators given that only a few precedents exist and so, similarity approach is not suitable.

## 2.5 Outline

On the whole, *Chap. 2* evinces how power consumption is distributed on a conventional plane nowadays: hydraulic and pneumatic power represent the highest amount of non-propulsive power consumption. It is because, both central hydraulic pumps and compressor are continuously operating during flight, reducing considerably the whole efficiency of the plane, hence they are not capable to offer power on demand.

Furthermore, the electrical system is described: the electrical AC power is generated by means of CSD mechanically linked to the shaft of the turbines, at the constant frequency of $f = 400[Hz]$ and with a voltage level of 115V for the biggest loads on medium and large civilian aircraft. For the avionics and control systems, a 28V DC electrical network is suited.

Finally, the flight mechanics of a plane are described together with the flight control surfaces: primary control surfaces are critical for flight, while secondary control surfaces are only used to increase the performance and comfort of the flight. On conventional commercial planes, control surfaces are typically actuated by linear hydraulic actuators driven by central hydraulic pumps: a similarity sizing method is often used. However, to accomplish the aim of this project, actuation requirements may be described and calculated: *Tab. 2.4.1* condenses the actuation loads and displacement rates of control surfaces on a small narrowbody aircraft –e.g. Boeing 737-800–, which will be worthwhile in further chapters.

# Chapter 3

# State of Art

In this chapter, a detailed description about background of this project is presented together with related work. MEA is analyzed from its beginnings to the current state; specially on those work-related to actuator systems. At the end, a synthesis of the ongoing trend towards MEA is recapitulated to set the basis for *Chap. 4* and *Chap. 5*.

## 3.1 More Electric Aircraft (MEA)

As explained before along *Sec. 1.1*, More Electric Aircraft is a technological philosophy which is trying to increase the percentage of electrical powered systems in aircraft [63]. This concept offers many potential benefits in the design and efficiency of future planes [78]. Power electronics will play a fundamental role for this important step change in aeronautics industry [57, 63].

To overcome flight delays and high maintenance costs caused by the location and repair of fluid leaks [57] and to increase the whole efficiency of aircraft systems, *"there has recently been a considerable and dramatic change in the system design of some future aircraft. Electrical systems are being used in applications that have traditionally been powered by hydraulic, mechanical, or pneumatic power sources"* – (Bozhko and Wheeler, 2014 [78]). As a consequence, future aircraft are able to be quieter and more fuel efficient, reducing their carbon footprint as well as their associated maintenance costs [78]: compared to centralized hydraulics, control surfaces electric actuators have the advantage of *Power on Demand* – energy is only consumed when the control surface moves or overcomes a load– [14].

### 3.1.1 Powered-by-wire systems

As aforementioned along *Sec. 1.1*, PBW systems transport power in wires between the devices instead of pipelines. Some studies have shown that PBW actuators offer many advantages due to their fault-tolerant capability and exclusion of pipes and fluids [53]:

- Safety and reliability are increased due to the lack of poisonous and flammable hydraulic fluids.
- Weight, volume and complexity of power transmission paths are reduced.
- Maintenance costs are simplified and diminished because of the absence of hydraulic fluids and an enhancement in diagnostic capability.
- Ground service equipment is expected to be cut-out by a 30% on an all-electric aircraft [47].
- Dynamic characteristics are improved and energy efficiency is increased, achieving a significant reduction in the fuel burn [31, 47].

Some successful examples of switching into a more electrical aircraft technology are Airbus A380 and Boeing 787, as presented in *Tab. 3.1.1*. As it is summarized, flight control surfaces are redefining their actuators: Airbus A380 has implemented electromechanical actuators for slats and trimmable horizontal stabilizer, while Boeing 787 have them installed for mid-spoiler surfaces actuation as well as trimmable horizontal stabilitor. Since primary flight control surfaces are critical for flight (*Sec. 2.4.1*) and it is important to avoid jamming faults on them (*Sec. 5.1.3*), electro-hydraulic actuators are suited into Airbus A380 for ailerons and elevators actuation.

Table 3.1.1: Electrification of systems in Airbus A380 and Boeing 787

<table>
<tr><th></th><th>Airbus A380 [26]</th><th>Boeing 787 [53]</th></tr>
<tr><td rowspan="3">EMA</td><td>Slats</td><td>Landing gear braking</td></tr>
<tr><td>Trimmable horizontal stabilizer</td><td>Trimmable horizontal stabilitor</td></tr>
<tr><td>Thrust reversal actuation</td><td>Mid-spoiler surfaces</td></tr>
<tr><td rowspan="2">EHA</td><td>Ailerons</td><td rowspan="2">Not in use</td></tr>
<tr><td>Elevators</td></tr>
</table>

Despite this apparent success, more development and improvement of systems is still required: MEA will surely need a highly reliable, fault tolerant, autonomously controlled electrical power system from the electrical power source to the loads [5]. Additionally there are still a number of areas where improvements are required in terms of power density, weight, volume and cost [78].

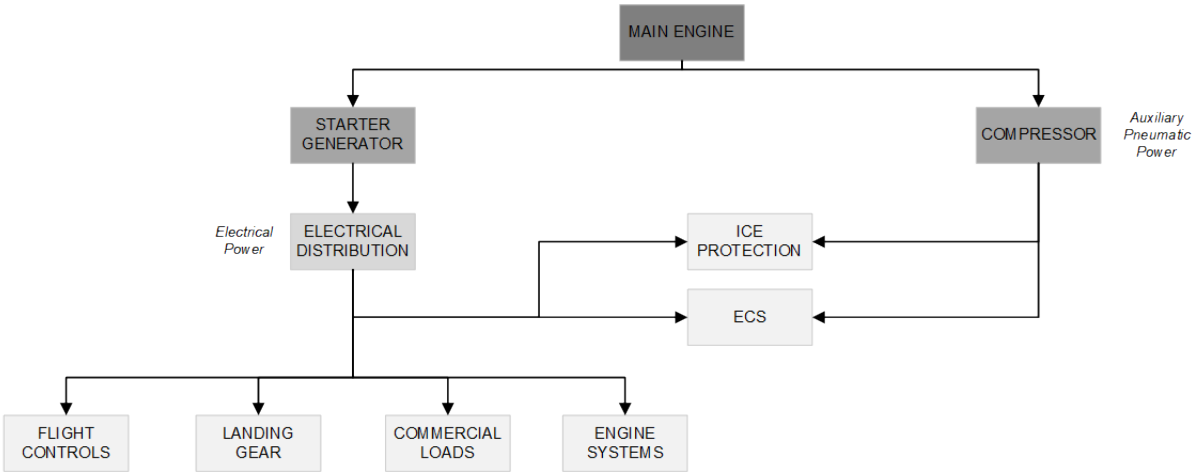



Figure 3.1.1: Schematic of MEA Power Distribution. Adapted from [57]

*Fig. 3.1.1* specifies a feasible approach to on-board energy power management and drive systems which consists of the electrification of most of the systems that make up the aircraft. In contrast to *Fig. 2.1.2*, *Fig. 3.1.1* submits the electrification of most of the systems in a plane, cancelling the need to install a central hydraulic pump and reducing considerably the power consumption of the compressor. IPS and ECS are still powered by pneumatic power from the compressor, although they also drain electrical power from the generator. The model presented in *Fig. 3.1.1* is expected to cut out the power consumption of plane systems by a considerably percentage [57].

## 3.2 Related work

The development of future commercial aircraft comes together with lots of research. These worldwide investigations have given rise to more advanced approaches to drive systems and on-board power management for planes [57]. Regarding energy efficiency, it is accepted that electrical systems are far more prospect than the conventional ones [30, 57] –specified along *Chap. 2*–.

Notwithstanding that MEA is a concept that involves the overall electrification of aircraft, steps towards the implementation of this notion into the actual industry are taken in two different ways [57]:

- Increasing electrical power generation capability of planes by removing current air and hydraulic engines, as demonstrated in *Fig. 3.1.1*. Still, significant changes in electrical generation, network techniques and fault protection must be contemplated.

- Substituting hydraulic actuators for electromechanical actuators, as evidenced in *Tab. 3.1.1* –thus, this is the aim of the project–. As aforementioned along *Chap. 1*, the indicated reduces at once weight and maintenance costs; being also easier to achieve [30].

Evidently these two tendencies can be coupled to obtain a combination of them both, thus expanding the electrification of aircraft. Even so, most of the research carried out to date has only investigated one of the aforementioned trends separately.

## 3.3 Outline

To sum up, weight and maintenance cost reduction and rise on safety and reliability of PBW systems in comparison to conventional systems could help to advance into MEA.

Consequently, there are two main ongoing trends towards MEA: in the one hand, the goal is to increase the electrical power generation capability of planes and the number of electrically powered systems (*Chap. 4*), as evidenced in *Fig. 3.1.1*. On the other hand, there is the objective to replace hydraulic actuators with electromechanical actuators (*Chap. 5*), thus reducing the maintenance costs and the total weight of the system, as shown in *Tab. 3.1.1* for Boeing 787 and Airbus A380.

# Chapter 4

# More Electric Aircraft Power Systems

This chapter summarizes relevant information about the power stage for MEA. Three different alternative for the generation and distribution of electricity into an aircraft system are presented, considering the different voltage standards of conventional aircraft systems (see *Sec. 2.3.1*).

Furthermore, power converter typologies are analyzed and compared, together with the discussion of where the power converter stage on the electrical system of an aircraft may be installed.

## 4.1 Voltage standards for More Electric Aircraft

In contrast to *Sec. 2.3.1*, if the power generation capability of new aircraft is increased, larger loads can be put together on the electrical system of the plane: 100 [kW] is approximately the typical installed capacity of the electrical system on an existing medium-range aircraft –such as a Boeing 737–, whereas on planes where electrical systems are significantly larger than any previous aircraft –such as the Boeing 787– this power level increases to more than 1 [MW] [78]. To satisfy this rank of power demand an increase in electric current is required and, as a consequence, bigger cables are needed, increasing the total weight of the system [57]. To suppress this trouble, higher voltage electrical systems are considered, including the emerging voltage standards presented in *Tab. 4.1.1*.

Table 4.1.1: Electrical emerging standards for MEA [78]

| Voltage [V] | Electrical Network | Frequency [Hz] |
|---|---|---|
| 540$V$ ($\pm$ 270V) | DC | N/A |
| 230$V$ | AC | Fixed at $f = 400$ $[Hz]$ |
| 230$V$ | AC | Variable between $f \in$ {300-800} $[Hz]$ |

Contrary to *Tab. 4.1.1*, most of large commercial aircraft use a combination of these supplies. *28V DC* is commonly used for flight critical electrical loads, such as avionics [75]. The largest loads, meanwhile, are supplied from a single or combination of high-voltage AC or DC systems [78].

## 4.2 Electrical generation systems

Apart from the typical generation systems on conventional aircraft explained along *Sec. 2.3.2*, it has been verified that the replacement for the gearbox using power electronics has obvious advantages regarding aircraft's industry new requirements such as lower cost, increased reliability, easier maintenance and higher operating temperatures and speeds [57]. Two different alternatives to CSD are presented below (*Sec. 4.2.1* and *Sec. 4.2.2*) together with their main advantages and drawbacks.

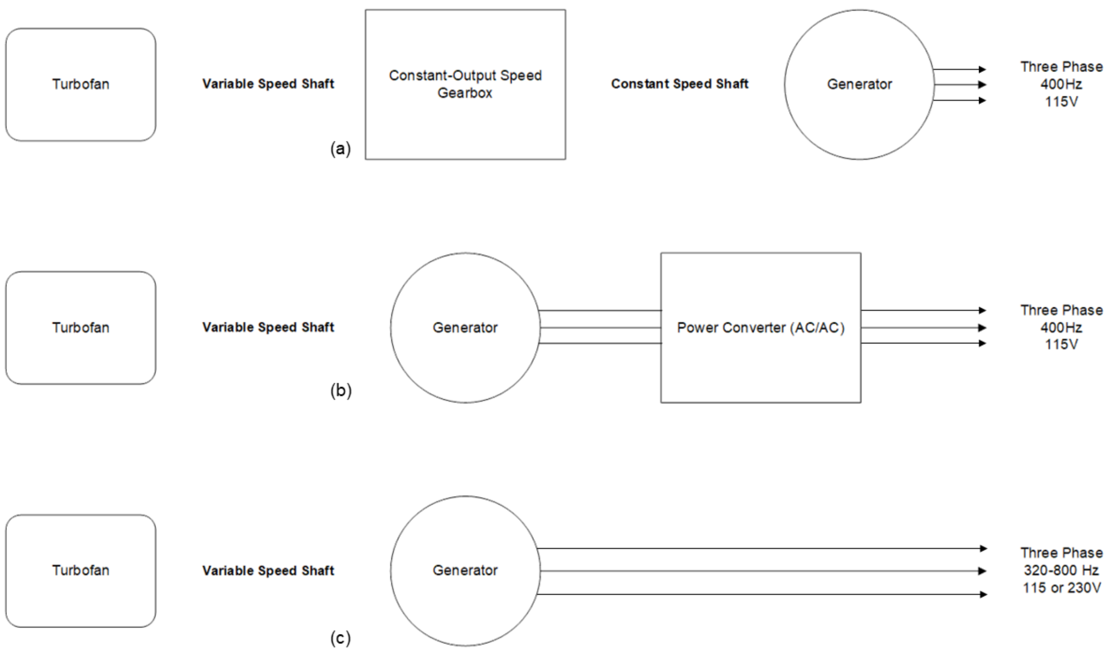



Figure 4.2.1: (a) IDG system using a CSD to generate electricity at a constant frequency. (b) Constant-frequency generation system using a power converter. (c) Variable frequency generation system. Adapted from [78]

### 4.2.1 Variable Speed Constant Frequency Generation with power converters

Power converters are a great alternative technique for generating a constant frequency supply (see *Fig. 4.2.1* (b)) with the generator connected directly to the engine shaft [78], or by means of a mechanical reduction [75]. During flight, the frequency of the generator will be variable due to the changing demands for the gas turbine speed [58]. To convert this variable-frequency supply into a fixed-frequency and voltage supply, one or a combination of the following Variable Speed Constant Frequency (VSCF) systems must be employed:

- Suitable AC/AC power converter and filter [78]
- High quality three phase AC-DC conversion plus subsequent DC-AC conversion [57]

Perhaps that AC/AC power converters are better for aeronautics than AC/DC/AC ones, specially because of the suppression of bulky reactive components [57].

It is of interest for MEA the use of bidirectional power converters because of its two different modes of actuation: in motoring mode, the system acts as a starter for the aircraft engine by supplying electrical energy through the power converter; while in the generating mode, electrical energy is generated in a variable frequency and transformed into a constant frequency by the power converter before being distributed to the whole aircraft [57].

Bidirectional power converters can be assembled by using different power converter types, being matrix converters and cycloconverters the most remarkable [78]. A matrix converter (see *R. Attachments A* for more information) "consists of nine bi-directional switches arranged as three sets of three so that any of the three inputs phases can be connected to any of the three output lines. The switches can be then controlled in such a way that the average output voltages are a three phase set of sinusoids of the required frequency and magnitude" (Rosero et al., 2007 [57]). These characteristics make them a promising technology for the near future of aircraft, including advantages such as a higher power ratio, unity power factor control and the nonexistence of bulky reactive components [57], as aforementioned.

The use of a power converter has the advantage that no gearbox is needed between the gas turbine shaft and the generator. Even so, the main drawback is that this power converter has to process all the generated power and, therefore, full power-rating and high reliability to get the required level of safety must be achieved [78].

### 4.2.2 Variable Speed Variable Frequency generation

Although power converters may be a feasible alternative to CSD for the generation of electrical power at constant frequency, the use of electricity at variable frequency could also be an interesting fallback. If the electrical systems and associated loads were able to operate with a variable frequency, the output of the generator could be directly connected to the electrical system of the aircraft [57, 75]. *Fig. 4.2.1* (c) shows another feasible alternative to generate electrical power: the output of the electrical generator provides a variable-frequency supply with the frequency directly related to the rotational speed of the gas turbine, typically between 320 to 800 [Hz] [78], known as Variable Speed Variable Frequency (VSVF).

