12-2016

More Electric Systems Architecture Effect on Business Jet Performance

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MORE ELECTRIC SYSTEMS ARCHITECTURE EFFECT ON BUSINESS JET PERFORMANCE

A Thesis

Submitted to the Faculty

of

Embry-Riddle Aeronautical University

by

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In Partial Fulfillment of the

Requirements for the Degree

of

Master of Science in Aerospace Engineering

December 2016

Embry-Riddle Aeronautical University

Daytona Beach, Florida

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A Thesis prepared under the direction of the candidate's committee chairman, Dr. Luis Gonzalez, Department of Aerospace Engineering, and has been approved by the members of the thesis committee. It was submitted to the School of Graduate Studies and Research and was accepted in partial fulfillment of the requirements for the degree of Master of Science in Aerospace Engineering.

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ACKNOWLEDGMENTS

I would like to acknowledge Dassault Aviation and Dassault Falcon Jet for their financial and technical support throughout the length of this project. In particular Mr. Lionel de la Sayette, Mr. Michel Lavanant and Mr. Don Pointer.

I would also like to thank Prof. Maj Mirmirani for his support of the Dassault Design Institute.

I would also like to extend my gratitude to Mr. Matthias Emeneth, from PACE, for all the help in setting up the models in Pacelab APD and SysArc.

TABLE OF CONTENTS

LIST OF TABLES

LIST OF FIGURES

Figure Page

SYMBOLS

C_D	aerodynamic drag coefficient
C_L	aerodynamic lift coefficient
C_M	aerodynamic moment coefficient
I	current
m	mass
P	power
v	velocity
V	voltage

ABBREVIATIONS

APU	auxiliary power unit
ATA	Air Transport Association
BPR	Bypass ratio
CG	center of gravity
DC	direct current
ECS	environmental control system
EDP	Engine Driven Pumps
EHA	electro-hydrostatic actuator
EMA	electro-mechanical actuator
GPU	ground power unit
MEA	more electric aircraft/architecture
MLW	max landing weight
MRW	max ramp weight
MTOW	max takeoff weight
MZFW	max zero fuel weight
NA	not applicable
NM	nautical mile
NBAA	National Business Aviation Association
OEW	operational empty weight
RAT	ram air turbine
SFC	specific fuel consumption
SL	sea level

ABSTRACT

Mazur, Kelsey M. MSAE, Embry-Riddle Aeronautical University, December 2016.

More Electric Systems Architecture Effect on Business Jet Performance.

A more-electric systems architecture, using current state of the art components, is proposed for the business jet Dassault Falcon 2000. A model of the existing aircraft, both of its overall flying characteristics and performance and of its detail systems layout, is created using Pacelab APD© and SysArc© programs. The model is validated with respect to the aircraft published data. Then, another model of the same aircraft is created but, this time, it is for the aircraft with the new architecture and their respective performance is compared. In particular, this results in changes in mission range and the overall system's weight.

The proposed more-electric architecture replaces the hydraulic actuators for the flight controls with electro-hydrostatic and electro-mechanical actuators; the engine off-takes (bleed) for cabin pressurization and wing de-icing with electrically driven compressors and electrical heating mats, respectively.

It is ensured that the new architecture possesses the same or higher level of safety by the incorporation of redundancy and auxiliary systems such as extra batteries and a ram air turbine.

The more-electric Falcon was found to be 700 lb heavier, which is a 27% increase. The thrust specific fuel consumption at cruise was reduced 0.5% by the practical suppression of the engine bleed. However, this was not sufficient to offset the weight penalty and the aircraft range was reduced from 3,275 to 2,600 NM, a 20% negative impact.

Therefore, it is concluded that, with the current technology, a more-electric aircraft based on the exchange of current by electrical systems is not an attractive proposition. However, the constant improvement in the performance and weight to power ratios of electronic and electrical components does not preclude their implementation in the future and, if the more-electric paradigm is incorporated from the conceptual and preliminary design stages higher benefits may be possible.

1. Introduction

An aircraft can be considered a system of systems, each of which is responsible for a specific role or task. For its safe operation, the following functions need to be performed and, typically, a system can be mapped to each of them: provide an environment that supports life for the passengers (pressurization, air conditioning, etc.); autopilot, communications, navigation, etc. (avionics); provide lighting and power; flight controls; deployment and retraction of the landing gear; propulsion system (fuel, power plant, etc.), etc.

All these functions require power, which is extracted from the engine as electric power (through a gearbox driven generator), hydraulic power (from pumps also driven by the accessory gearbox) and pneumatic power, typically obtained by bleeding air off the intermediate or high pressure engine compressor stages. The choice of which means to use to achieve the function has been determined by the characteristics of the function and by the efficiency of the technology available at the time. For example, it seems natural that the pressurization be accomplished using air bled from the compressor or that the high power required for the flight controls be achieved by hydraulic pumps. This has been the traditional system architecture for the last few decades.

However, with advances in technology, more and more interest has been devoted to more electrical architectures, in which the main means of powering the systems is electrical. There are several reasons for this. One of them is "while the engine is in effect a highly optimised (sic) gas generator, there are penalties in extracting bleed air which are disproportionate when compared to the power being extracted. This becomes more acute as the bypass ratio increases: original turbofans had relatively low bypass ratios of ∼1.4 (bypass) to 1 (engine core); more recent designs ∼4:1 and next generation turbofans such as the GE GEnex and Rolls-Royce Trent 1000 are close to 10:1. Modern engines have pressure ratios of the order of 30 to 35:1 and are more sensitive to the extraction of bleed air from an increasingly smaller and much more highly tuned engine central core. The outcome is that to realise (sic) fully the benefits of emerging engine technology, a different and more efficient means of extracting power or energy for the aircraft systems becomes necessary. Efficient energy extraction for the aircraft without adversely affecting the performance of the engine core and the engine as a whole becomes an imperative reason for changing the architectures and technology utilised (sic). Figure 1.1 illustrates the differences between conventional power extraction using bleed air on the left versus a more-electric version on the right. (Moir & Seabridge, 2008)."

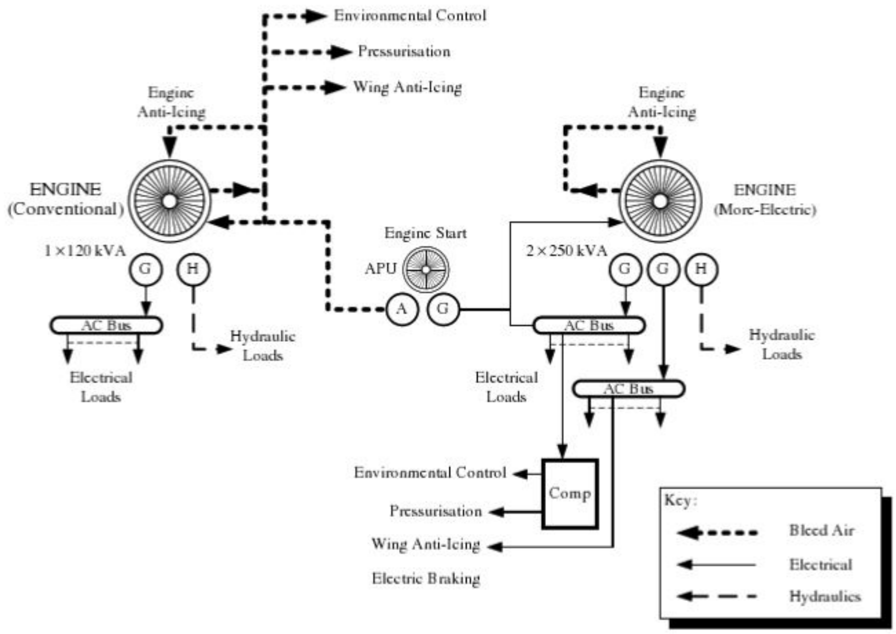


Figure 1.1: Comparison of conventional and more-electric power offtakes (Moir & Seabridge, 2008)

Another potential advantage of a MEA is that, if the hydraulic and pneumatic systems are replaced by a single electrical system, a more efficient energy usage distribution would be possible and the systems could be significantly downsized. In a conventional architecture, the overall power supply is the sum of the requirements of all the individual systems (pneumatic, hydraulic and electric). This implies the addition of their specific power peaks that, normally, do not occur simultaneously. If a single power source is used, the designers could address these different requirements with a dramatically reduced power capability, as shown in Fig. 1.2. In the conventional architecture, the average power supply is about seven times lower than the total power capability, whereas in a MEA the ratio between average power supply and power capability could be halved (Dassault, 2011).

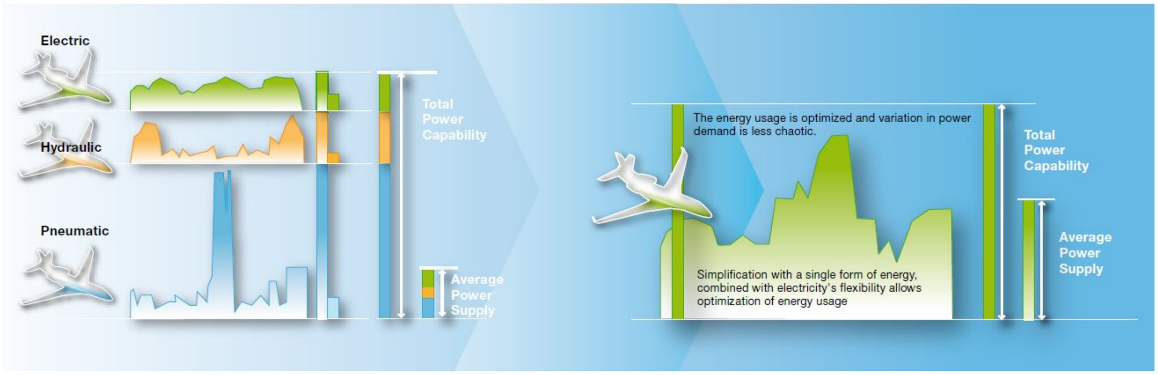


Figure 1.2: Comparison of conventional and more-electric energy management for the entire mission (Dassault, 2011)

This thesis studies the potential of the MEA applied to an existing business jet, specifically, the Dassault Falcon 2000, by proposing an architecture that replaces hydraulic and pneumatic components with current, state of the art, electrical components. The effect of this architecture change on the Falcon performance, at the aircraft level, is evaluated and compared to that of the conventional baseline and conclusions and recommendations are drawn.

1.1 Methodology

Models of the baseline aircraft, hereafter referred to as F2000, and the more electric one (F2000MEA) will be generated in the aircraft design programs Pacelab APD© and SysArc© (described in the following section). These models will be calibrated and then, the resulting differences between the two aircraft, in terms of weight and their impact on the overall fuel consumption and aircraft range will be quantified and analyzed.

First, a careful study of the current system architecture is conducted to understand it and to identify the systems or subsystems that lend themselves more easily to "electrification." It should be noted that the F2000MEA is required to have the same, or better, level of safety and reliability as the baseline.

1.2 Modeling Software

This section provides information on the software used to model the system and aircraft performance, namely Pacelab APD© and SysArc©. They are produced by PACE, a subsidiary of TXT e-solutions S.p.A., an international specialist in advanced aerospace software through its engineering division.

1.2.1 Pacelab APD©

Pacelab Aircraft Preliminary Design (APD) is an off-the-shelf software application which supports the modeling, sizing, analysis and optimization of new and derivative aircraft in the conceptual and preliminary design phases. It provides a standard set of analysis methods (mainly semi-empirical,handbook methods) for mass, high-speed and low-speed aerodynamics, flight performance and static stability which can be flexibly complemented with methods input by the user. The program also includes a comprehensive flight performance model which allows the calculation of point, segment and mission performance. Even complex, multi-leg missions can be set up easily in a flexible, graphical-interactive manner. And, it comes with a large, extensible set of predefined design tasks. By drawing on these ready-to-use calculation cases, users can automate the highly iterative task of parameter calibration by simply defining the target value and calibrating the influencing parameters (Pacelab, 2016a).

1.2.2 Pacelab SysArc©

SysArc© uses an existing APD© model and adds parametric models of system components (blocks), allowing the assessment of their impact on the overall aircraft. It contains a library that includes generic parametric models of a variety of system components such as electrical, hydraulic or pneumatic systems. Each component is accompanied by an in-depth hypertext documentation detailing design intent, parameter descriptions and suggested usage. In addition, the aircraft geometry can be subdivided into compartments, which, for example, allow aggregating the heat dissipation of the system components contained therein and feeding detailed geometric data to thermal analysis methods.

The program provides customizable schematic views where logical connections are defined by simply drawing lines between components. These are automatically translated into mathematical relations that compute the physical behavior of the architecture. It possesses an auto-routing algorithm that creates the logical connections' physical counterparts (e.g. cables, pipes or ducts) along user-definable pathways. Selecting connections of suitable diameters from an extensible library of standard sizes, the algorithm also gages their weight impact, the resulting electric or fluid power losses and other parameters. Sizing with flight conditions and failure modes to establish the limiting conditions for systems sizing, Pacelab SysArc analyzes combinations of user-definable flight phases and failure scenarios. Critical cases and limiting values are identified by monitoring and ranking key parameters such as power consumption, size or weight (Pacelab, 2016b).

2. Literature Review

2.1 General Aircraft Systems

Current aircraft systems can be divided into three categories: electric, hydraulic, and pneumatic. Typically, lighting, avionics, computers and cabin amenities are electrically powered. The electricity is produced by generators that take mechanical power from the engine through the accessory gearbox. The power is then distributed throughout the aircraft and is controlled using buses and switches. Emergency power is supplied by batteries and, in some aircraft, by a ram air turbine (RAT). Breakers or other power controllers are used to protect the system. Depending on the type of electric circuit, inverters, transformers and different means of energy storage may be used. Both AC and DC current are employed and each requires a different system architecture (Moir & Seabridge, 2008). Figure 2.1 shows a simplified electric systems architecture.