The advantage of this configuration lie in its simplicity: maintenance costs are reduced considerably because of the discard of the mechanical gearbox and the power converter, while the reliability of the whole system is increased [57]. Nevertheless, aircraft's systems must be designed to operate with variable frequency generation and distribution, which has not yet been chalked up [57]. As a consequence, if this system is used, nearly all the aircraft loads will require power converters for control [75]. Despite this fact, many applications –including actuators– require a power conversion stage for control, also when a fixed-frequency supply is used [78]. Indeed, the installation of distributed power converters extends the reliability and safe design of aircraft systems, so redundancy can be fixed at a systems level, avoiding any single points of failure within the design [78].

Power-conversion in actuators stage seems to be the most feasible alternative nowadays because of the degrees of freedom it offers while remarkably increasing the level of safety [75].

Table 4.2.1: Comparison between electric power generation systems on aircraft

| Technology | Advantages | Drawbacks |
|---|---|---|
| **CSD** | - Widely used technology<br>- Reliability and airworthiness certification | - Expensive to purchase<br>- Expensive to maintain<br>- Lower speeds and temperatures of operation |
| **VSCF** | - No gearbox needed<br>- Reduction of maintenance costs | - Required full power rating to process all the power<br>- Required high reliability for the safety standards |
| **VSVF** | - No gearbox needed<br>- Reduction of maintenance costs<br>- Simplicity<br>- Redundancy fixed at systems level<br>- Increase on systems reliability | - Aircraft systems must operate with variable frequency<br>- Required distributed power converters aircraft loads [1] |

## 4.3 Power converters

Power electronic converters are electronic devices used to transform electrical energy. *Fig. 4.3.1* presents a basic building block of a power electronic converter. Electrical energy is transformed from defined initial characteristics to desired final attributes inside the power electronic circuit: type of power (AC or DC), voltage, current, frequency, etc. [18].

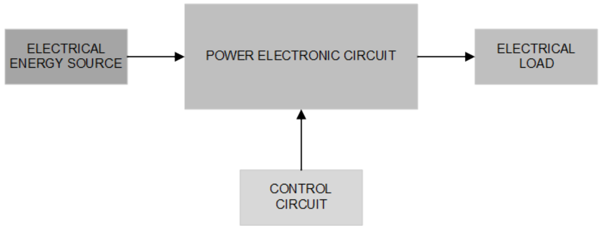



Figure 4.3.1: Basic block diagram of a power electronic converter. Adapted from [18]

Power converters can be used in aeronautics for power conversion in the generation stage (*Sec. 4.2.1*) or just before the power loads for control purposes and power conversion (*Sec. 4.2.2*). *R. Attachments A* summarizes the most important power converters technologies to be implemented in MEA.

### 4.3.1 Overview of power electronic converters

*Tab. 4.3.1* summarizes the main characteristics of three type of power converters: DC-Link, cyclo-converters and matrix converters –those are extensively described along *R. Attachments A*–. While DC-link uses bulky reactive components to storage DC energy that is later converted into an AC wave, cyclo converters and matrix converters delete the need of an intermediate DC phase thanks to the use of multiple switches. Furthermore, the losses are reduced in direct AC-AC power converters, hence increasing the efficiency of the power conversion.

In contrast to cycloconverters, matrix converters are preferred because of no limitations concerning the resulting output frequency: cycloconverters only can offer an output frequency of a 33% the input frequency or lower [39]. DC-link power converters can also offer a smooth control on the output frequency.

With regard to the applications they are suitable for, DC-link converters are employed in high frequency applications, cyclo-converters for low speed applications while matrix converters are fitted when using high inertia applications, being that the case for flight control surfaces actuators [78].

[1]The use of distributed power converters at system level is also used in fixed frequency generation for control purposes. It increases the reliability of systems and helps to get a more accurate control loop.

Table 4.3.1: Comparison between DC-Link Converters, Cyclo-converters and Matrix-Converters [18, 33, 54]

| Parameter | DC Link | Cyclo-Converter | Matrix Converter |
|---|---|---|---|
| **Principle** | D.C link converter uses two power stages A.C to D.C converter and then D.C to A.C converter | Cyclo-Converter converts A.C at one frequency to A.C at some other with a single stage of power conversion. | Matrix Converter converts A.C. at one frequency to A.C. at some other without limitations, with a single stage of power conversion |
| **Forced commutation required** | Required | Not Required | Not required |
| **Losses** | High | Low | Low |
| **Power factor** | High | Low | High |
| **Output frequency** | Smooth control is possible | Can vary in steps | Smooth control is possible |
| **Number of Thristors** | Small | Large | Medium |
| **Applications** | Suitable for high frequency applications | Suitable for low speed large power drives | Suitable for high inertia applications |

## 4.4 Outline

Overall, MEA electrical power system may be redefined since it is starting to represent an important percentage of the non-propulsive power of the plane: it has increased from 100 [kW] on a conventional aircraft to more than 1 [MW] for the Boeing 787, one of the most electrified aircraft nowadays. New voltages standards are set to satisfy the electric requirements for this important amount of electrical power, as summarized in *Tab. 4.1.1*.

To cut out on construction and maintenance costs, as well as putting up a more flexible and tolerant electrical power system, new tendencies have been set up with respect to electrical generation systems that will give the opportunity to evolve from the conventional CSD generation systems to constant frequency generation –by using fault-tolerant and wide-power range power converters– or variable frequency generation. If variable frequency generation is implemented, systems and loads must be capable of working with a variable frequency power source, which is not yet possible.

Moreover, normally planes use power converters at actuators stage for control purposes. As a consequence, power is converted multiple times: when generating the power and just before using it at the actuators stage. This fact can be behold as an opportunity to install only power converters at actuators and loads stage and to start using variable frequency generation technology.

AC power converters can be classified in two different groups: direct AC-AC and DC-link power converters. Cyclo-converters and matrix converters are examples of direct AC-AC power converters. *Tab. 4.3.1* reviews the basic characteristics of this three types of power converters: the conclusion is that –although some new developments may be required– matrix converter is the most suitable power converter for aircraft systems.

# Chapter 5

# Electromechanical Actuators

More Electric Aircraft can not be conceived without electric-based actuators. *Chap. 5* focuses on the studies of electromechanical actuators, its advantages and drawbacks and which solutions may be considered to overcome the undesirable characteristics that move them away from the implementation on primary flight control surfaces for aircraft.

Additionally, the electromechanical actuator system is broke-down into its two main technologies: electrical and mechanical, with the aim to discover, abductively, the best combination of solutions for a flight control surface actuator. Finally, different projects are reviewed and assessed, looking for the main benefits of each of them for MEA.

## 5.1 Electromechanical Actuators for MEA

The trend towards More Electric Aircraft causes a strong need for novel optimized electrical actuators [30]. As seen along *Chap. 2*, modern aircraft control systems can be segregated into two different kind of actuators: primary flight-control actuators and secondary flight control actuators. Typically, hydraulic actuators are employed, but, when replacing them with electrically powered actuators, EMA is the most apparent solution to be implemented [78].

The studies for the electrification of flight control surfaces began on the $1980_s$: experiences for EMA used for primary flight control surfaces were increased [53]. *Tab. 5.1.1* presents two research activities for the implementation of EMA on primary flight control surfaces that have been successfully completed: EMAS for the aileron on C-141 airplane and EPAD using two three-phase BLDC motors for the actuation of the aileron on F/A-18 aircraft. Furthermore, *Tab. 5.1.2* shows a comparison between EMAs and conventional central hydraulic pump actuators within EMAS project.

Table 5.1.1: Characteristics of EMAs in EMAS and EPAD projects [40]

| Project | EMAS | EPAD |
|---|---|---|
| Date | 1985-1986 | 1990-1992 |
| Aircraft and application | C-141/Aileron | F/A-18/Aileron |
| Electric-power technology | 115$V$ AC to 270$V$ DC power converter [49] | 115$V$ AC to 270$V$ DC power converter [31] |
| Motor technology | Two DC motors [49] | Two 3-phase Brushless-Direct Current (BLDC) motors [31] |
| Mechanical assessment | Gear train and linear ball screw [49] | Differential gearbox to drive a single ball screw [31] |
| Flight test hours | 12.5 | 25 |
| Blocked force $[kN]$ | 84.7 | 53.7 |
| No-load speed $[mm/s]$ | 118 (motor at 9600 $min^{-1}$) | 214 |
| Stroke $[mm]$ | 138 | 113 |
| Output backlash $[mm]$ | 0.45 | 0.51 |
| Mass $[kg]$ | 29.5 | 12.5 |
| Bandwidth $[Hz]$ | 4 | 7 |

The architecture of EMA, as presented in *Fig. 5.1.1*, generally consists of a servomotor and an ECU to convert mechanical energy into electrical power in a controlled manner –as detailed along *Sec. 5.1.1*–, together with a mechanical actuation assembly to transform rotational movement into linear displacement –presented in *Sec. 5.1.2*–.

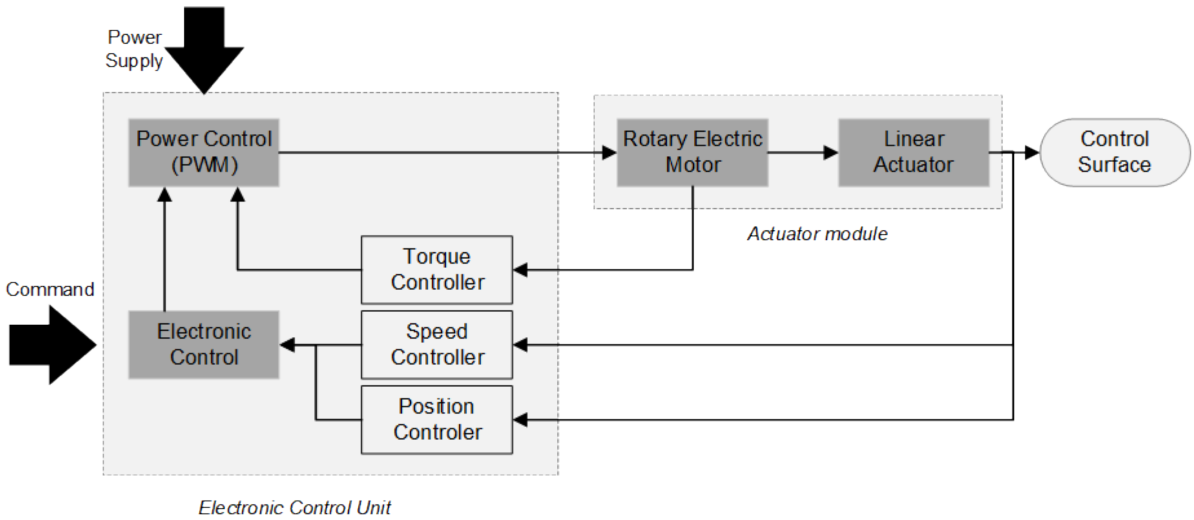



Figure 5.1.1: Conventional EMA architecture. Adapted from [53].

### 5.1.1 Electrical layout of EMA

The basic building block for control surfaces actuator include solid-state power electronics and variable speed motor drives: subsystems such as inverters, converters, controllers and associated components play a fundamental role on the viability of MEA and electric control actuators [57].

*Fig. 5.1.2* shows a generic building block for an EMA system. As it can be seen, a three-phase current supplies a power converter (refer to *Chap. 4*). Once the energy is modulated, is sent

Table 5.1.2: Comparison of EMAS and Conventional Hydraulic PCU Actuators for aileron actuators of C-141 aircraft [49]

| | **EMAS** | **Hydraulic PCU** |
|---|---|---|
| Maximum output force $[kN]$ | 84.7 | 84.7 |
| No-load speed $[mm/s]$ | 118 | 118 |
| Stroke $[mm]$ | 138 | 138 |
| Bandwidth $[kHz]$ | 4 | 4 |
| Weight $[kg]$ | 29.5 | 26.3 |
| Reliability $[hrs]$ | 448632 | 115004 |
| Stiffness $[MPa]$ | $4.1{\cdot}10^3$ | $3.9{\cdot}10^3$ |

to an electric motor, who is responsible for the movement of a reduction gearbox. Each turn of the gearbox displaces the ball screw for a fixed amount. Since this linear actuator is directly attached to the control surface, each linear displacement generates a constant angle of rotation on the control surface.

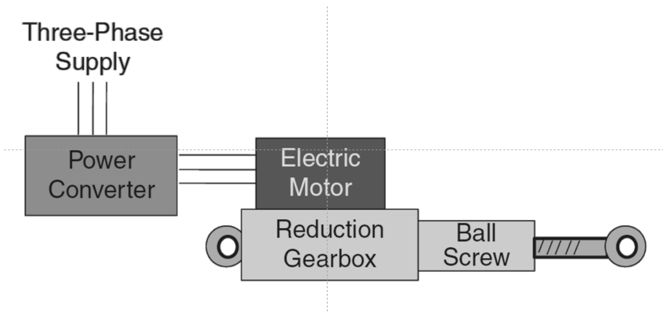



Figure 5.1.2: A system diagram for EMA [78].

Moreover, *Fig. 5.1.2* reveals that the movement of the control surface can be directly commanded by a simple rotary electric motor. The most attainable way out in this case is to develop electrical actuators based on Permanent Magnet Synchronous Motor (PMSM) and the associated power electronics [30], thanks to the precise control loop that can be achieved [75], the high efficiency throughout the full speed range, the ease of refrigeration and the high power ratio [57] compared –as *Tab. 5.1.3* presents– to wound machines such as DC motors, BLDC, induction AC motors [61] or Switched Reluctance (SR) motors [53]. However, the choice of the device normally hardly depends on the power supply onboard [53].

Table 5.1.3: Comparision between electric motors typologies [45, 61]

| **Electrical Typology** | **Power Density** | **Torque Control** | **Positioning control** | **Efficieny** | **Heat generation** | **Cost** | **Noise** |
|---|---|---|---|---|---|---|---|
| PMSM | High | Good | Good | High | High @ low speeds | High | Low |
| Brusheless DC | High | Poor | Medium | High | Low | High | Low |
| SR | Low | Good | Medium | Low | High | Medium | Low |
| Induction AC | Low | Medium | Poor | Low | High @ low speeds | Low | High |

### 5.1.2 Mechanical layout of EMA

To link the rotary motor together with the movement of the actuator some alternatives are attainable. *Fig. 5.1.3* depicts a classification of EMA and their major components based on their mechanical assembly, including linear-geared EMA, linear-direct drive EMA and rotary EMA.

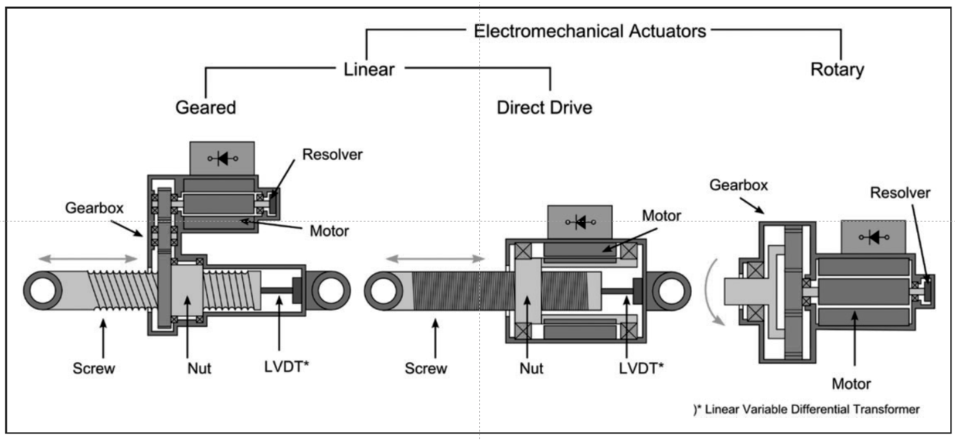
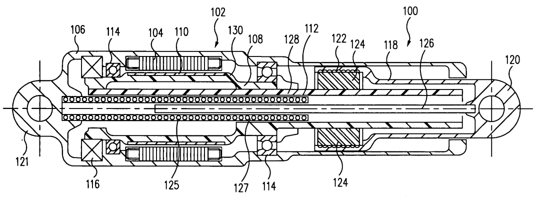


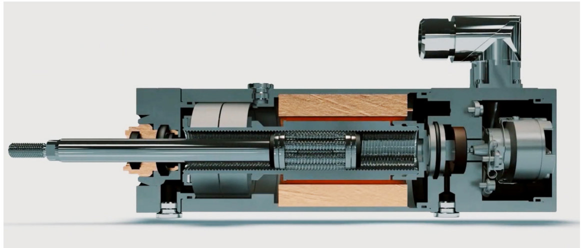

Figure 5.1.3: Classification of different EMA types based on their mechanical design [53]. Adapted from Figure I of Wagner et al. [72]

When installing a linear-geared EMA a jack screw mechanism is used –either on the basis of an ACME screw [57], a ball screw [30] or a planetary roller screw [53] principle, being that the case for *Fig. 5.1.2*–, to transform rotary into linear motion [30, 57]. A hollow screw shaft is selected when stiffness has to be maximized while minimizing weight: a Linear Variable Differential Transformer (LVDT) is placed inside the hollow screw to measure the screw rod's linear position to close the control loop [53]. Additionally, a linear output movement can be also achieved by the installation of "high gear ratio planetary gearboxes with a rotating output lever providing quasi-linear output" (Jänker et al., 2008 [30]).