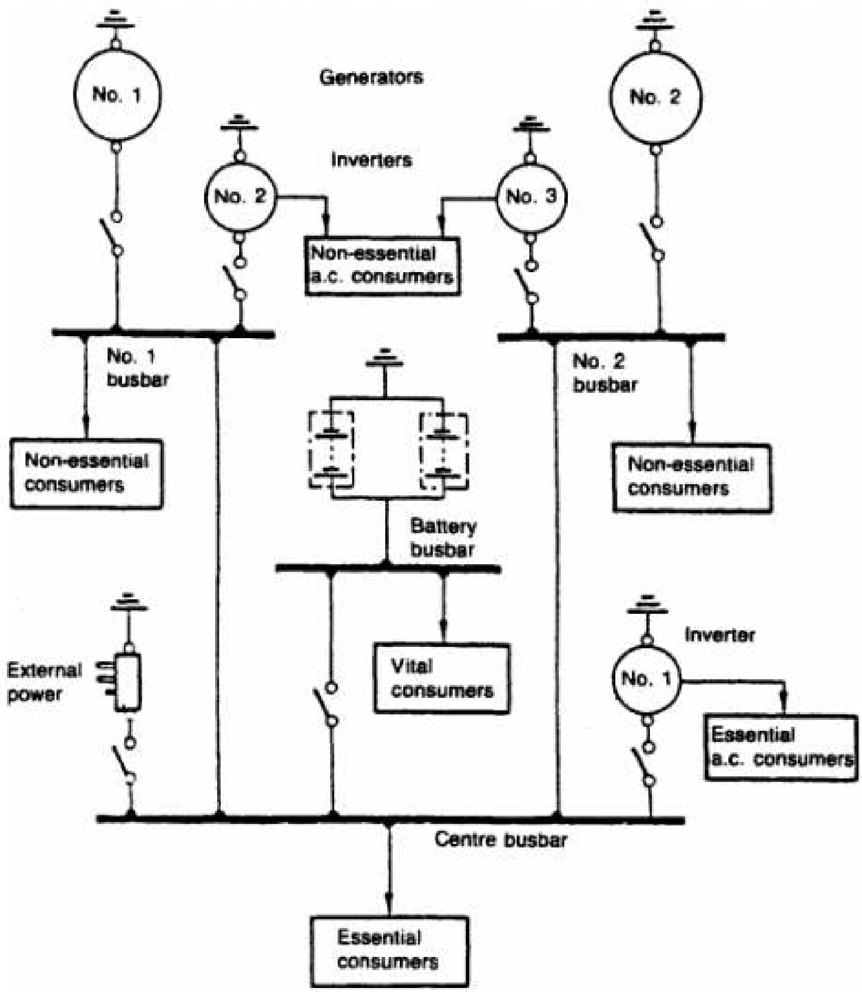


Figure 2.1: Basic electric DC system architecture (Moir & Seabridge, 2008)

As aircraft have increased in size and speed, for augmentation for the flight controls has become necessary and, therefore, the hydraulic system has become more and more critical. "Hydraulic power was seen as an efficient means of transferring power from small low energy movements in the cockpit to high energy demands in the aircraft." (Moir & Seabridge, 2008) Hydraulic power is mainly used to actuate the flight controls, the landing gear, and for braking. A typical hydraulic system is composed of a pump, a reservoir, an actuator, an accumulator, a heat exchanger, and a filter. The pump is powered by the accessory gearbox, attached to the engine (Moir & Seabridge, 2008). Figure 2.2 shows a simplified hydraulic system set up.

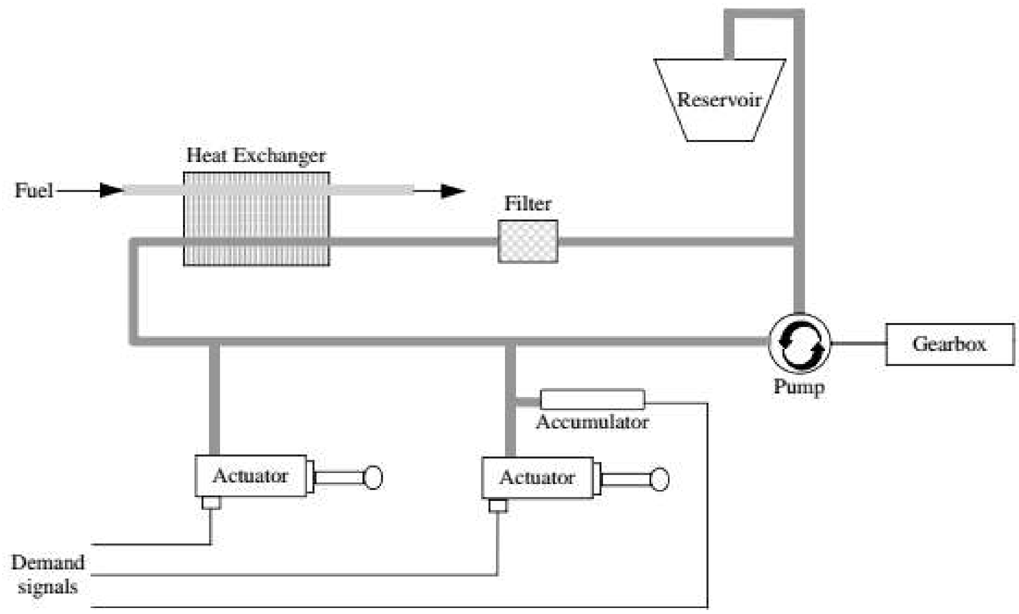


Figure 2.2: Basic hydraulic system architecture (Moir & Seabridge, 2008)

The pneumatic system takes bleed air from the intermediate or high pressure stages of the engine for use in the anti- or de-icing of the wing and engine cowlings. Bleed air is also used for cabin pressurization and for the air conditioning. Figure 2.3 shows a schematic of a pneumatic system.

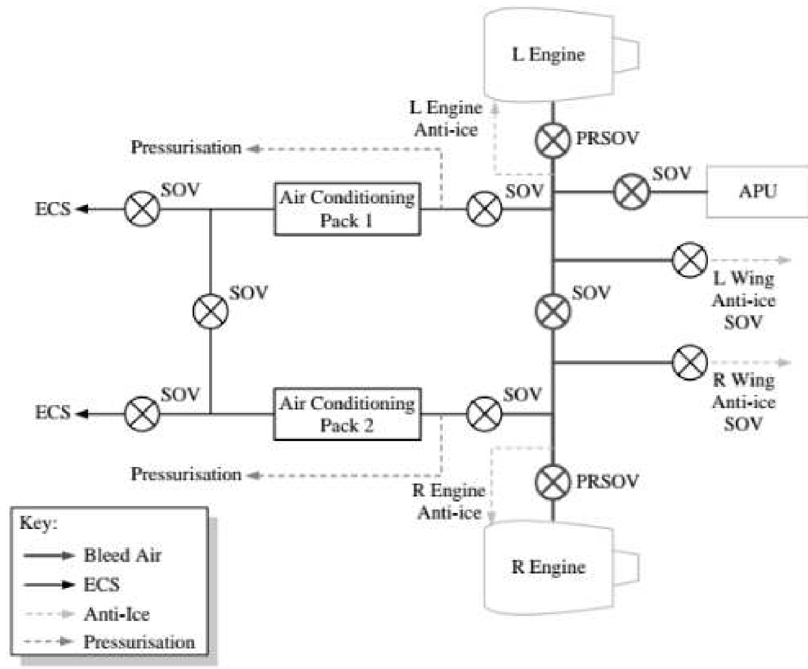


Figure 2.3: Basic pneumatic system architecture (Moir & Seabridge, 2008)

2.2 More Electric Aircraft

In an MEA the pneumatic and hydraulic systems are replaced by electric components. The engine off-takes for the pneumatic system are removed and their function is instead carried out by electrically driven compressors, for example. Likewise, electromechanical actuators are used instead of hydraulic ones (Rosero & Romeral, 2007).

The aircraft industry is focused on the development of more environmentally friendly aircraft with ever improving performance. The advances in power electronics suggest that a more electric architecture could be an avenue for fulfilling these goals. By optimizing the energy consumption, an MEA it would be possible to reduce the aircraft weight and to eliminate most of the engine off-takes that decrease its efficiency, therefore resulting in lower fuel burn and its associated emissions. Currently each system is sized to handle the maximum load. These peaks occur at different phases of the flight. If all the systems are electrically powered, the new system can be downsized because now we do not have the addition of the individual peaks but the actual power required for each flight stage (Fig. 1.2). Other expected advantages of an MEA are: lower maintenance costs, more reliability and, therefore, less delays from unexpected maintenance and repair operations, and lower weights (Rosero & Romeral, 2007).

The concept of an MEA has been around for decades. For many years now, aircraft have used computers in the actuation loop of the flight control surfaces ("Fly-By-Wire"). This could be considered an important step in the "electrification" of the flight controls (Naayagi, 2013). "Since the early 1990s, research into aircraft power system technologies has advanced with the aim of reducing or eliminating centralized hydraulics aboard aircraft and replacing them with electrical power." (Rosero & Romeral, 2007) In the 1990s, the US Air Force began an initiative to ultimately create a fully electric aircraft. It is considered that, for most applications, the engine cannot be replaced by an electric motor in the short term but the implementation of electrical systems would represent an intermediate step in that direction (Cao & Atkinson, 2012).