In regard to direct-drive linear EMA, *Fig. 5.1.4a* and *Fig. 5.1.4b* present respectively a patent and an already in service linear direct drive electromechanical actuator. The common design of this type of actuators consists of a fixed and non rotating lead screw passing through the motor, making the lead nut the only moving part. The motor makes this lead nut rotate, forcing the lead screw to be extended outbound or retracted inwards, depending on the rotating direction [27]. Some other mechanical assessments can be contemplated, such as the use of a planetary nut with helical threads to obtain a mechanical advantage when converting the rotation of the rotor into linear movement of the planetary nut [73].




(a) Patent US20060266146A1 [73]



(b) Inmoco DA99 from Diakont [27]

Figure 5.1.4: Direct-drive linear electromechanical actuator

For rotary EMA, a gear-box is connected to the control surface either directly to the hinge line or by a connecting rod assembly [68] which reduces the speed of the electrical motor whilst increases the applied torque to the control surface [30].

The gearbox used on both rotary and linear-geared actuators helps to transform motor's high speed and low torque to low speed and high torque [53]. Various types of gearboxes are suitable, including harmonic gear reducers, cycloidal reducers or planetary gear reducers, thanks to their compact structures, high efficiency and knack of reaching zero fallout [53].

Direct drive EMA concept seems to be suitable for aerospace applications due to the reduction in the number of components, gearboxes and mechanisms –thus each component of an actuator can fail– and in the weight and size of the actuator –being them critical parameters for the design of aeronautic systems– [72].

Table 5.1.4: Comparison between linear-geared, direct drive and rotary actuators architecture [72]

| Mechanical Design | Advantages | Drawbacks |
|---|---|---|
| **Geared** | - High gear ratio with roller screw | - Need of a gearbox with ball-screw<br>- Wear resulting in backlash and flutter<br>- Susceptible to jamming |
| **Direct drive** | - Small parts count<br>- Concentrated weight<br>- Decreased mechanical jamming susceptibility [53]<br>- Increased efficiency and reliability [53]<br>- Reduced system inertia [53] | - Buffering effect (no mechanisms)<br>- Required servomotor of high torque density and reliability<br>- Overheating |
| **Rotary motor** | - Concentrated weight | - Excessive space to supply the required torque<br>- Need to install linkage mechanisms to place the actuator in a thicker part of the wing |

*Tab. 5.1.4* summarizes the most relevant advantages and disadvantages of the three different architectures for the mechanical assembly of electromechanical actuators. As evidenced, linear actuators are more suitable for aeronautic applications because do not require a big space to be placed in, in comparison to rotary actuators. Furthermore, direct drive EMA is more appropriate than geared EMA because of the reduction of mechanical parts, which reduces the possible failures of the system [53].

#### 5.1.2.1 Rotary to linear motion

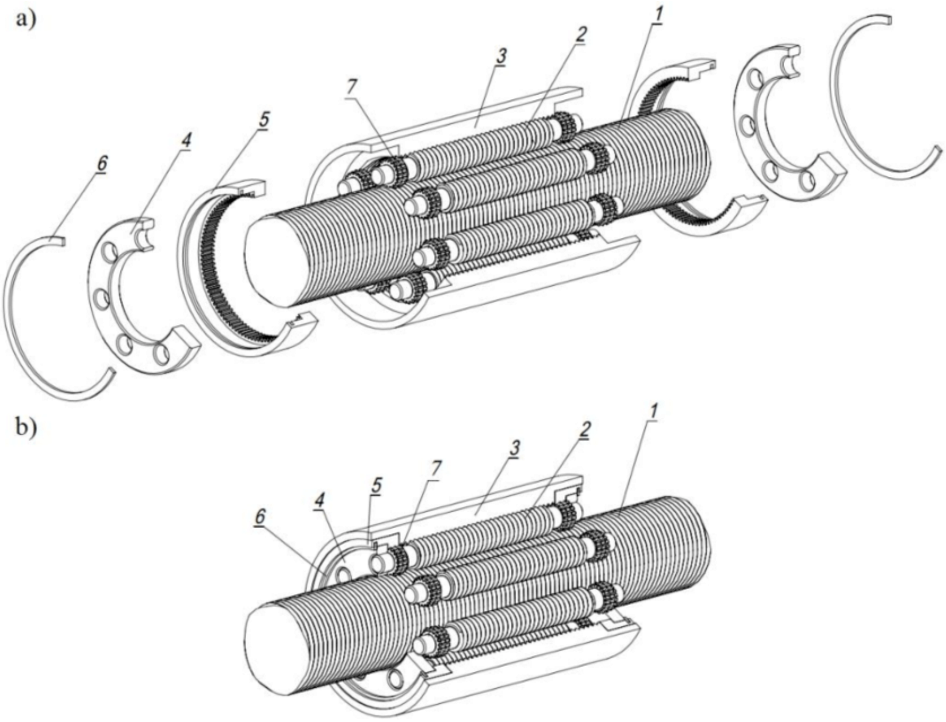



Figure 5.1.5: Planetary roller screw; a) Exploded view, b) Assembly view 1– screw, 2–roller, 3– nut, 4– end plate, 5– ring gear, 6– retaining ring, 7– planetary gear [37]

Motion transformation for both direct-drive and geared EMA is completed by screw mechanisms: acme screw mechanism, ball-screw mechanism and planetary roller screw mechanism (*Fig. 5.1.5*), as aforementioned. *Tab. 5.1.5* summarizes a comparison between these three type of screw mechanism to transform rotary motion into linear displacement. As evidenced in *Tab. 5.1.5*, Planetary Roller Screw Mechanism (PRSM) offers far more advantages compared to acme screw and ball-screw mechanisms.

Thanks to the incremental diameter and number of contact areas compared to ball-screw mechanism, PRSM are more suitable for applications where a significant shock or large external load is applied [53]. Consequently, PRSM seems to be the preferable technology for EMA application [53]. Although the promising PRSM technology, studies must be carried out regarding "multi-body contact, thermo-mechanical coupling, vibration, failure and lubrication, jamming and diagnosis methods of PRSM" (Qiao et al. 2018 [53]) before its application as part of EMA for flight control surfaces.

Table 5.1.5: Performance comparison of screw mechanisms [53]

| Item | Planetary roller screw | Acme screw | Ball screw |
|---|---|---|---|
| Load ratings | Very high | High | High |
| Speed | Very high | Low | Moderate |
| Acceleration | Very high | Low | Moderate |
| Lifetime | Very long, $\ggg$ than ball screw | Very low, due to high friction and wear | Moderate |
| Stiffness | Very high | Very high | Moderate |
| Shock load | Very high | Very high | Moderate |
| Efficiency | High | Low | Very high |
| Electronic positioning | Easy | Moderate | Easy |
| Space requirements | Minimum | Moderate | Moderate |
| Maintenance | Very low | High, due to poor wear characteristics | Moderate |

### 5.1.3 Potential failures of EMA

Any component of an aircraft must satisfy airworthiness requirements (refer to *Sec. 7.1*), specially those regarding Reliability, Availability, Maintainability and Safety (RAMS). In order to fulfill those requirements, it is fundamental to find the potential failures that can occur and their causes. *Tab. 5.1.6* recaps the possible failures of EMA and their origin: the electric part, the structure or the gearing. As stated, a remarkable amount of failures come either from the electric or the gearing systems.

Table 5.1.6: Actuator failure cases and their failure sources [72]

| | Electric | Structure | Gearing |
|---|---|---|---|
| Actuator jam | x | | x |
| Actuator run away | x | | |
| Actuator disconnect / float | | x | x |
| Efficacy loss | x | | x |

*Tab. 5.1.7* summarizes the causes of EMA failures, the affected components and some alternatives to be considered in order to prevent these malfunctions. To start with, the causes of malfunctioning of EMA can be grouped into three main parts, as *Tab. 5.1.6* also presents: electric, structure and gearing.

Electrical failures are originated by the control electronics of the motor, the motor itself or due to the feedback devices. However, the most critical consequence is the jamming of the actuator or its runaway because the actuator becomes no longer operative and hence the airplane loses manoeuvrability, being extremely unsafely if occurs in a primary flight control surface [57, 78].

Malfunctioning related to the structure, meanwhile, can cause the disconnection of the actuator from the control surface, being also critical if it takes place during flight. Fortunately, this problem can be easily prevented if a fail-safe design of the components is carried out and effective quality controls are performed during manufacturing processes.

Moreover, failure cases coming from a gearing fault are normally motivated by mishandling of the EMA itself: overloading, inadequate lubrication, deficient insulation or incorrect sealing. Indeed, most of its consequences can be tracked and repaired before a major issue occurs. However, inadequate lubrication can lead to a blockage and fracture of the bearing, resulting in an actuator jam, which is, as aforementioned, notably unsafely if occurs during flight.

As stated in *Tab. 5.1.7*, the use of redundant electronic design together with the appropriate maintenance program and the installation of monitoring devices to check the correct environmental and operational conditions during the functioning of those systems is fundamental to prevent, detect and settle those problems. Even so, a catastrophic jamming failure can appear, being this an undetectable phenomena where predictive maintenance is impossible because it happens without premonitions [29].

Table 5.1.7: Failure summary of electromechanical actuators [72]

| Part | Components | Failure Causes | Failure consequences | Alternative |
|---|---|---|---|---|
| Electric | - Motor control electronics | - Loss of power supply, short circuit in the motor winding or fault in the motor control electronics | - Actuator jam | - Redundant design of the electronics<br>- Fault detection mechanism |
| | - Electrical motor | - Defect in the motor control or defective position sensor of the EMA | - Actuator runaway | - Redundant design of the electronics<br>- Plausibility check of the measured position by the LVDT against the position calculated from the motor RDS |
| | | - Corrosion of aged connectors | - Efficacy loss: reduction of the nominal output motor torque considering the same current consumption | - Periodic mandatory maintenance checks |
| | - Feedback devices | - Demagnetization caused by e.g. overheating of the PM | - Efficacy loss | - Temperature monitoring of the actuator |
| | | | | |
| Structure | - EMA housing | - Material fatigue | - Disconnection of the actuator from the control surface | - Fail safe design of components |
| | - Rod ends | - Corrosion | | |
| | | - Manufacturing defects | | |
| | | | | |
| Gearing | - Bearings<br>- Screw drive | -Overloading | -Flaking of the raceways, cracks on the surface (surface distress) | - Monitoring |
| | | -Vibration | -Small particles breaking away from the surfaces and thus leading to pits in the raceways | - Monitoring |
| | | -Particles | -Flakening of the raceways and lifetime reduction | - Monitoring the wear of the actuator<br>-Periodic checks of the actuator sealings |
| | | -Inadequate lubrication | -Cage fracture and blockage of the bearing, micro-pitting on the screw | - Monitoring |
| | | -Passage of electric current | -Metal melting resulting to craters on the raceway | - Monitoring |

## 5.2 Electromechanical actuators for flight control surfaces

This section will present the consequences of EMA malfunctioning on flight control surfaces and in relation of RAMS, given that primary flight control surfaces are critical for flight and its safety and correct operation is a must. Additionally, different solutions and successful applicable cases of EMA for primary flight control surfaces will be introduced. *R. Attachments B* presents solutions for secondary flight control surfaces that may also be considered to find a successful solution for a primary flight control surface actuator.

### 5.2.1 Electric-based actuators for primary flight control surfaces

In regard of RAMS, the use of gear-drive EMA in primary flight actuators appears to be problematic as, to date, it has been very complex to certify that the gear will never jam [30, 78], for whatever reason that might be. Primary-flight control surfaces, as seen along *Sec. 2.4.1*, are critical for the manoeuvrability of the aircraft: unless a benign failure mode can be guaranteed, the control surface of the aircraft would not be always controllable [78]. If one ball-screw actuator has jammed, none of the other actuators on the same control surface would be able to move this surface, hence the aircraft becomes no longer controllable [30]. Consequently, it is well accepted that EMA for primary flight control applications face a long path before the compliance of airworthiness and safety regulations [53]. Underneath, some feasible solutions that are nowadays achievable to avoid or prevent jamming fault on primary flight control surfaces actuators are described.

#### 5.2.1.1 *EHA*

The use of EHA is an alternative solution for electric-based actuators in order to avoid jamming and it also aids to reduce weight and maintenance costs compared to conventional hydraulic systems, given that the bulky piping systems and external hydraulic source –as presented in *Sec. 2.2*– are eliminated [30].

This system is steered by local hydraulics and controlled by a fixed displacement pump driven by a variable speed electrical motor [53, 57]. *Fig. 5.2.1* evidences that the actuator position is moved a constant amount for each revolution of

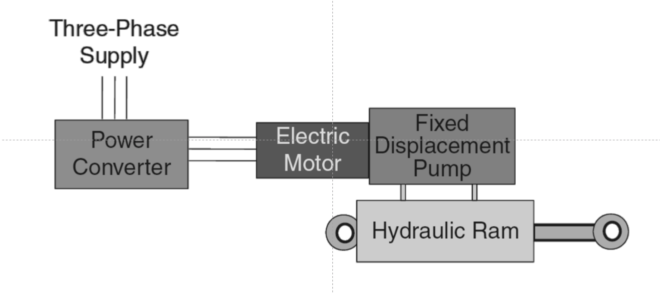



Figure 5.2.1: A system diagram for EHA, following the same philosophy as for *Fig. 5.1.2* [78].

the motor, as it activates straight away
the fixed displacement pump: the pump's speed is directly related to the variation of the flow and thus, to the hydraulic power generated [60]. The transfer of the fluid back and forth from one cylinder chamber to the other is fundamental to control the position of the piston connected to the driven load [60].

As a result of the lack of direct mechanical connection between the motor and the actuator arm, the EHA has benign failure modes, preventing the appearance of undesirable jamming faults [78]. Consequently, EHA owe to date significant advantage when compared to EMA for primary-flight control applications [78]. Airbus A380 transport aircraft uses EHA technology as detailed in *Tab. 3.1.1*, being the first generation of electrically powered flight control actuators [70] and thus, only using two instead of three hydraulic systems [67].

#### 5.2.1.2 *Decentralized monitoring systems for EMA*

Some projects, such as More Open Electrical Technologies (MOET), are trying to demonstrate the feasibility of EMA with a sufficiently low jamming failure rate for fixed wing aircraft primary flight control actuation [32]. To evolve from EHA to decentralized jamming-free EMA the prevention of jamming cases by appropriate technology and monitoring is of major importance nowadays [57], as summarized in *Tab. 5.1.7*.

*Fig. 5.2.2* shows a building block of a feasible solution for an electric actuator control system with jam-prevention functionality. As it can be seen, both supervision and motion systems interact with each other to get a fully controllable actuator. The motion subsystem gets three different feedback signals: torque reference and speed reference from the electrical rotary motor –by means of a Rotary Displacement Sensor (RDS)– and a position reference from the linear actuator –thanks to a Lineal Displacement Sensor (LDS)–. All these signals are analyzed and monitored at real-time: if a fault detection occurs, the system is directly driven into emergency actuation. In addition, running-state and fault detection signals are continuously sent to the communication bus. As a result, decentralized actuation systems are created with a fully fault tolerant control management and communications [57].

#### 5.2.1.3 Wachendorf at al. actuation system

Wachendorf et al. (2008) [71] proposed an actuation system that included a differential gearbox with three shafts but with only two active parallel operating load paths. In case of failure, jamming is avoided thanks to the mechanical self-synchronization, which ensures the free movement of both actuators.

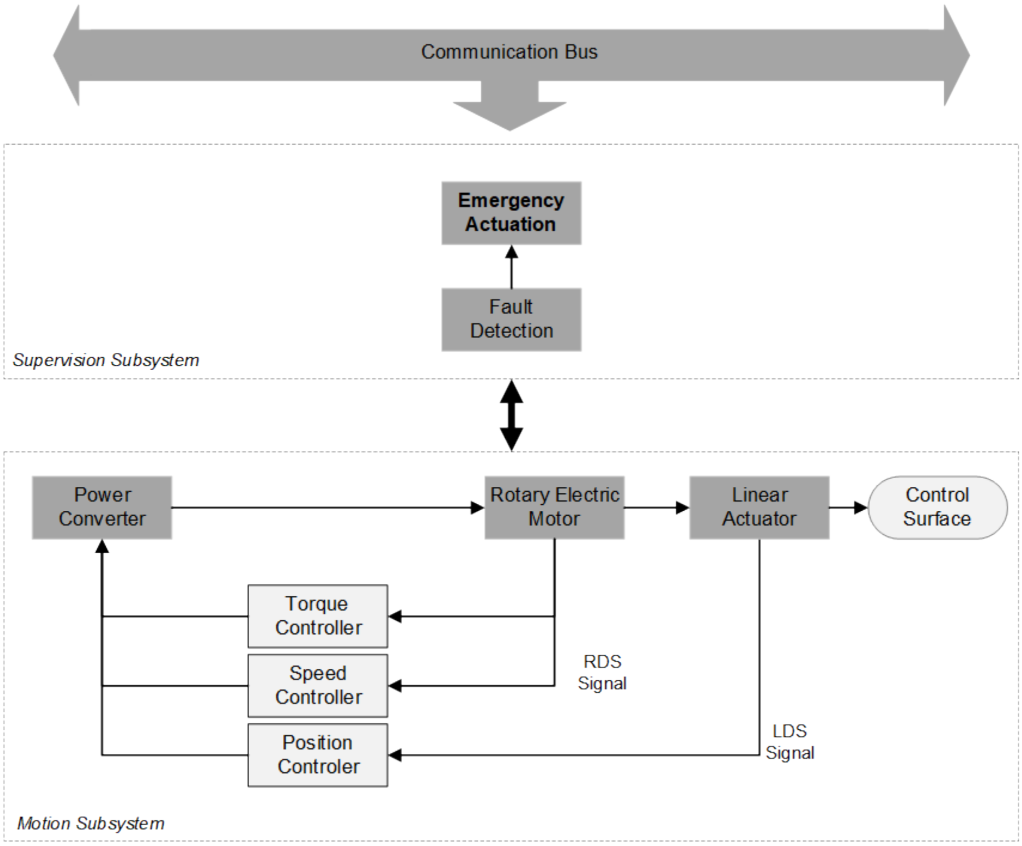



Figure 5.2.2: Direct drive architecture for EMA. Adapted from [57]

## 5.3 Comparison between EMA, EHA and conventional hydraulic actuators

Table 5.3.1: Comparison between actuators systems: conventional hydraulics, EHA and EMA [16, 70]

| Concept | Conventional Hydraulics | EHA | EMA |
|---|---|---|---|
| Power density | High | Medium | Need to be increased |
| Weight | Highest | Medium | Lowest |
| Maintenance | Hydraulic fluid | Hydraulic fluid | Leak-free |
| Background | Extensively used | Already used | In study |
| Monitoring systems | Effective | Effective | In development |
| Geometry of the control surface | Fixed | Can be modelled | Very flexible |

A comparison between conventional central hydraulic pump based actuators, EHA and EMA for flight control surface actuation is reviewed in *Tab. 5.3.1*. As it can be appreciated, EMA still needs to increase its power density, although its weight is the lowest if only the actuator itself is considered [53]. Additionally, EMA reduces considerably the maintenance costs because neither hydraulic fluid nor pipelines are required.

To date, EHA seems to be more suitable than EMA for primary flight control surfaces actuation because they are able to prevent jamming fault, thanks to the elimination of a direct mechanical link –as stated in *Sec. 5.2.1.1*– between the control surface and the structure of the actuator. Despite this apparent advantage of EHA, EMA actuators are easier to maintain and provide a better option for leak-free operation compared to EHA ones as a result of the elimination of local hydraulic devices and hydraulic flow to drive the screw rod [70]. In addition, it is generally accepted that EMA have a weight advantage set side by side with EHA when sized for the same actuation condition [16].

On the contrary, the major challenge on the extensive application of EMA in comparison to EHA and conventional central hydraulic actuation systems is the lack of accumulated knowledge on the prevention and detection of potential jamming failures, on health monitoring and assessment and on thermal management. As a consequence, further experimentation, characterization and validation of EMA systems are needed to ensure that the required performance and reliability would be applied in accordance with airworthiness regulations [6].

## 5.4 Outline

In conclusion, the use of electromechanical actuators may represent an important step towards the whole electrification of aircraft. Before its definitive application into aircraft systems, they must be able to overcome different encumbrances (*Tab. 5.1.7*), specially those directly related to RAMS. Actuator jam and runaway are the most important issues to be studied to date because if one of them occurs in an individual actuator, the whole control surface becomes blocked, and hence, the plane becomes uncontrollable.

The best EMA solution for flight control surface will come from multidisciplinary investigations: electromagnetic, thermal, mechanical and automatic control. The most feasible answer nowadays seems to be the application of distributed PRSM direct-PMSM driven electromechanical actuators (*Tab. 5.1.3*, *Tab. 5.1.4* and *Tab. 5.1.5*) and an efficient monitoring decentralized system (*Fig. 5.2.2*) to continuously control that the actuators are working correctly. If a fault is detected, the individual actuator may be sent into a damping mode, without interfering to the movement of the others. Besides, redundant systems would be used to satisfy and maintain the safe operation of the aircraft even in the event of a failure on any component.

A distributed architecture will also represent an advantage for wing's aerodynamics because multiple configurations of the control surface would be set, whether or not the control surfaces is subdivided into multiple panels. This principle has already been developed for secondary flight control surfaces together with the development of the appropriate EMA, as *R. Attachments B* presents.

# Chapter 6

# Redundancy and monitoring architectures

Redundancy of systems is a must on airplanes for safety reasons (*Sec. 7.1*). As it is stated in *Chap. 5*, distributed actuators architectures will need an efficient decentralized health monitoring system in order to prevent any potential failure of each individual actuator. *Chap. 6* will recapitulate which redundancy and monitoring architectures and devices may be considered for the successful application of electromechanical actuators for flight control surfaces.

## 6.1 Redundancy of EMA systems

Redundancy design technique is fundamental to provide a system tolerance to failure [55]. *Tab. 6.1.1* reviews the three main types of suitable redundancy for EMA: complete redundancy, electrical redundancy and fault-tolerant redundancy.

To start with, complete redundancy composes dissimilar redundancy by using two or more sets of independent actuators, such as an EMA combined with an EHA to make a hybrid redundant actuation system [53], which includes active/active mode, active/no-load mode and active/passive mode [74]. Active/active and active/no-load mode suffer from force fighting due to manufacturing tolerances and channel difference, that can lead to surface fatigue [74]. In contrast, active/passive mode is not affected from force fighting, but the active actuator must overcome the external load caused by the passive one, reducing the response time [19].

Second, electrical redundancy can be achieved by using two or more sets of electrical motors and control devices, as presented in *Fig. 6.1.1*. Two or more electrical path converge until there is a mechanical summing (torque or velocity) to drive the mechanical transmission [53]. However, reliability of those systems decreases because failure isolation cannot be fulfilled [53].

Finally, the next generation of redundancy is fault-tolerant redundancy of EMA architectures. Fault-tolerant redundancy is accomplished by using electric drives with multi-phase motors: this provides fault tolerance against stator coil and power module failures [53]. Cao et al. (2012) [13] defines that the reasonable range of phase number is 3, 4 or 5. Moreover, the power module should be easily replaceable and may also include an autodetection failure system to send this information to the monitoring system [59], following *Fig. 5.2.2*'s philosophy.

Table 6.1.1: Main advantages and drawbacks of three kinds of redundancy [28, 53]

| | **Advantages** | **Drawbacks** |
|---|---|---|
| Complete redundancy | - Increased dissimilarity<br>- A progressive step to introduce EMA into primary flight controls<br>- No electromagnetic coupling | - Force fighting<br>- High failure transients and dormant failures<br>- Complex control loops and reconfiguration strategy |
| Electrical redundancy | - Decreased motor size<br>- Easy control | - No failure isolation<br>- Torque unequal because of unbalanced motor current<br>- Derating use for temperature control and load requirement |
| Fault-tolerant redundancy | - Decreased volume and mass<br>- Good failure isolation<br>- High utilization level | - Complex control<br>- Uncontrolled harmonic currents even when slight magnet field distort |

## 6.2 Monitoring of EMA systems

Monitoring systems are helpful for preventive maintenance tasks as well as for systems supervision during flight in order to detect potential malfunctioning of a single component.

Different monitoring devices can be used to control the proper functionality of EMA, such as the ones presented in *Fig. 5.2.2* and *Fig. 6.1.1*. The philosophy of monitoring devices is to transmit an electrical feedback signal to the controller that is afterwards processed and analyzed to detect if any malfunctioning is happenings [72].

Those supervision subsystems can emit different signals and thus, specific detecting devices are required. As EMA is a multidisciplinary complex concept [53], a mathematical model is needed for simulation works of control strategies and also for the health monitoring system [53]. Yet, it is fundamental to identify measurable parameters to the component fault [72],

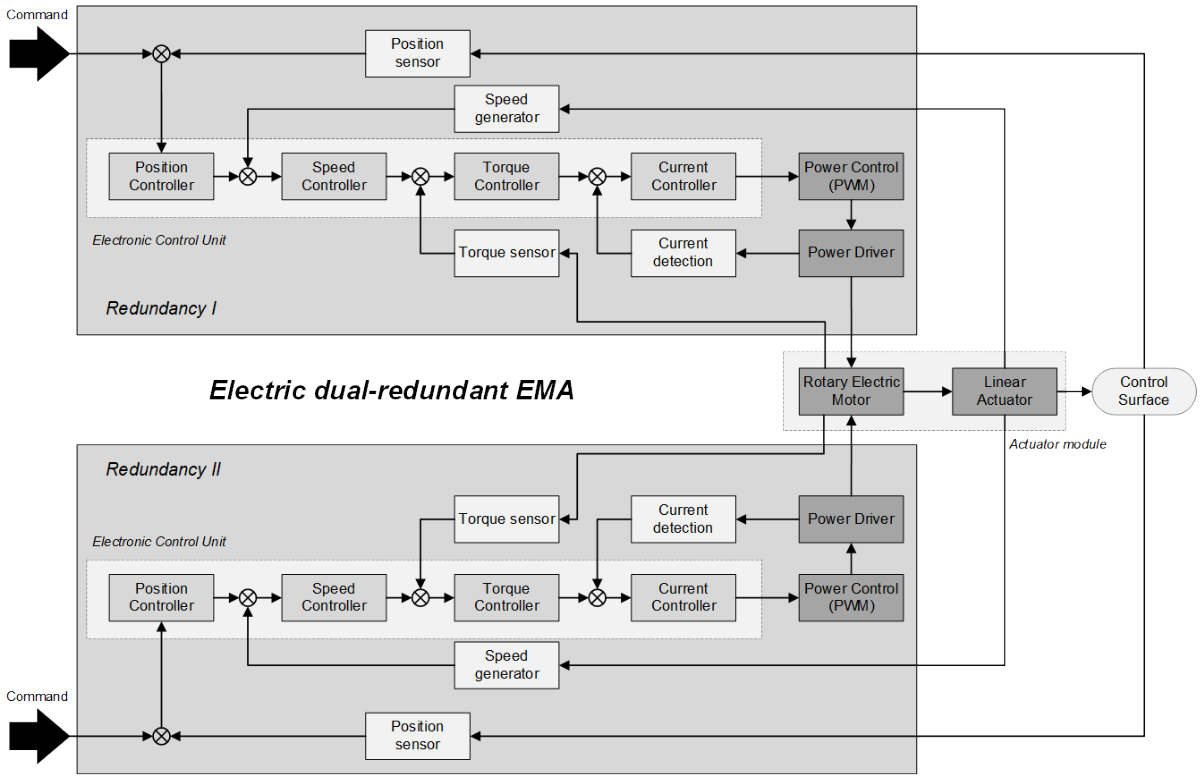



Figure 6.1.1: Electrical dual redundant structure. Adapted from [17]

being this one of the major challenges nowadays for EMA.

*Tab. 6.2.1* summarizes some sensors that may be used for EMA monitoring subsystems. As presented, the efficiency of the actuator can be computed by means of the motor output torque and the axial load. Moreover, backlash can be revealed thanks to the angular position of the nut and the displacement of the screw, whereas the temperature helps to identify flaws in lubrication. In addition, vibration measurements can detect wear as part of a preventive maintenance. Nevertheless, it is difficult to identify faults from a vibration signal in aerospace applications because they tend to have high noise level [72]. This is why ”an alternative to the classical accelerometer-based vibration measurement is currently under development at the DLR Institute of Flight Systems” (Wagner et al., 2011 [72]).

As presented in *Fig. 5.2.2*, ones a single fault is detected in the actuator, it is sent to an emergency actuation state. The consequences of this emergency actuation state will be defined previously depending on the failure and will be directly related to the designed redundancy architecture [72], e.g. start using the second electrical redundancy subsystem if an electrical component malfunctions –as *Fig. 6.1.1* does–, or entering into a passive damping mode if the actuator jams, as happens with Mini-TED (*R. Attachments B*).

Table 6.2.1: Summary of variables and sensors for EMA health monitoring systems

| Variable | Part | Sensor | Principle | Associated fault | Source |
|---|---|---|---|---|---|
| Torque | Motor shaft | Rotary torque sensor | Strain gauge with Wheatstone bridge | Efficiency | [77] |
| Angular position | Nut | RDS: incremental encoder, Biss... | Digital synchronous signal | Backlash | [21] |
| Linear displacement | Screw | LDS: LVDT, potentiometer... | Induced voltage on three solenoid coils | Backlash | [35] |
| Temperature | Bearings Roller screw | Thermistor | Variable resistance | Flaws in lubrication | [42] |
| Axial load | End-plate | Active axial load sensor | Magnetic field | Efficiency | [65] |
| Radial load | Bearings | Thin film sensors | Wheatstone bridge | Disarrangement | [72] |
| Vibration | Wear points | Accelerometer | Capacitor electric field | Wear | [72] |

## 6.3 Outline

To satisfy the safety standards of planes and their systems, both a redundant design of systems and a health monitoring interface are imperative.

Different redundancy design techniques exist, although the most important for EMA are complete redundancy, electrical redundancy and fault-tolerant redundancy. *Tab. 6.1.1* comes to the conclusion that fault-tolerant redundancy would be the most suitable for future EMA: this decreases the volume and mass, possesses a good failure isolation and has a high utilization level.

Apart from redundant systems architectures, health monitoring systems are also important. Those help to track the performance of each component and identify potential malfunctions: for instance, the backlash of the nut and the screw can be detected by evaluating the angular position of the nut and the linear displacement of the screw –by means of RDS and LDS, respectively–, as summarized in *Tab. 6.2.1*.

Although health monitoring systems are helpful to prevent potential failures of planes, it must be stated that they are also prone to failure, which increases systems complexity and costs: the costs of these monitoring systems should not be as high as to exceed maintenance cost savings [53].

# Chapter 7

# Requirements

Each component of a plane must meet specific requirements and regulations in order to satisfy airworthiness regulations and be suitable for the use it has been designed for.