However, at the present time, with the current technology, it is not clear whether the promised benefits could be achieved. Some electrical components are currently heavier than their hydraulic or pneumatic counter parts (Faleiro, 2005) and the additional weight that would be necessary for backup power would be substantial (Naayagi, 2013). The main advantage of an MEA comes from the consolidation of the system. Therefore, partial substitution of subsystems by electrical ones may not result in any significant gain and may well prove to be detrimental to the overall aircraft performance (Jones, 2002).

Recent aircraft programs, such as the Boeing 787, have increased the amount of electrification. For example, the 787 is a completely bleedless aircraft, with de-icing and the environmental control system powered electrically. The electrical compressors, for cabin pressurization, require a substantial amount of additional power which, in turn required more batteries, with improved performance, and the entire electrical system now should be capable of handling 1 MW (Karimi, 2007).

For the purposes of this work, the selection of what systems to electrify is based on the individual component efficiencies. Hydraulic actuators can be replaced either by electromechanical actuators (EMA) or electro-hydrostatic actuators (EHA). Conventional hydraulic actuators are continually pressurized, independently of whether there is any demand or not. For a large part of the flight, actuator demands are minimal. Therefore this is a wasteful approach as lost energy ultimately results in higher energy off-take from the engine and hence higher fuel consumption. The EHA is more efficient in that the actuator only draws power when a control demand is sought and for the remainder of the flight the actuator is quiescent (Fig. 2.4). The EHA accomplishes this by using the three-phase AC power to drive electronics which in turn drive a variable speed pump together with a constant displacement hydraulic pump. EHAs are currently applied across a range of aircraft and Unmanned Air Vehicle (UAV) developments. The Airbus A380 and Lockheed Martin F-35 Lightning II both use EHAs in the flight control system (Moir & Seabridge, 2008).

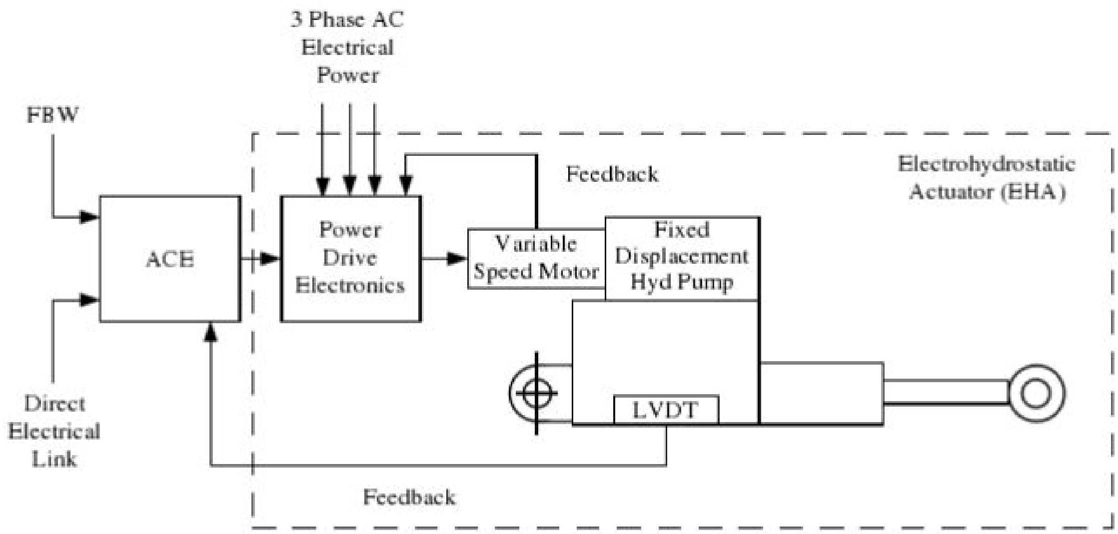


Figure 2.4: Electro-hydrostatic actuator (EHA) (Moir & Seabridge, 2008)

The EMA replaces the electrical signaling and power actuation of the EHA with an electric motor and gearbox assembly that applies the force to move the ram. EMAs have been used on aircraft for many years for such uses as trim and door actuation; however the power, motive force and response times have been less than required for flight control actuation. The three main technology advancements that have improved the EMA to the point where it may be viable for flight control applications are: the use of rare earth magnetic materials in 270 VDC motors; high power solid-state switching devices; and microprocessors for lightweight control of the actuator motor (Moir & Seabridge, 2008).

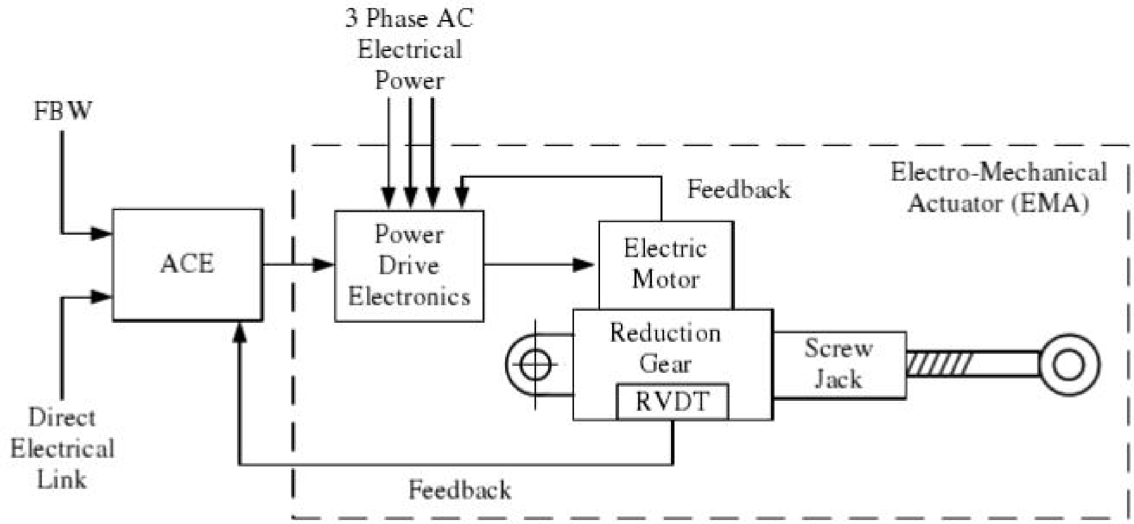


Figure 2.5: Electro-mechanical actuator (EMA) (Moir & Seabridge, 2008)

As the EHA is the more-electric replacement for linear actuators so the EMA is the more-electric version of the screwjack actuator as shown in Figure 2.5. Both concepts are identical except that the power drive electronics drive a brushless DC motor operating a reduction gear that applies rotary motion allowing the jack ram to extend or retract to satisfy input demands. EMAs are therefore used to power the horizontal stabilizer trim on civil aircraft, together with flap and slat drives. A major concern regarding the EMA is the actuator jamming. This has prevented the incorporation of this type of actuators as primary flight controls on conventional aircraft (Moir & Seabridge, 2008). EMA have shown some issues with regards to mechanical wear and there is some concern regarding the leaking of EHAs (Anonymous, 2001). However, EHAs are more established and have been used on secondary flight controls for the A380 and G650 (Derrien, 2012). Work is ongoing to improve their reliability.

Presently, the brakes are typically actuated hydraulically (Tarter, 1991). They offer the potential or regenerative power. Electric brakes have been used in testing on Bombardier aircraft but, at the time of this writing, they have not been used in production (Kirby, 2008).