*Chap. 7* recaps the most relevant Airworthiness Regulations to be considered when laying-out an EMA for flight control surfaces actuation regarding sizing methodologies and fundamental safety issues. Moreover, the design requirements that come to the conclusion from the previous chapters (*Chap. 4*, *Chap. 5* and *Chap. 6*) are recapitulated and summarized.

## 7.1 Regulations

The life cycle of an airplane comes across different stages: initial ideas, project, construction, service and seclusion [58]. Aviation safety agencies –e.g. EASA or FAA– must ensure that the safety regulations regarding the entire lifetime of a plane are fulfilled. Those regulations are gathered together in Certification Specifications and Acceptable Means of Compliance –being those the names given by EASA– for each specific aircraft type, such as large aeroplanes, helicopters, balloons, etc. [64]. The following sections are intended to summarize the most relevant CS involved in this project: those are all included in CS-25 *Certification Specifications for Large Aeroplanes*, that "are applicable to turbine powered Large Aeroplanes" (EASA, CS 25.1, 2020 [23]).

### 7.1.1 Flight envelope

As seen along *Sec. 2.4.3*, airworthiness directives define the flight condition that yields the maximum hinge moment for each primary flight control surface. *Fig. 2.4.5* depicts the flight envelope of Boeing 737: from there, the flight condition that respectively gives the maximum hinge moment for each specific control surface can be found.

CS 25.335 defines the design airspeeds for a large plane, which are Equivalent Airspeed (EAS): *Tab. 7.1.1* summarizes those design airspeeds. *Fig. 7.1.1* shows the manoeuvring envelope included in CS-25, considering the EAS and the load factors ($n$). Those speeds defined in *Tab. 7.1.1* help to delimit, as aforementioned, the flight conditions for the maximum hinge moment of each primary flight control surface, as presented afterwards (*Sec. 7.1.2*).

Table 7.1.1: Design airspeeds [23]

| Speed | Definition | Conditions |
|---|---|---|
| $V_C$ | Design cruising speed | $V_{C_{min}} > V_B$ |
| | | $V_{C_{min}} \geqslant V_B + 1.32U_{ref}$ |
| | | $V_{C_{min}} \leqslant V_{level}$ at maximum power |
| $U_{ref}$ | Reference gust velocity in equivalent airspeed | $U_{ref}\ [m/s]\ = 17.07 - 800 \cdot 10^{-6}h$ with $h \in \{0; 4572\}$ [MAMSL] |
| $V_D$ | Design dive speed | $0.8\frac{V_D}{M_D} \geqslant \frac{V_C}{M_D}$ |
| $V_A$ | Design manoeuvring speed | $V_A > V_{s_1}\sqrt{n}$ |
| $V_B$ | Design speed for maximum gust intensity | $V_B > V_C$ |
| | | $V_B \geqslant V_{s_1}\left[1 + \frac{K_g U_{ref} V_C a}{498w}\right]$ [1] |
| $V_F$ | Design wing-flap speeds | $V_F \geqslant 1.6V_{s_1}$ with wings-flaps in take-off position at MTOW |
| | | $V_F \geqslant 1.8V_{s_1}$ with wings-flaps in take-off position at MLW |
| | | $V_F \geqslant 1.8V_{s_0}$ with wings-flaps in take-off position at MLW |

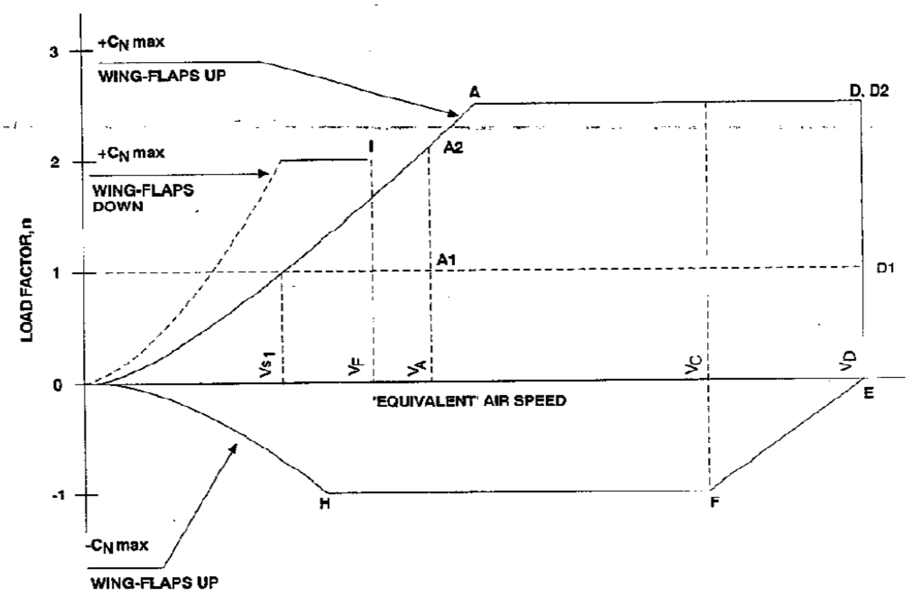



Figure 7.1.1: Manoeuvring envelope for Large Aircraft [23]

[1] Where $K_g = \frac{0.88\mu}{5.3+\mu}$ and $\mu = \frac{2w}{\rho cag}$

### 7.1.2 Flight control surface deployment conditions

Subpart D–*Design and construction* of CS-25 for large airplanes defines the flight conditions to be considered for the deployment of flight control surfaces. To compute the maximum hinge moment for each specific surface the following subsections and the speeds of *Tab. 7.1.1* apply.

#### 7.1.2.1 Rolling conditions

CS 25.349 recaps the rolling conditions, being them directly associated to ailerons deflection, as described in *Sec. 2.4.1*. The maximum hinge moment for the aileron deflection corresponds from among three flight conditions:

(a) Abrupt and full deflection at $V_A$.

(b) The required aileron deflection to produce a rate of roll not less than the produced on (a) at $V_C$.

(c) The required aileron deflection to produce a rate of roll not less than one-third of the produced on (a) at $V_D$.

#### 7.1.2.2 Yawing conditions

CS 25.351 reviews the permissible rudder deflections, resulting them into a yaw rotation. The conditions regarding the maximum permissible loads for rudder deflection are summarized afterwards:

(a) ”With the aeroplane in an unaccelerated flight at zero yaw, it is assumed that the cockpit rudder control is suddenly displaced to achieve the resulting rudder deflection, as limited by” (EASA, 2020 [23]) one of the following:

- The control surface stops.
- A limit pilot force of 1335N.

(b) The aircraft yaws to the overswing $\beta$ angle with the cockpit rudder control deflected to maintain the maximum available rudder deflections defined by (a).

(c) With the aeroplane yawed to the static equilibrium $\beta$ angle the cockpit rudder control must be held to achieve the maximum permissible rudder deflection defined by (a).

(d) With the aeroplane yawed to the static equilibrium $\beta$ angle, it is assumed that the cockpit rudder control is suddenly returned to neutral.

#### 7.1.2.3 Pitching conditions

CS 25.255 specifies the required positive and negative recovery load factors that must be available following a runaway failure of the trim horizontal stabilizer. Chakraborty et al. (2015)[14] analyzed those qualifications over the Boeing 737-800 altitude-speed envelope (*Fig. 2.4.5*) to determine the most critical conditions yielding the maximum hinge moment for the elevator deflection.

#### 7.1.2.4 Spoilers conditions

CS 25.373 reviews the considerations regarding speed control devices if their are installed for use in en-route conditions. Additionally, CS 25.549 specifies that ”the loading for special devices using aero-dynamic surfaces (such as slots, slats and spoilers) must be determined from test data” (EASA, 2020 [23]).

Chakraborty et al. (2015) [14] states that the maximum hinge moment for the deployment of spoilers occurs when they are extended during an emergency descent performed at $V_D$. Their actuation load requirement (*Tab. 2.4.1*) was computed using the maximum rated tire speed since the deployment of ground-spoilers can only occur once the plane has landed [14].

### 7.1.3 Equipment, systems and installation

Any equipment or system that is installed in the aeroplane must meet the requirements specified along CS 25.1309, although jams of flight control surfaces –included in CS 25.671(c)(3)– are excepted from the requirements of CS 25.1309. CS 25.1309 is described as follows:

- Any installed system in the plane must be designed and installed so that any failure is extremely improbable and none catastrophic failure would not result from a single failure.
- There must be redundant systems of the systems the latent failures of whom would result in a catastrophic failure.
- Detailed Continuous Airworthiness directives must be described for maintenance and preventive maintenance tasks.

### 7.1.4 Power source capacity and distribution

Regarding the electrical system of a plane, CS 25.1310 specifies that it must be capable of offering the required continuous power for essential loads[2] in any probable operating situation and for probable duration, for instance after the failure of any prime mover, power converter or energy storage device [23].

### 7.1.5 System safety: EWIS

CS 25.1709 declares that the Electrical Wiring Interconnection System (EWIS) must be designed and installed so that any failure is extremely improbable and does not result from a single failure.

### 7.1.6 Control Systems

The definitions and regulations regarding the control systems of the plane are established from CS 25.671 to CS 25.705. However, the most relevant information in relation to this project is announced in CS 25.671 and summarized as follows:

- ”Each flight control system must operate with the ease, smoothness, and positiveness appropriate to its function (...) and continue to operate with these conditions when the plane is in any altitude or experiencing any flight dynamics parameter that could occur due to operating conditions” (EASA, CS 25.671(a), 2020 [23]).
- Each element of each flight control system must be designed to minimise the probability of incorrect assembly that may result into failure or malfunctioning of the system.
- The aeroplane must be shown by analysis or tests to be capable of continued safe flight and landing after any failure or jam, including:
  - Any single failure.
  - Any combination of failures shown to be very improbable.
  - Any failure or event that results in a jam of a flight control surface.

As defined in CS 25.671, ”the jam must be considered at any normally encountered position of the control surface and must be assumed to occur anywhere within the normal flight envelope and during any flight phase from take-off to landing” (EASA, CS 25.671(c)(3), 2020 [23]).

[2]Essential loads, as defined by CS 25.1310, are those required for type certification or operating rules.

## 7.2 Design requirements

The design of any plane always focus in four main considerations: weight, efficiency, safety and reliability [64], as does for any single aeronautic component.

EMA have to highlight and solve different issues before its successful application into flight control surfaces, specially under the conditions of restrictive space –inside the wing– and high power density –because weight is a critical parameter for planes design– [53]. The removal of centralized hydraulic system introduces new constraints such as redundancy of systems, effective heat transfer and overload protection [53]. As a consequence, new high requirements must be contemplated including fault tolerance, reliability and thermal management [53].

Furthermore, other requirements that are important in terms of future application of EMA systems may be listed:

- Scalability and flexibility to ensure that the applicability would be easy and that the existing equipment would not need to be redefined [59].
- Testability, to make real-time verification and monitoring easier [59].
- Reduced complexity by decomposing the main CPU in smaller distributed controls [59].
- Low maintenance costs in comparison to conventional central-hydraulic pump based actuators and EHA.
- Increased efficiency of EMA by means of PBW and the use of ”power-on-demand”.
- Tolerance to operate with a variable frequency power source.

## 7.3 Outline

On the whole, the most significant airworthiness regulations regarding the sizing of EMA are those that define the manoeuvring speeds (CS 25.335) and the flight conditions that yield the computation of the maximum hinge moments for the deflection of each specific control surface. Additionally, the design and installation of systems must meet safety, reliability and operativeness among all the flight conditions of the plane, as defined along CS25.1309, CS 25.1310, CS 25.1709 and CS 25.671 – CS 25.671 also reveals that jam of the actuator may be considered to happen in any flight condition–.

Finally, the highest design requirements of EMA include fault tolerance, reliability, overload protection and thermal management.

# Chapter 8

# System design

As previously defined along *Chap. 5*, the most feasible answer nowadays for the application of EMA to deploy flight control surfaces seems to be the application of distributed PRSM direct-PMSM driven electromechanical actuators.

*Chap. 8* will present a preliminary sizing methodology to compute the maximum hinge moment of a specific flight control surface. Besides, an introduction to the mathematical model of EMA will also be granted.

## 8.1 Sizing

Electromechanical actuators have not yet been implemented in aircraft control systems, thus it is difficult to prior define its size and power using similarity method. As explained along *Sec. 2.4*, it is extremely important to understand the flight mechanics that make up the problem being studied. This section will present a semiempirical methodology using Roskam's (2000) [58] equations, data and plots.

A computer code will be generated –see *R. Attachments C*– to calculate the maximum hinge moment for a specific control surface of a particular plane. However, an important amount of geometrical input parameters regarding each specific plane and surface are needed, which are not always easy to track down. As a consequence, this project will only use the geometrical data for Boeing 737-800's ailerons –presented in *R. Attachments D*– in order to establish a link with Chakraborty et al. (2015) [14] article to validate the code and to be in accordance with the objectives and scope of this project.

### 8.1.1 Hinge Moment along a control surface hinge line

When defining the flight mechanics of an airplane many references could be used. However, only a few give tools to compute the torque needed to deploy the control surfaces.

$$H = \frac{1}{2}\rho V^2 S\bar{c}C_H \tag{8.1}$$

*Eq. (8.1)* [43] shows the general equation to compute the hinge moment $H$ $[Nm]$ –or the torque to generate along the hinge axis–. To calculate this value in the proper way, both chord ($\bar{c}$) and planeform area ($S$) of the control surface may be used. As it can be seen, this equation also depends on $C_H$, which is the hinge moment derivative.

#### 8.1.1.1 Hinge moment derivatives

The hinge moment derivative can be expressed as follows *Eq. (8.2)* [43]:

$$C_H = C_{H_0} + C_{H_\alpha}\alpha + C_{H_\delta}\delta + C_{H_{\delta_t}}\delta_t \tag{8.2}$$

These coefficients represent the variation of hinge moment coefficient with respect to the angles of attack ($\alpha$), control surface deflection ($\delta$) and tab deflection ($\delta_t$): $\frac{\delta C_H}{\delta\theta}$ $-\theta$ to simplify the explanation–. Roskam (2000) [58] gives a semi-empirical approximation to solve this value, depending on the geometry of the wing, the geometry of the control surface and the flight conditions, i.e. air speed, air density, altitude, etc.

As explained in Roskam's (2000) [58] book, the hinge moment coefficient for a control surface can be estimated from *Eq. (8.2)*, which is separated into four different components:

- $C_{H_0}$ represents the zero-angle-of-attack, zero-control-surface-deflection and zero-tab-deflection hinge moment coefficient.
  For main surfaces whit symmetrical airfoils, $C_{H_0} = 0$. Yet, if the main surface is composed by a cambered airfoil, experimental data may be used to estimate this quantity.
- $C_{H_\alpha}$ is the three dimensional control surface hingemoment derivative due to angle of attack.
- $C_{H_\delta}$ is the three dimensional control surface hinge moment derivative due to the deflection of the control surface.
- $C_{H_{\delta_t}}$ is the three dimensional control surface hinge moment derivative due to the deflection of the control surface's tab.

With the aforementioned, the hinge moment derivatives can be calculated. As explained along Roskam's VI volume (2000) [58], most of the values depend on Reynolds number ($Re$) and Mach number ($Ma$). However, assuming that $Re$ remains almost constant during cruise greatly simplifies the calculations. Given that $Re$ depends on altitude, air-speed and a characteristic length, it may be expected to remain almost constant during cruise [8].

As a consequence, *Eq. (8.2)* may be expressed in standard form as follows [14], depending only in effective angle of attack ($\alpha_{eff}$), Mach number ($Ma$) and deflection angle ($\delta$). Furthermore, as the effect on hinge moment derivative due to tab deflection is very little, $C_{H_{\delta_t}}$ can be nullified [14]. The final equation results in *Eq. (8.3)*.

$$C_H = C_{H_0} + C_{H_\alpha}(M)\alpha_{eff} + C_{H_\delta}(M,\delta)\delta \tag{8.3}$$

*Eq. (8.3)* is based on $C_{H_0}$, which has not yet been specified. It depends on the main surface geometry, as well as on the control surface area: further analysis may be considered in order to obtain this value, as mentioned in Roskam (2000) [58]. Some wind-tunnel reviews specify the value of the hinge moment derivative depending on both $\alpha$ and $\delta$ for a given airfoil. For instance, [7, 41] shows the results for NACA rm-a7l02 and NACA rm-l5ij14, respectively, studying different control surface geometries and areas. As it is shown there, the value for $C_{H_0}$ is plotted when both angle of attack and surface deflection are equal to zero. Even so, airfoil experimental data given by other authors, such as Abbot (1959) [1] does not include hinge moment derivatives coefficients.