Alternative methods for de-icing, not using bleed air, have been under development for some time. For example, the Sonic Pulse Electro-Expulsive Deicer (SPEED), which uses electromagnetic actuators along the wing leading edge that apply impulsive loads to the aircraft skin causing the ice to be loosed from the surface. This method has problems with structural fatigue and electromagnetic interference, so it has not been widely adopted. Electro-Expulsive Separation System (EESS) utilizes electromagnetic forces. A boot is places over the leading edge with conductors inside. When a sensor indicates ice formation, the current creates a repulsive force expanding the boot, shattering the ice. It tends to use less energy than thermal deicing. Electro-Mechanical Expulsion Deicing System (EMEDS) utilizes an electrical pulse that changes the wing leading edge shape, loosening the ice. Electrical heating using graphite has been explored. Because it can heat up and cool down rapidly, it can loosen the bond between the ice and the structure at, potentially, lower energy expenditures. Other technologies that have been investigated are shape memory alloys that change leading edge geometry or cause vibrations (Goraj, n.d.). De-icing heater mats are fairly well established now and are used, for example, on the Boeing 787 (Karimi, 2007).

Fuel Cells have been considered as alternatives to the Auxiliary Power Unit (APU) (Spencer, 2013). A fuel cell is a device that converts chemical energy from a fuel, typically hydrogen, into electricity through a chemical reaction of the positively charged ions with oxygen or another oxidizing agent. Fuel cells are different from batteries in that they require a continuous source of fuel and oxygen to sustain the chemical reaction, whereas in a battery the chemicals in the battery react with each other to generate an electromotive force. Fuel cells can produce electricity continuously for as long as these inputs are supplied. Compared to the APU, a gas turbine, they would have a low environmental impact since they require no combustion (Spencer, 2013). Fuel cells are potentially more effective at supplying electrical power than batteries or an engine (Whyatt & Chick, 2012). Fuel cells are currently under development and are not at the technology readiness level desired for the study in this thesis.

Chakraborty has an extensive and in depth study of the impact of using a more electric architecture on clean sheet designs (Chakraborty, 2014). This thesis, though, is focused on the implementation of such an architecture on an existing aircraft.

2.3 Falcon 2000 Systems Description

This section is taken directly from the Falcon 2000 Technical Specifications (Dassault, 2004). Only the systems relevant to this thesis are presented.

2.3.1 Hydraulic System

The Falcon 2000 is equipped with two independent hydraulic systems. The Left Hand side (LH) system is supplied through one accumulator by two self-regulating pumps; one driven by the LH engine, the other by the Right Hand side (RH) engine. The RH system is supplied by a self-regulating pump driven by the RH engine and by a DC electric standby pump. One accumulator is installed on the system. On each hydraulic system the fluid is stored in a bleed air pressurized reservoir located in the rear service bay. The RH side reservoir has a special compartment which is used to feed, through the standby pump, the rudder and elevator number 2 barrels, independently of the other RH hydraulic equipment. An electrical solenoid valve, normally closed in flight, allows to connect, if necessary, the stand-by pump to the whole RH hydraulic system. The various hydraulic functions are connected as described in Table 2.1. Note that the ARTHUR unit is a flight-control artificial-feel-adjustment system. A schematic of the hydraulic system is shown in Fig. 2.6.

Table 2.1: Falcon 2000 Hydraulic System

	LH system	RH system
Servo Actuator		
- elevator	Barrel n. 1	Barrel n. 2
- rudder	Barrel n. 1	Barrel n. 2
- aileron	Barrel n. 1	Barrel n. 2
Outboard slats		
-manual control	X	
-automatic extension	X	X
-emergency extension		X
Flaps		X
airbrakes		X
Landing gear	X	
Wheel braking		
- normal	X	X
- emergency	X	X
- park		X
Nose-wheel steering	X	
LH Thrust reverser	X	
RH Thrust reverser	X	
Arthur UNIT		
- elevator	X	

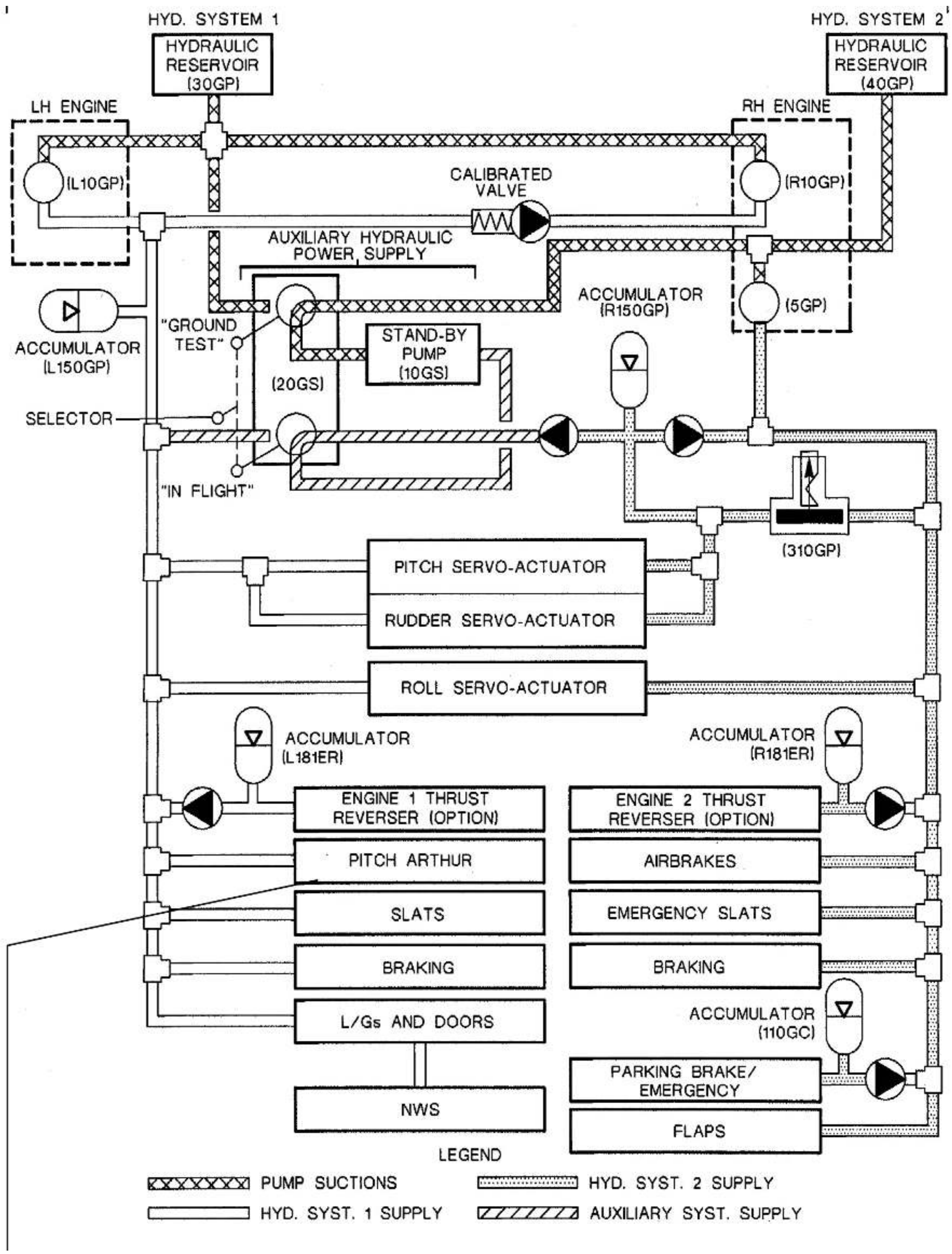


Figure 2.6: Falcon 2000 hydraulic system.

2.3.2 Flight Controls

Primary flight controls

The elevator, rudder and ailerons are hydraulically operated by dual barrel servo-actuators. The first barrel of each servo-actuator is powered by the LH hydraulic system and the second barrel is powered separately by the RH hydraulic system. The loss of one hydraulic system has no effect on the handling characteristics of the aircraft. The airplane can be flown manually in the event that both hydraulic systems fail, in a reduced flight envelope.

Rudder and aileron trim is provided by adjusting the neutral point of the artificial feel system, associated with the control linkage, connecting the servo-actuators to the pilot/copilot controls. The trim system is operated by means of electric control switches on the pedestal in the cockpit.

The horizontal stabilizer actuator is powered by "normal" and "emergency" controls located respectively on the control wheels and cockpit pedestal. The "normal" control is powered through a cross-relay box which allows to cut off a permanent control given by a horizontal stabilizer actuator control switch, using the other horizontal stabilizer actuator control switch. Rudder authority is certified for operation with crosswind up to 35 kts. A yaw damper unit is installed in series on the rudder control linkage.

The following figures illustrate the primary flight controls.

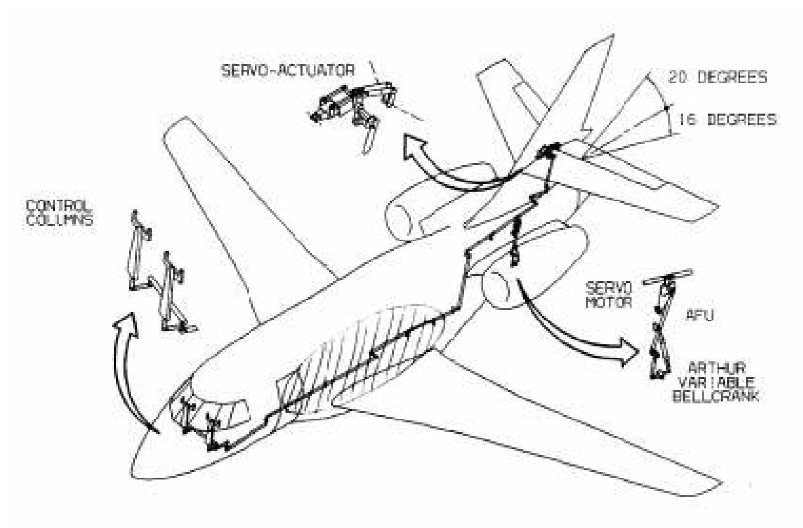


Figure 2.7: Pitch control.

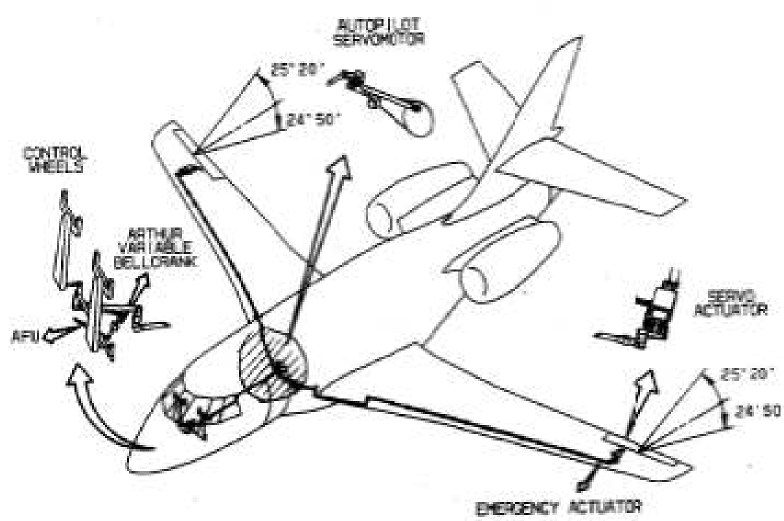


Figure 2.8: Roll control.

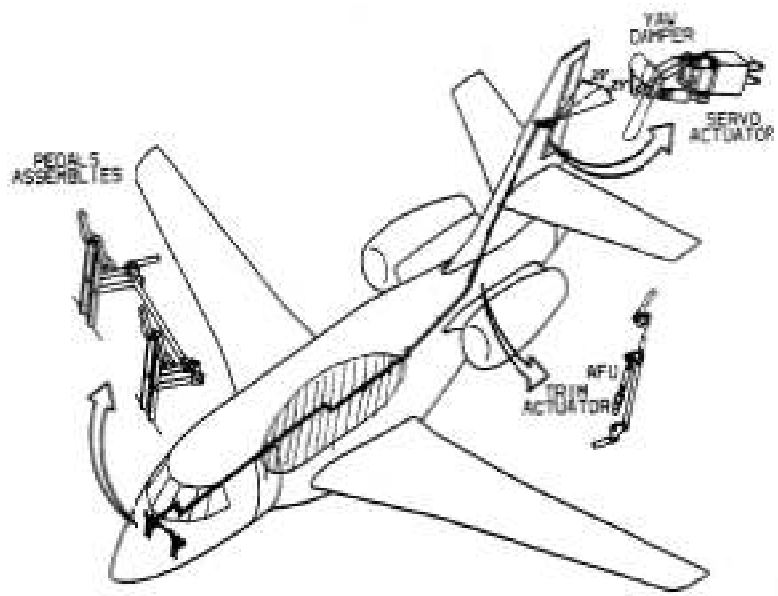


Figure 2.9: Rudder.

Secondary flight controls

The six airbrake sections are actuated hydraulically by a control lever on the center pedestal. Two-step airbrake extension is provided: the first step controls the medium sections and the second step controls the inboard and outboard sections (Fig. 2.10).

The trailing edge flaps are operated from a four-position control lever on the center pedestal. This lever operates the screw actuators through a hydraulic motor. A flap position indicator is provided. In the event of flap asymmetry, a warning light comes on and flap movement is automatically stopped (Fig. 2.11).

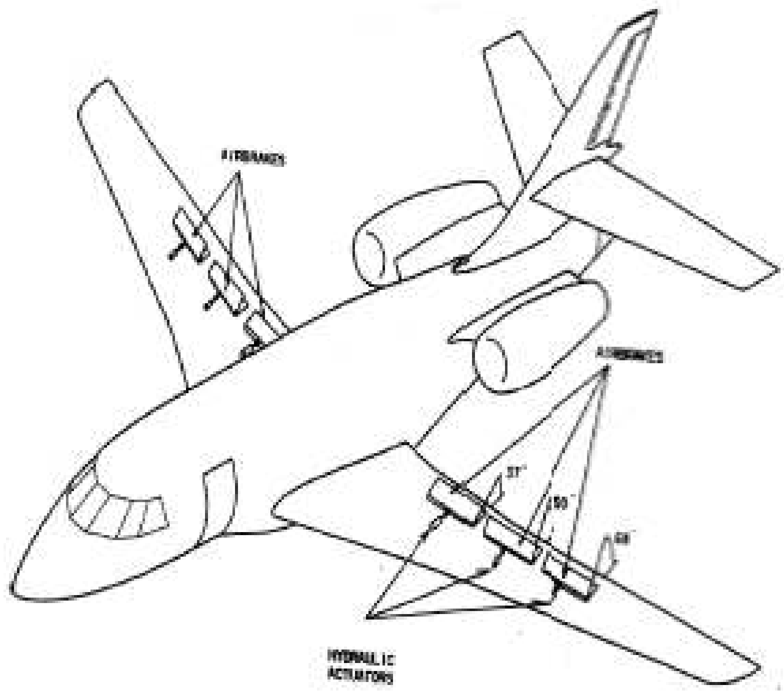


Figure 2.10: Airbrakes.