Since $C_{H_0}$ depends on control surface geometry and airfoil, bibliographical experimental data may be found for each single case, unless a symmetrical airfoil is used. Notwithstanding that better solutions may be found, for instance some that include computer fluid dynamics analysis, for the development of the project and the associated mathematical tool, a value of $C_{H_0} = 0$ have been set for all the given examples. This decision have been taken because $C_{H_0}$'s contribution on the final value of $C_H$ might be negligible: as explain along the aforementioned reports ([7, 41]), $C_{H_0}$ is always more than ten times smaller than the contribution of the other terms, also including the angles involved in *Eq. (8.3)*, in radian.

#### 8.1.1.2 Hinge moment

As a conclusion, the final equation, considering the things mentioned above and both *Eq. (8.1)* and *Eq. (8.3)*, for the calculation of the control surface hinge moment shall be expressed as follows:

$$H = \frac{1}{2}\rho V_{eff}^2 S\bar{c}\big[C_{H_\alpha}(M)\cdot\alpha_{eff} + C_{H_\delta}(M,\delta)\cdot\delta\big] \tag{8.4}$$

*Eq. (8.4)* will be used to obtain the required torque to be made along the hinge of an specific control surface. The torque that the actuator must produce will strongly depend on the chosen gearing [12, 43], or type of actuator (see *Sec. 5.1.2*) and in the chosen architecture (*Sec. 2.4.1*).

The angular velocity to deploy the control surface can be determined by the regulations, depending on the type of control surface and also the critical it is on the proper manoeuvrability and airworthiness of the aircraft [23]. However, *Tab. 2.4.1* will be used as a valid reference for the rate of rotation of the flight control surfaces.

In this regard, if the actuators are composed by an hydraulic system, the force to be applied along the hinge may be obtained from the pressure of the system, while the velocity would be given by the flow of the hydraulic fluid [14]. On the other hand, if those are composed by an electromechanical one, the torque will be produced by the force of a magnetic field, whereas the velocity will be directly related to the variation of an alternating-current electricity wave [27]. Although the latter may be part of another study, it is important to distinguish how this two different systems would be able to achieve the same result: a torque and an angular velocity in the hinge line of the control surface, ergo, a generated power [12]:

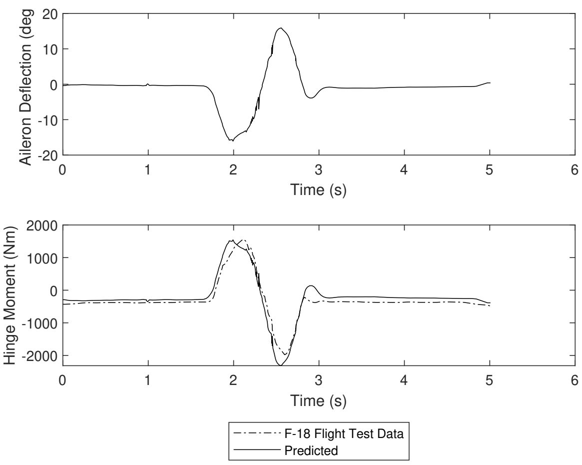
$$P = \tau\omega \tag{8.5}$$

The required power to be generated by the actuator helps to link the mechanical and electrical or hydraulic subsystems involved, an thus, is the initial tool for a successful preliminary sizing of the actuators, although the application of EMA into flight control surfaces will surely need a complex mathematical model to include all the multidisciplinary approaches involved [9].

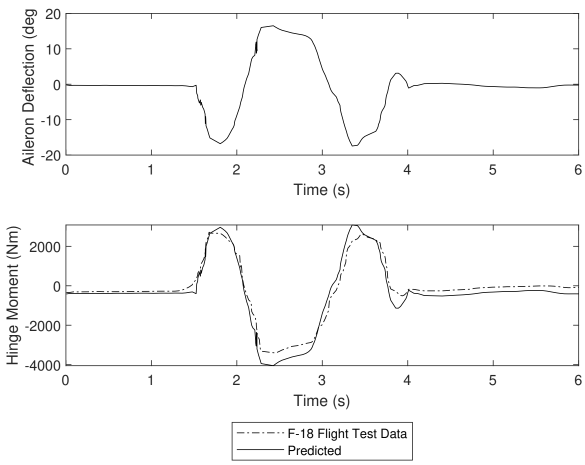
### 8.1.2 Validation

Once the code is configured (see *R. Attachments C*), it shall be validated in order to conclude whereas it is useful or not for preliminary sizing and simulation purposes. *Fig. 8.1.1* presents two different plots comparing NASA's F18 SRA flight data obtained by Navarro (1997) [47] with the results of the calculation code.

*Fig. 8.1.1a* depicts the aileron deflection with the hinge moment generated along the hinge line when an abrupt roll doublet is performed. As it is shown, the flight test data and the predicted results only differ a little: there is only a maximum relative error of $\epsilon_r$ = 0.06 between both results.




(a) Abrupt roll doublet – full stick (left): M = 0.72, h=7,620m (25,000 ft), q-bar =13.60 kPa




(b) Abrupt aileron reversal – full stick (left): M = 0.84, h=7,620m (25,000 ft), q-bar =18.29 kPa

Figure 8.1.1: Comparison of hinge moments predicted using *Sec. 8.1.1* methodology with flight test data from *Fig. 15* and *Fig. 18* of NASA's F-18 SRA [47]

*Fig. 8.1.1b* also illustrates and compares the predicted results with flight test data of F-18. In this case, an abrupt aileron reversal is carried out. The maximum relative error between both results is $\epsilon_r$ = 0.10, which may be considered as valid in a preliminary design phase [69]. The main divergence between these two curves occurs just when the hinge moment is the lowest (considering that it is a negative value), i.e. when the aileron is positively deflected at its maximum during the manoeuvre.

Those errors can be attributed to different causes, including:

- Inaccuracy in reading Roskam's (2000) [58] plotted results.
- Fault in considering some geometric parameters of the plane, so not all the needed parameters are reflected in technical manuals.
- Oversight scanning the flight test data results because they are only presented in plots.
- Error due to the simplifying assumptions made.

In spite of the little divergence ($E_r \leqslant 0.10$) between the flight test data and the predicted results, and as evidenced previously, the code (*R. Attachments C*) is successfully validated and hence, it can be used afterwards for sizing or simulating purposes.

## 8.2 EMA layout

Overall, the preferable layout for EMA for flight control surfaces would consist of a direct drive technology composed of a PMSM that drives a PRSM, as stated in *Chap. 5*. This block would be directly attached to the control surface and, thanks to the linear displacement generated by the PRSM, the control surface would deflect at the commanded position.

Furthermore, a power converter will need to be placed just before the PMSM for control purposes: this is intended to modulate the input power and convert it to the required voltage wave for the PMSM. As concluded from *Tab. 4.3.1*, matrix converters would be the most suitable converter type for aeronautic applications given that they do not need an intermediate DC phase, that a smooth control is possible and that they are appropriate for high inertia applications.

As stated along *Chap. 4*, and with the aim to reduce the complexity of systems and its associated maintenance costs, the electrical power conversion stage may be installed just before the loads, as a VSVF generation system (*Fig. 4.2.1*(c)), thus reducing the electrical power-conversion stages. However, harmonics and variable frequency tolerance of airplane systems may be further analyzed [53].

## 8.3 EMA architecture for aileron

This section studies the most suitable architecture of an EMA for the ailerons actuation including the calculation of the required loads and the health monitoring and redundancy architectures.

### 8.3.1 Required performance

The code generated in *R. Attachments C* is used initially for sizing purposes. To start, the required geometrical data of Boeing 737-800 –presented in *R. Attachments D*– is introduced to the program using a spreadsheet. The aforementioned conditions regarding the roll of the airplane (*Sec. 7.1.2.1*) are considered with the aim to obtain the maximum hinge moment that the ailerons must overcome. The rate of rotation in this case is taken from *Tab. 2.4.1* because it is conservative enough to be valid for a preliminary design phase. After the computation, the result was compared to *Tab. 2.4.1* and, as it can be appreciated from *Tab. 8.3.1*, it is verified that the calculated value does not differ from the one calculated by Chakraborty et al. (2015) [14], as the relative error between the two calculations is $\epsilon_r$=0.07.

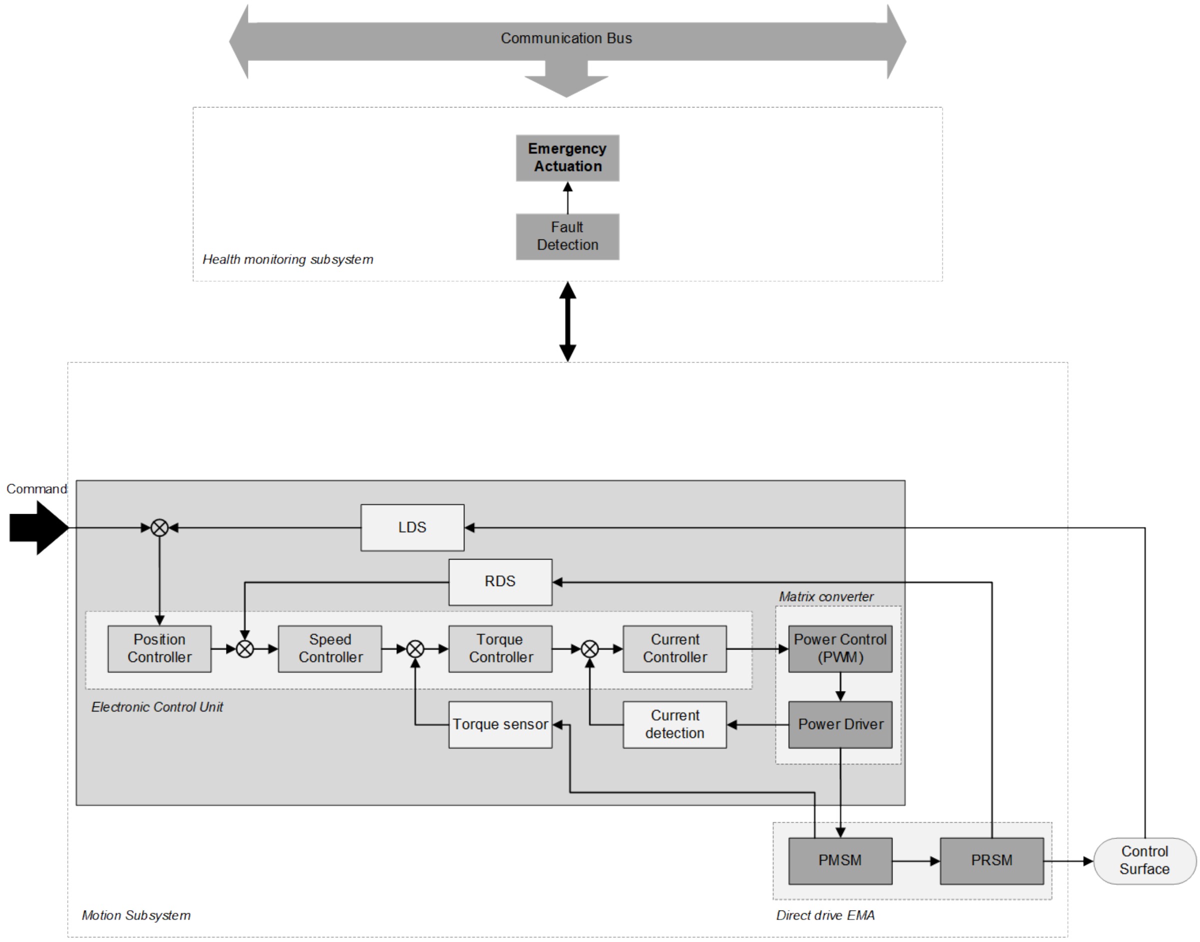



Figure 8.2.1: Layout of EMA systems: direct PRSM PMSM-driven actuator with a matrix power converter. Health monitoring systems are also included.

Table 8.3.1: Results of the computation for the actuation load of ailerons on Boeing 737-800 using *R. Attachments C* code

| Control Surface | Actuation Load | Rate | #/Aircraft |
|---|---|---|---|
| Ailerons | 5.6 $[kNm]$ | 60 [º/s] | 2 |

### 8.3.2 Architecture

The architecture of EMA systems for primary flight control surfaces is as important as their layout. As presented in *Chap. 6*, there must be means to satisfy a successful redundancy and health monitoring of the EMA installed into an aircraft, for instance using *Fig. 8.2.1* layout. To achieve so, fault-tolerant redundancy is going to be the implemented in EMA because its reduced mass and weight, the failure isolation it possesses and the high utilization level it has.

Furthermore, in order not to require big EMA and also to offer redundant systems and a more flexible aerodynamic configuration (*Sec. 5.2*), multiple actuators can be installed along the aileron and it also can be fragmented into smaller parts. Thanks to the use of this type of distributed EMA, once the pilot sends a rolling command to the control system, it would decide which configuration of the aileron if the desirable considering the flight conditions and the current health state of the actuators. Subsequently, the control system will send the appropriate deflection command to the power converters, that will power afterwards the PMSM to move the PRSM and deploy or retract the control surface accordingly.

## 8.4 Mathematical model

The use of EMA for flight control surfaces involves multiple disciplines. Consequently, a complex mathematical model is needed for simulation purposes. Many studies, such as Qiao et al. (2018) [53] or Cusidó et al. (2010) [59] state that one of the most important challenges nowadays to the successful application of EMA into flight control surfaces would be the finding of measurable physical parameters involved in the performance of EMA.

A valid mathematical model would need to include, for instance, thermal management [53], reliability [53, 57], frequency response [9], backlash, jamming [53], inertia, damping [9], harmonics and variable frequency tolerance [53], among others. Evidently, there is still a long path before the validation of a complete mathematical model to describe the functionality of EMA. Once it is performed, the design of components such as power converters, motors and screws would be much more efficient and succesful [9].

## 8.5 Outline

To sum up, a methodology to estimate the aerodynamic loads for a control surface have been developed (*R. Attachments C*) and successfully validated. The results regarding the actuation of ailerons of a Boeing 737-800 are presented in *Tab. 8.3.1*, being the actuation load of $H$ = 5.6[kNm].

Moreover, the layout of an EMA for flight control surfaces (*Fig. 8.2.1*) will include a motion subsystem and a health monitoring subsystem that will verify if the actuator is working properly or if any malfunctioning is happening. The motion subsystem will include different sensors to track down and monitor all the variables involved during the actuation. Matrix converters would be the preferable power converters and should be installed just before the electric motors. A direct drive architecture of the actuator whose includes a PMSM that drives a PRSM that is responsible for the movement of the control surface would be the most appropriate solution to be applied for the actuation of primary flight control surfaces.

Furthermore, a distributed architecture of EMA placed along the control surface would help to increase the aerodynamic performance of the plane because multiple configurations of the control surface could be arranged. This will also suppose an increase in safety given that those multiple EMA can be part of a fault-tolerant redundancy architecture.

Finally, it must be stated that the development of a complex mathematical model to simulate the performance of EMA is of major importance nowadays: this would be one of the most challenging next steps towards the use of EMA for MEA.

# Chapter 9

# Practical development

With the aim to demonstrate the functionality of EMA and perform preliminary simulations and tests, a mock-up will be designed. This chapter includes the basic information regarding the design parameters and performance of this miniature.

## 9.1 Requirements and limitations

Requirements and limitations are always the starting point of any design. Consequently, the ones related to the construction of a miniature for this project are listed below.

It is important to note that, as a consequence of the limitations, this mock-up will not be composed of a PRSM direct-PMSM driven electromechanical actuator, although the functionality and aim of the miniature is to demonstrate and perform preliminary analysis of the physical parameters regarding the design of the innovative EMA technology and its associated mathematical model.

- A flight simulator software may be used for simulating purposes.
- The hinge moment generated along the control surface hinge line must be computed and simulated.
- A positioning motor must act as an EMA and shall be correctly positioned following an input command.
- A Programmable Logic Controller (PLC) must be installed to perform the required calculations.
- Digital and analog inputs must be placed for control purposes in the even of a PLC failure.
- There is the need to monitor the variables regarding the functionality of the mock-up.

- The mock-up should be easily portable.
- The ”plug and play” concept must be applied to the miniature to expand its usage.
- The miniature will be designed with the already existing Control Techniques Variable Frequency Drive (VFD) and servomotors.

## 9.2 Design

The design of the mock-up can be separated into three main phases: analysis of the requirements and first approach, drawing lines into the functionality of the miniature; design of the mechanical assessment and its components; and finally, establishment of the electrical wiring, including power circuit, control circuit, monitoring circuit and also the associated communications between these systems.

### 9.2.1 Functionality

If the aforementioned requirements (*Sec. 9.1*) are correctly analyzed, a flow-chart can be created. *Fig. 9.2.1* presents the flow-chart that may be followed for the design of the demonstration case.

As it can be appreciated, a flight simulator software[1] will be used to simulate the physical variables during the flight of a chosen plane. Via a joystick, a positioning command will be sent to the computer, which will be translated as a control surface deflection. The flight data will continuously be sent at real-time to a PLC controller who will calculate the position of the control surface (related to the command and the associated deflection) and the equivalent hinge moment generated along the hinge line. To do so, fixed hinge moment derivatives data for a particular plane will be needed, which will result from the calculation of hinge moment derivatives (*Sec. 8.1.1*) performed in Matlab (*R. Attachments C*) and that will be stored, afterwards, into the PLC.

Once calculated both, the position and the torque, the position reference and the torque reference will issue a command to two different servo-VFD, that will activate, respectively, a positioning servomotor and a torque-generator servomotor, that are mechanically linked to each other. As the required maximum torque for the Boeing 737-800’s aileron is 5.6[kNm] (*Tab. 8.3.1*) which is inconceivable with the available components to construct the mock-up (*Sec. 9.2.2*), this quantity will be proportionally reduced following *Eq. (9.1)*.

[1]XPlane-11, developed by Laminar Research, is going to be the flight simulation software in this project because of the flexibility it offers in using flight output data

In *Eq. (9.1)*, $\tau$ and $\tau_N$ are, respectively, the torque that the motor must generate and the specific rated torque, and $H$ and $H_N$ are, respectively, the hinge moment of the control surface during the simulated flight and the maximum actuation load of the specific flight control surface –in this case the presented in *Tab. 8.3.1*–.

$$\tau = H\frac{\tau_N}{H_N} \tag{9.1}$$

In order to demonstrate that the high-speed and low-torque output of the electric motor can be easily converted to low-speed and high-torque by means of the appropriate mechanical assessment, a servoreducer will be placed just after the positioning motor, which will be mechanically linked via a pulley-belt assembly to the output of the torque-generator motor. In order to show how the simulated control surface is deflected, a little airfoil panel will be installed just after the servoreducer, with its hinge line directly conjunct to the axis of rotation of the pulley.

Additionally, a closed control loop will be set-up for the motors to control parameters such as the torque, the position or the speed. For monitoring and configuration purposes, a Human-Machine Interface (HMI) will be installed and will display the most relevant data of the performance of the motors, including the speed, the positioning, the equivalent deflection of the surface and the current magnitude.

### 9.2.2 Components

As evidenced from *Fig. 9.2.1*, the demonstration may be build-up with different components. In order to accomplish the requirement regarding the portability of the mock-up and also the limitation to use the already-in-market Control Techniques devices, the littlest available components from Control Techniques are considered in the design.

A support structure must be framed so as to hold all the components together and help the transportation of this demonstration case. In addition, to satisfy security standards, protective structures will be installed to protect the user from the movable parts: the motors, the pulleys and also the airfoil panel. The drawings of all the single components, the sub-assemblies and the assembly of the whole mock-up are gathered together in *Chap. 1* of *Drawings*.

The mock-up is designed to be suitable for different simulation purposes apart from the initial one: actuators for flight control surfaces. This is why the control surface hardware of the mock-up is intended to be constructed with rapid-manufacturing technology, including additive 3D printing and hot-wire foam cutting.

All the parts are listed inside *Chap. 1* of *Drawings*, while all required components, materials and manufacturing costs are registered in *Budget*. The rendering of the demonstration case is presented in *Fig. 9.2.2* and its technical data is summarized in *Technical Sheets*.

### 9.2.3 Electrical wiring

The electrical wiring of the mock-up (hosted in *Chap. 2* of *Drawings*) have to be designed regarding power, control and communications, thus different voltage standards may be contemplated (*Tab. 4.1.1*). As the demonstration case is intended to be portable and used into common spaces, this must be powered with 220V AC voltage standard at $f = 50[Hz]$. Control connections will use 24V DC voltage source for digital connections, while analog ones, associated to a potentiometer, will use 10V DC as a reference.

The chosen VFD from Control Techniques includes an On-Board PLC of 16kB of memory, which will be enough for the program that is going to compute and simulate the positioning and the required hinge moment of the control surface. Consequently, and in regard to the communications, Ethernet protocol will be used to link the flight simulator software –on the computer and at real time– to the PLC On-Board of the VFD. In addition, also the Ethernet protocol will be held between the VFD and the HMI. The constant hinge moment derivatives for each specific plane and control surface may be saved to the PLC by means of the Ethernet protocol every time that the user desires to simulate a different control surface or another plane. However, this project will only consider the data for Boeing 737-800's ailerons, as aforementioned (*Sec. 8.1.1*) and presented in *R. Attachments D*.

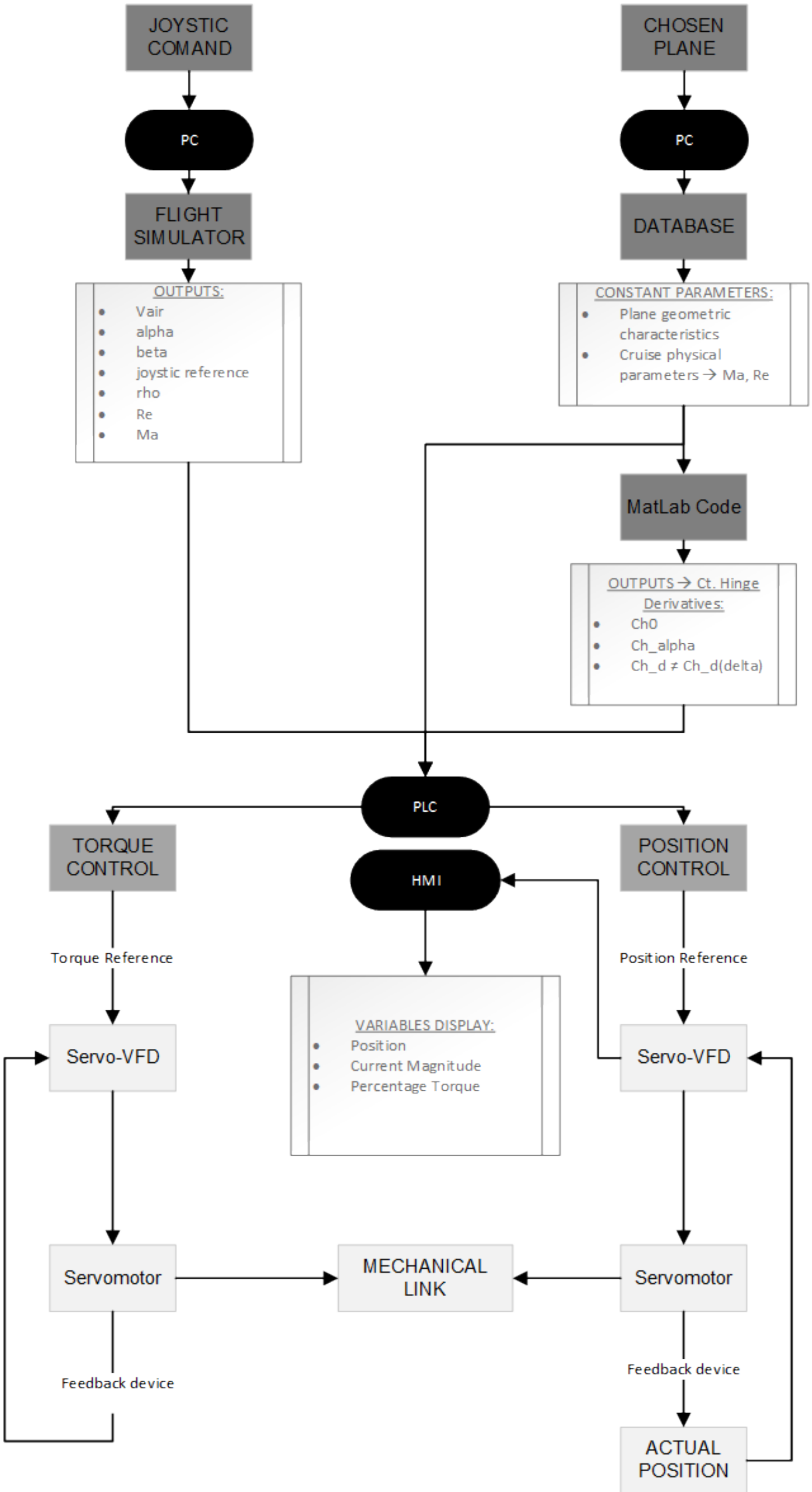



Figure 9.2.1: Flow chart for the construction of the demonstration case

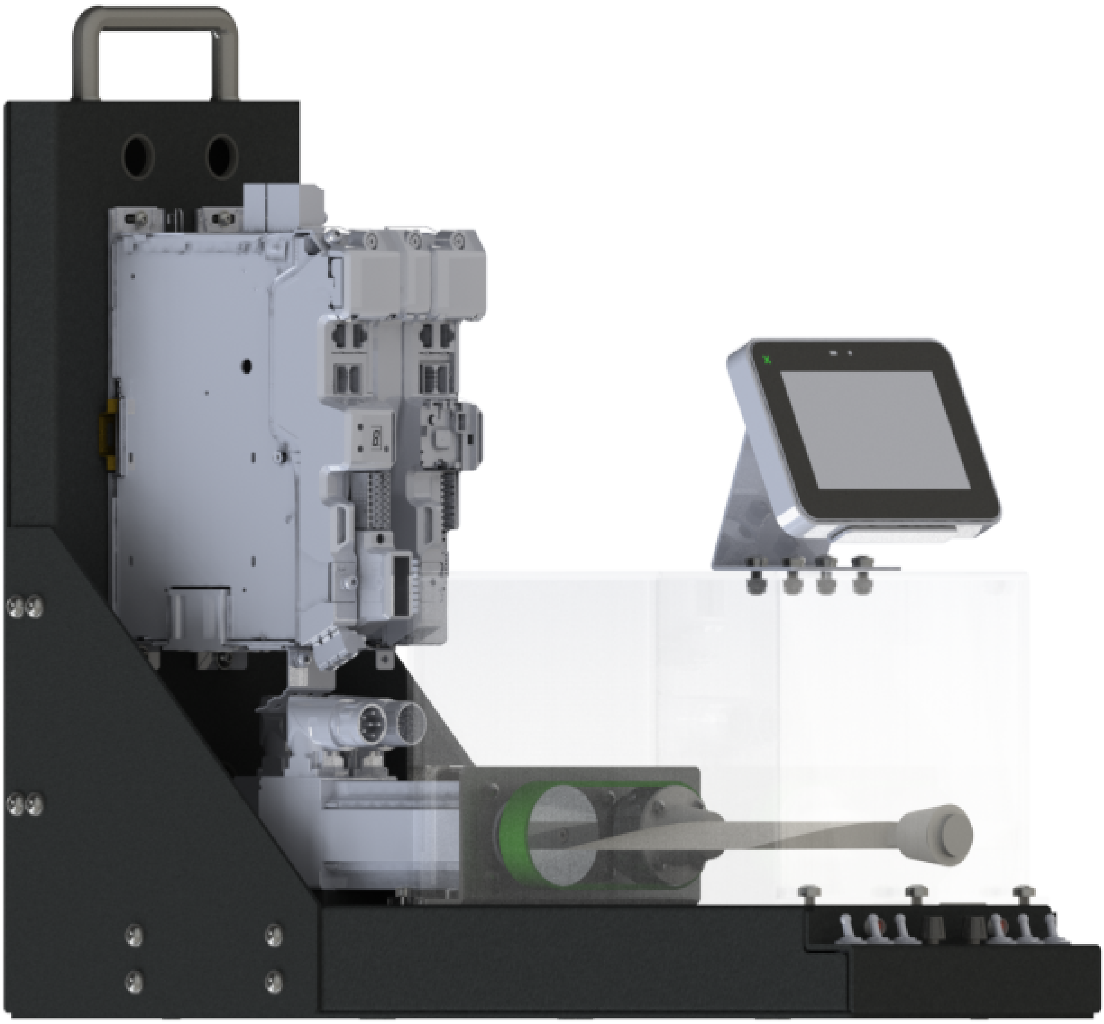


Figure 9.2.2: Render of the demo to be constructed

## 9.3 Outline

In conclusion, it can be stated that the mock-up have been designed considering all the initial requirements and limitations. Consequently, it will be used to simulate, monitor and study the physical parameters regarding the performance of an electric servomotor for the deployment of a flight control surface. In addition, as it will also do the job for demonstration purposes, it is designed to be easily portable and powered with mains electricity. Technical and practical documentation of the mock-up have been generated, including *Drawings*, *Technical Sheets*, *Budget* and also a rendering of the design (*Fig. 9.2.2*).

# Chapter 10

# Next steps

*Chap. 10* will present and summarize the most relevant future steps towards MEA and the application of EMA on flight control surfaces. Besides, the procedure to be followed in regard of EASA regulations in order to certify, manufacture and install the proposed system in a real aircraft will be recapped. *R. Attachments E* presents a Gaant chart and a network diagram of the next steps to be performed.

## 10.1 Future steps towards MEA and the application of EMA for flight control surfaces actuation

As stated along *Chap. 1*, the cut-out on greenhouse effect gases coming from transportation industry cannot be conceived without a reconsideration on the power sources. Consequently, MEA will be a concept to deem when thinking off using greener technologies. However, to expand its usage, the electric standards on planes will need to be rethought as the electrical power on-board increases.

Another important step is to improve the generation stage of electrical power in order to increase the efficiency and reduce purchase and maintenance costs: nowadays, the use of huge power converters between the generator and the loads seems to be the previous step to the installation of distributed power converters, which will represent that the electrical system of the plane and its associated loads will operate with a VSVF power source.

To sum up, MEA is the future of aviation although there is still a lot to do before the effective electrification of airplanes, specially tasks related to the design and certification processes established by the Airworthiness Authorities. Despite the successful use of electric-based actuators on B787 and Airbus A380 (*Tab. 3.1.1*), it will still take years until MEA new generation of aeroplanes represents the majority of planes in service.

## 10.2 Further works of the project

The purpose of this project is to recap studies regarding EMA for flight control surfaces and to study its applicability, to draw-lines towards future EMA architectures and layout and to give tools for future simulation tasks. The aim of the future works to be done in this project is to convey EMA towards an efficient, reliable and fault tolerant application for primary flight control surfaces. The further works to be carried out are presented below:

- Analysis and simulation with the demo-case to find quantifiable magnitudes related to the performance of EMA.
- Description of a mathematical model that shows and considers the most important parameters of EMA for flight control surfaces.
- Simulation, by means of a computational software, of this mathematical model and improvement of it.
- Analysis of the simulation model and definition of the most suitable technologies for EMA for primary flight control surfaces, regarding:
  - Flexibility and adaptability for different planes sizes.
  - Health monitoring systems and devices.
  - Fault-tolerant reliability architecture, or another one that is able to overcome jamming and runaway faults.
  - Means to reduce the probability of failure of EMA that appear nowadays (*Tab. 5.1.6*).
- Prototyping of new EMA.
- Test of the developed prototypes, specially in four main fields, that will help to better understand the various faults and operating characteristics under varied working conditions of EMA [53]:
  - Thermal management
  - Reliability
  - Fault-tolerance
  - Response
- Design of the most suitable architecture.