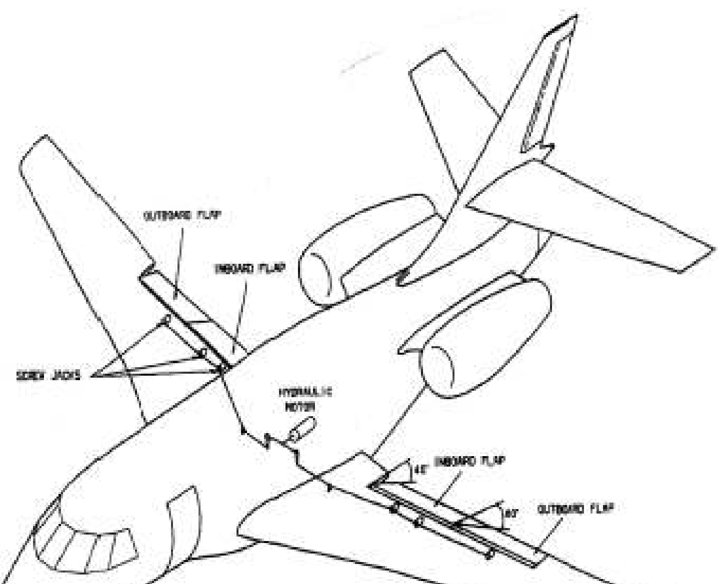


Figure 2.11: Trailing edge flaps.

The outboard leading edge slats are controlled automatically with the flaps and move to the fully extended position as soon as the control lever is moved out of the "clean" position. In addition, the outboard slats can extend automatically as a result of control signals issued by the two angle-of-attack sensors to improve the stall characteristics of the airplane in clean configuration (Fig. 2.12).

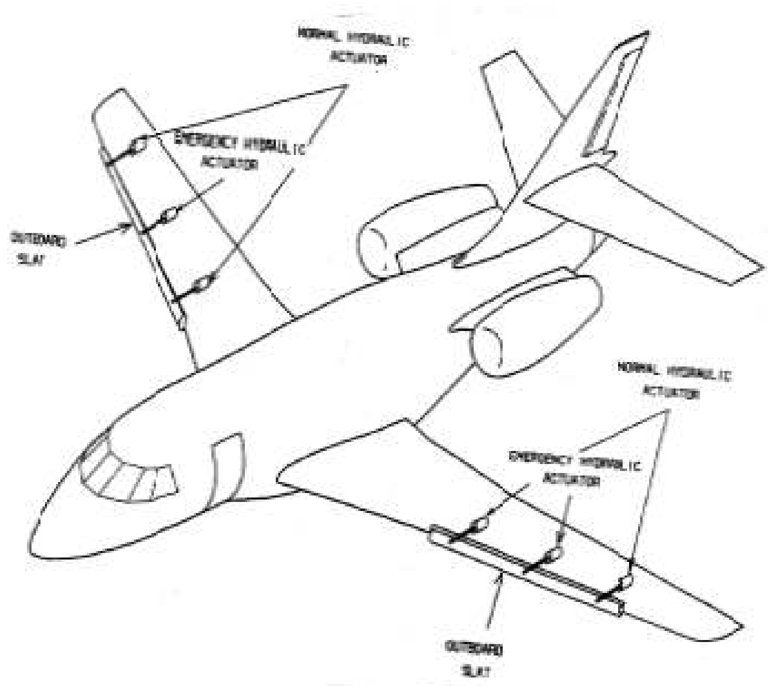


Figure 2.12: Leading edge slats.

Full flap extension or retraction does not exceed 25 seconds at normal operating speeds.

Air conditioning and pressurization systems

Conditioned air is mainly supplied to the cabin. It is obtained from air bled from one low pressure port of each engine or from the APU. There are two supply systems (with crossfeed capability); cold air, generated by a two-wheel boot-trap cooling unit, is delivered by both supply lines. This cold air is also used for cooling the avionics behind the instrument panel. The system incorporates a high pressure (HP) water separator with, upstream, a condenser that improves the efficiency of the HP separator. Air supplied to the cabin is free from disagreeable or hazardous concentrations of gases or vapors.

Cabin air is recirculated by use of a turbine jet pump. Therefore, the total air conditioning engine bleed flow does not exceed 15 kg/min.

In the whole flight envelope, the pressurization system is capable of maintaining the cabin up to the rated pressure differential of 9.0 psi (620 hPa) for all flight altitudes below 47,000 ft (cabin pressure altitude of 7,650 ft for a flight altitude of 47,000 ft). With one engine operative, pressurization is maintained up to the corresponding maximum flight altitude.

After take off the cabin rate of climb does not exceed 1000 ft/min.

Two cabin air exhaust control valves are installed at the rear part of the baggage compartment to provide normal control of cabin pressure from an automatic pressure controller in the cockpit. A manual control is also provided, as well as pressure vacuum relief, overpressure relief and altitude limitation.

Fuel and hydraulic tank pressurization is independent from cabin pressurization. The nose cone compartment is semi-pressurized and air-cooled for satisfactory operation of the avionics.

All of the anti-icing system can operate without causing objectionable variations in cabin pressure, even with one engine inoperative insofar the minimum power engine settings are applied.

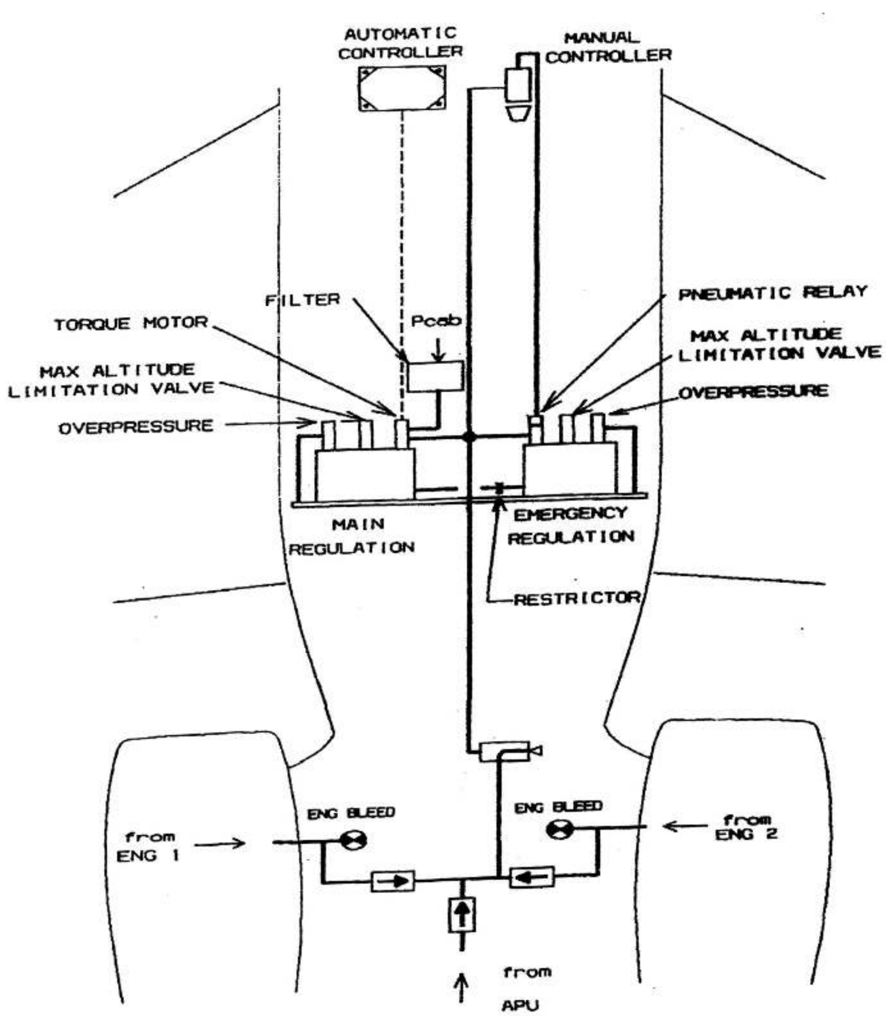


Figure 2.13: Pressurization system.