Once the prototyping stage begins, Airworthiness Authorities will start to play their role. Below, the typical Certification Process (CP) that may be followed before the certification of any aeronautic component, previous to its production and installation, is explained.

### 10.2.1 Certification Process (CP)

The CP is carried out in conjunction with the Airworthiness Authorities with the aim to satisfy all the airworthiness regulations regarding the design, certification and production of any aeronautic component. Consequently, 5-10 years are the usual times between the initial design ideas to the Type Certificate (TC) approval prior to the production of the component [57]. The process involves Design Organisation Approvals (DOA), the approval of the TC during the CP and the Production Organisation Approvals (POA) before producing any component, as explained in EASA Part 21 (EASA, 2019 [24]) and summarized thereupon.

#### 10.2.1.1 Design Organisation Approvals (DOA)

The first stage before being able to design any aeronautic component is to hold the authorization by EASA in doing it. For this reason, all the process –summarized in *Fig. 10.2.1*– starts with the required formalities for being authorized as a Design Organisation, as reflected in Part 21 – Subpart D of EASA (2019) [24]: it starts with the submission of the application request for DOA which is preceded by an interview with the DOA Team Leader assigned by EASA. After that, the Design Organisation Handbook (DOH) is presented, verified by the DOA Team Leader and corrected prior to its approval.

The DOH includes information and presentation of the organisation, the scope of its work –describing which devices or components will be designed by the DOA– and also the management staff and the available human resources. In regard to the management staff, there must be a Head of Design Organisation (HDO), responsible of the technical aspects developed into the DOA; a Head of Office of Airworthiness (HOA) –that can be the same person who is in charge of HDO–, at the helm of the airworthiness regulations and to interact with the Airworthiness Authorities; and finally a Head of Independent System Monitoring (HISM) in the driving seat of the quality and audit completion of the DOA.

#### 10.2.1.2 Type Certificate TC

Once the DOA is approved, the Design Organisation can start designing the component. Just as soon as the Design Organisation considers that the project is in a fair degree of maturity, it is presented to EASA in order to set the Certification Basis and to appoint the EASA's employees in charge of this CP, as evidences *Fig. 10.2.1*. The next step is to establish the

Certification Program, which includes the expected calendar and the Means of Compliance (MOC)s. The MOC describes how the design of the component will be evaluated in order to satisfy the standards and regulations: engineering evaluation, tests or inspections. After that, the Compliance Checklist would be created, reflecting the CS involved in the design of the component and its associated MOCs. A Compliance Document that includes the calculations, test data or inspection forms is generated for each particular entry of the Compliance Checklist to demonstrate, by means of the required MOC, that the component is in accordance with each specific CS. Finally, EASA carefully examines all this documentation included in the Compliance Demonstration and, if it is in accordance with the airworthiness regulations, EASA emits a Technical Closure and an Issue of Approval that are prior to the emission of the TC.

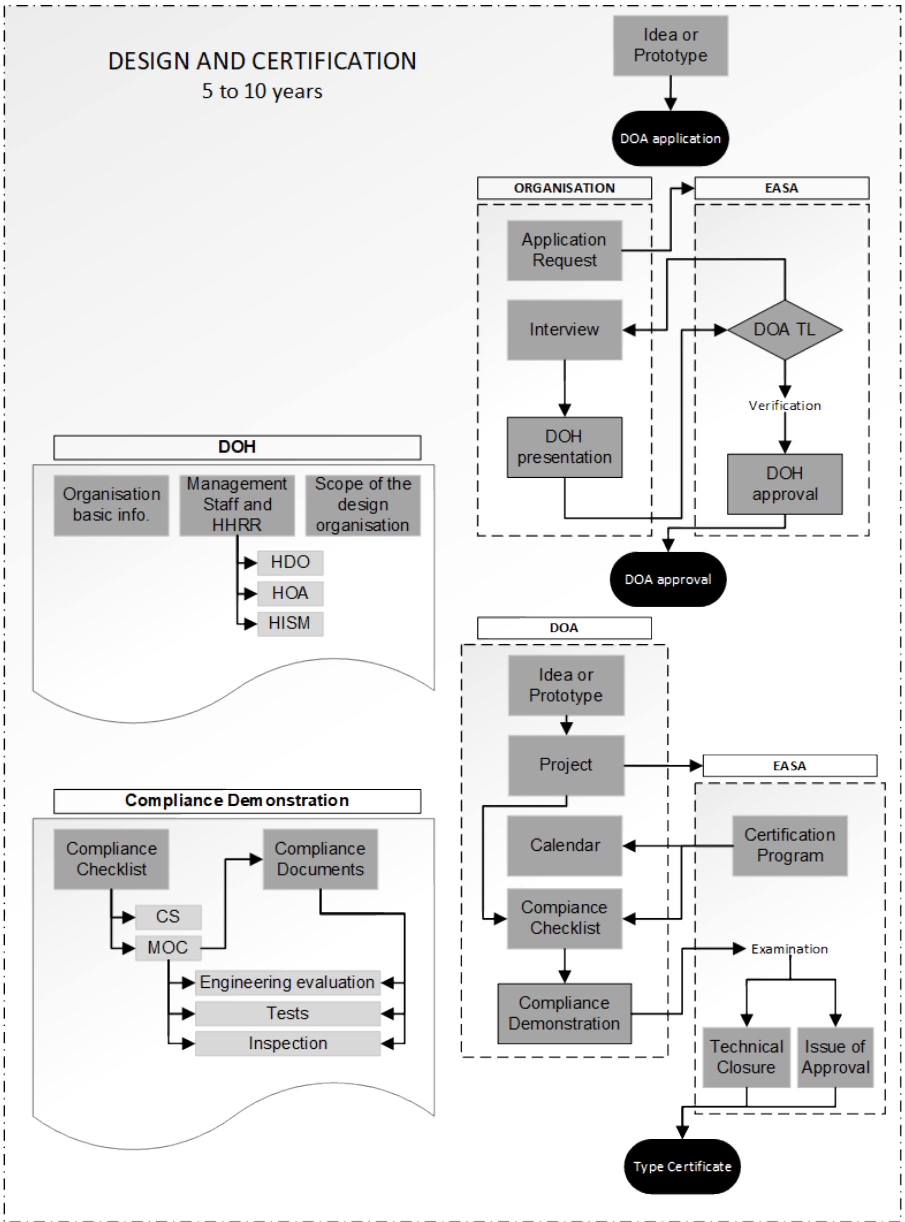



Figure 10.2.1: DOA approval and certification process

The TC is the technical document that reflects all the technical data of any aeronautic component or a plane. This reflects that this component has been approved by the competent Authority and that satisfies all the required standards to be safely installed into an aircraft.

### 10.2.2 Production Organisation Approval (POA)

Part 21–Subpart G of EASA (2019) [24] describes the process to be authorized as a POA – reviewed in *Fig. 10.2.2*–. It starts with an approved design (e.g. TC or Minor Change), that the organisation wants to produce[1]. An application request is sent for the production approval to EASA that may include the appropriate scope of the components to be produced. After that, EASA delegates the responsibility to the correspondent regional aeronautic authority, for instance Agencia Estatal de Seguridad Aérea (AESA) in Spain.

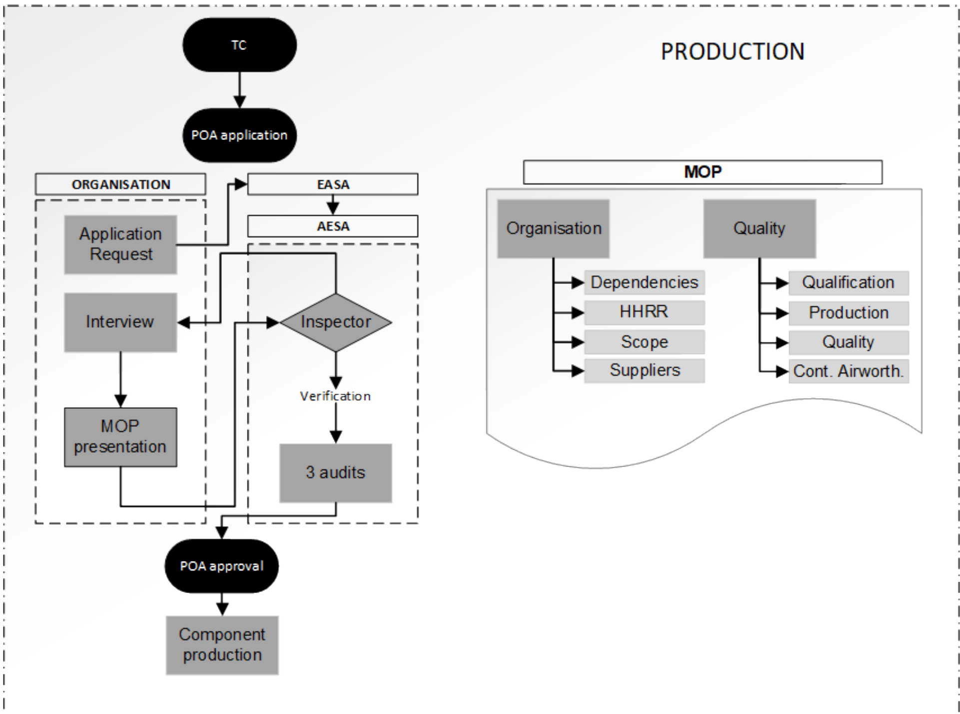



Figure 10.2.2: POA approval

AESA is then in charge of the approval process and starts with an interview with the production organisation that is performed by an AESA inspector [62]. After that, the production organisation presents the Memoria de Organización de Producción (MOP), which is the document that reflects the characteristics of the organisation. The MOP is separated in two main parts: *Organisation* and *Quality*. In the first section, the organisation is described: its dependencies, its personnel and management staff, the scope of the production organisation and also its suppliers. On the contrary, the second section summarizes the procedures that the

[1] A design organisation independent from the organisation that wants to produce the component can be the Type Certificate holder of the component

organisation is performing to ensure that the quality of the component is in accordance with the standards, reflected in EASA's Form-1 [24], including the qualification of its employees, the production procedures involved, the quality of the checking fixtures and also reflecting the continued airworthiness requirements that apply.

Once the MOP is presented, it is evaluated and checked by AESA and then corrected appropriately by the organisation. After that, a verification process is carried out by AESA, thus three audits are executed in the dependencies of the production organisation to check if it is organized and is making its job in accordance with the MOP. If all these requirements are fulfilled, the POA is emitted and hence, the production organisation can start to effectively produce the component. As evidenced, this is such a large process that takes between one and two years of bureaucracy [62].

## 10.3 Outline

On the whole, there is still a load to do in regard to MEA and the successful application of EMA for primary flight control surfaces. One of the most challenging further works is to find quantifiable physical parameters of the performance of EMA in order to establish a mathematical model that will permit the simulation and improvement of EMA systems.

Moreover, once the simulations are performed, a testing and prototyping stage will begin, which must be in accordance with EASA. Consequently, the design institutions in charge of EMA development must be authorized as DOA by EASA. DOA can then start a Certification Process for the designed EMA systems that will lead, if it fits the airworthiness regulations, to the emission of the TC. The TC certifies that the component have been designed in accordance with the airworthiness regulations and that its installation into any aircraft is safe. Prior to produce the component, the organisation must be authorized by EASA as a POA, which give the organisation permission to start producing the component that has been already certified accordingly to the specific TC.

Overall, this bureaucracy may take from six to twelve years since the prototyping stage to the effective production of the developed EMA systems to be installed in aeroplane's flight control surfaces.

# Chapter 11

# Environmental Impact

This chapter summarizes the most important features of this project regarding the environmental impact. Different concepts may be considered: from the composition of this project to the real implementation of EMA for flight control surfaces.

## 11.1 Study

The environmental costs associated to the development of the project are directly related to the consumption of electrical power from the network to power a computer and screen during all the 600h of work involved in the project. In addition, an Ethernet connection was required for the research of useful information, which, apart from being electrically powered, generates electronic waves that may interfere with the environment. Finally, all the report of this project was printed twice for revision purposes: black ink, paper and electricity was consumed in the process.

## 11.2 Mock-up

The design of the mock-up has considered the use of new materials, even so recycled ones could be used because of the reduced size of it, such as the aluminum and steel sheets, the power inlet, the switches or even the plastic protector (refer to *Drawings*).

Nevertheless, this miniature is designed for an extended use for simulation and demonstration purposes. Further, the structure and electrical wiring of it could be reused for the demonstration of another technology. Truly, at the end of its useful life, it may be processed following the wast-processing standards because most of the materials are reusable or recyclable.

## 11.3 Electromechancial Actuators for More Electric Airfcraft

As stated along *Chap. 1*, MEA will be of extreme importance when thinking of reducing the environmental impact of air-transportation industry. As the thrust power source of planes cannot be redefined nowadays because of the big amount of power required (*Fig. 2.1.1*), there is the need to increase the efficiency of the non-propulsive power sources. This is why MEA is important and why this study focus on the application of EMA for primary flight control surfaces. One of the most important points of EMA regarding the efficiency of plane is that they are capable of consuming power on demand. Furthermore, as EMA would replace central hydraulic pump systems, the need of using hydraulic fluid, which is difficult to recycle and requires hard maintenance tasks to prevent potential leakages, would also disappear. Evidently, one of the motivations of this project is to downscale the environmental impact of air-transportation.

## 11.4 Outline

Overall, the environmental impact has been one of the motivations of this project: the rethinking of control surface actuators will suppose a reduction on green-house effect gases emissions by increasing its efficiency thanks to EMA, while getting rid of the difficult to recycle hydraulic fluid of the conventional central-hydraulic pump based actuation systems. Moreover, the mock-up, that is intended to be used for simulation and demonstration purposes, is designed for extended useful life and recycled materials can be used for its construction; while for the development of the study only electricity have been consumed.

# Conclusions

With all things considered throughout the thesis, it can be concluded that MEA will play a fundamental role to downscale the green-house effect gases emissions of air-transportation industry. As stated along *Chap. 1*, MEA is a philosophy that is trying to evolve to an aircraft where the majority of the systems are electrically powered. Consequently, MEA cannot be comprehended without PBW systems and the redefinition of the whole electrical system into the plane: from the generation to the loads. Evidently, efficiency, maintenance, reliability and fault-tolerance are going to be the fundamental pillars for the next generation of aeroplanes.

To start with, the generation stage, that is nowadays constructed with difficult-to-maintain complex mechanical gearboxes, should be replaced with distributed power converters such as matrix converters, and hence, the systems designed and constructed for a VSVF operation. Additionally, given that the electrical power into the plane will be higher, the voltages standards must be also updated, as *Tab. 4.1.1* states. The next step would be the substitution of the central-hydraulic pump flight control surfaces based actuators with increased power-density, fault-tolerance, efficient and low-maintainability EMA.

EMA must be designed so as to improve both the electrical and mechanical performance and also increasing the reliability and fault-tolerance of the actuators. As stated in *Chap. 5*, the next generation of EMA will consist of distributed PRSM direct-PMSM driven linear actuators in conjunction with a reliable, fault-tolerance based and efficient health monitoring decentralized system, that will lead the control system of the aeroplane to prevent any potential malfunctioning an so, be in accordance with the Airworthiness Regulations.

Overall, EMA still requires improvements, specially because it consists of a multidisciplinary approach which involves different engineering specializations apart from aeronautic engineering. Consequently, mathematical modelling and simulation before its successful application to flight control surfaces is of extreme importance –the designed mock-up is intended to be useful for this purposes–. For this reason, in the years to come, the research projects will focus on finding quantifiable physical parameters to mathematically model the operation of EMA. Besides, in regard to the Airworthiness Regulations, these are tough and require lots of bureaucracy that enlarge the elapsed time between the design of a new component and its effective installation into the aircraft. Their significance, however, lies in the safety and reliability of air-transportation; hence, it is a must for every single component of a plane to be in accordance with the Airworthiness Regulations.

Redefining the power sources of our day-to-day and evolving to a greener world should be our top priorities as a society. We must be aware of the consequence of global warming and every human shall try to do its bit. The change is still possible.

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