Temperature control

Automatic passenger cabin and cockpit temperature controllers with manual control capability are installed in the cockpit.

- Heating: Temperature of 21±2°C (70±3°F) (dry bulb) whenever heating is required with 2 crew members and 2 passengers.

- Cooling: Whenever cooling is required, the following temperatures can be obtained, with 2 crew members and 8 passengers:

Table 2.2: Temperature control limits

Altitude, temperature humidity	Max cabin temperature ±2° C
Sea level, 40°C, 18.6 g/kg	24°C (75° F)
10,000 ft 22°C, 18.6 g/kg	21°C (70° F)
20,000 ft and above	21°C (70° F)

For crew and passenger comfort and elimination of odors, the system is capable of delivering fresh air at a minimum flow rate of 4 kg/min (9 lb/min) in normal cruise condition.

The cabin temperature will not exceed 26±2°C (80±4° F) in normal cruise in the event of a bootstrap failure.

- Pressurization of nose cone: The nose cone compartment is provided with a ventilation system on the ground and at low altitude. The nose cone is slightly pressurized in flight.

Ice and rain protection

The aircraft is certified with an ice protection system designed to allow safe flight through continuous maximum and intermittent maximum icing conditions as required for certification; this condition is met with one engine inoperative. Anti-icing ducts are insulated in the vicinity of the fuel tanks.

Low pressure engine bleed air is used for thermal anti-icing of the wing leading edges and for the exchangers air inlet.

Tail surfaces are not anti-iced.

The engine air inlets are anti-iced using high pressure bleed.

The CFE 738 engine complies with the ice protection requirements of JAR E and FAR Part 33.

Electrical system

The electrical system is a 28 volt DC system. DC power is generated by two 12 kW rectified alternators driven by the 2 engines and one 9 kW DC generator driven by the APU when operating. A 36 ampere-hour nickel-cadmium battery is provided.

A 28 volt ground power receptacle with over-voltage protection is installed on the right side, aft of the wing, to permit power to be supplied from an external power source for routine servicing purposes and APU starting, in the event that the battery is completely discharged.

The two rectified alternators installed on the 2 engines supply power to three bus bars (LH , RH and essential), through line contactors operated by switches installed on the overhead panel in the cockpit. These bus bars can be tied or untied by switches installed on the overhead panel. A static voltage regulator and an over-voltage protector are provided for each machine.

The battery is connected directly to a battery bus and to the essential bus through a line contactor operated by a battery switch. The DC generator driven by the APU is connected to the essential bus through a line contactor. The ground power is connected to the right bus through a line contactor.

Space provision is provided for a second battery and for its associated control inside the RH main electrical box.

The electrical systems and associated controls are so designed as to permit continuation of instrument flying in the event of power system failure, electrical fire, or any other similar emergency.

An emergency battery is provided for operation of the stand-by horizon.

All electrical system wiring is protected by appropriately calibrated circuit breakers installed in a circuit breaker panel in the cockpit within reach of the pilots. Space provision for spare circuit breakers is provided.

3. Aircraft Model in APD©

The Dassault Falcon 2000 (F2000) was modeled in APD© to serve as the starting point and baseline for analysis and comparison for the performance changes of the more electric Falcon (F2000MEA). This model was calibrated with respect to available published data of the real aircraft, which is presented in the following paragraphs. Most of the information proceeds from the Falcon 2000 Technical Description Manual (Dassault, 2004).

It is recognized that the model will not be a perfect representation of the actual aircraft, something that is beyond the scope of the present study. However, since the final comparison will be given in terms of deltas between the MEA and the baseline, and both will be based on the same model, this process was considered adequate.

The Falcon 2000 is a long range business jet, for 8-10 passengers with their luggage and crew. The aircraft is shown in Fig. 3.1 and the most important dimensions are presented in Table 3.1. It should be noted that the Falcon 2000 EX is identical to the 2000 with the exception of the engines.

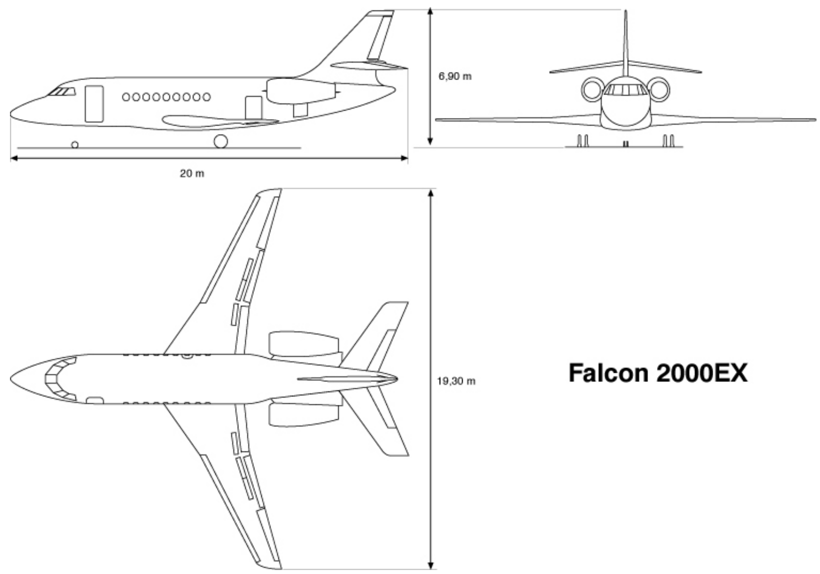


Figure 3.1: Falcon 2000 EX 3-view dragwing.

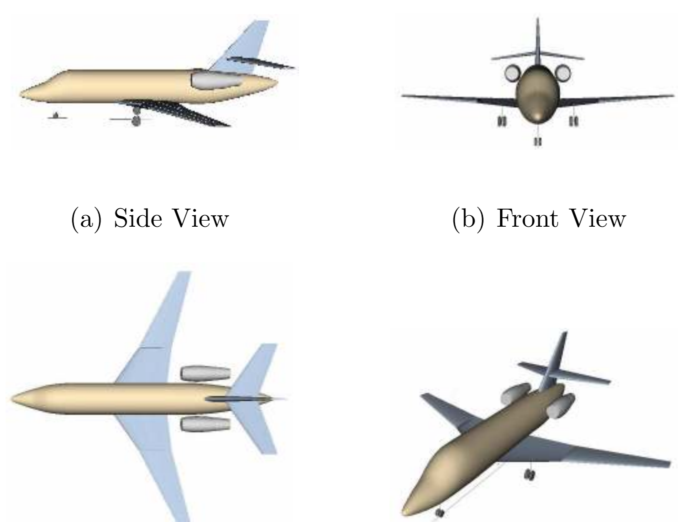
Table 3.1: Falcon 2000 dimensions.

Wing aspect ratio	7.66
Mean aerodynamic chord	9.48 ft
Wing root chord	13.38 ft
Wing tip chord	3.67 ft
Fuselage length	64.03 ft
Diameter	8.20 ft
Horizontal tail span	24.39 ft
Fin area	91.49 sqft
Landing gear wheelbase	24.26 ft

The aerodynamic characteristics of the aircraft were not available. Therefore, the model in APD© and SysArc© used an existing polar for a similar aircraft, modified with corrected factors, applied iteratively until the range, speed and fuel burn of the model matched the F2000 performance data.

The Falcon 2000 uses two CFE738 turbofan engines, with a sea level static thrust of 5,918 lbs each and weighing 1,325 lbs. The actual engine deck was not available either and, therefore, the generic engine data included in APD© was used, corrected with a calibration factor matching the engines performance given in the Falcon 2000 Series Systems Description (Dassault, 2008).

The final APD© model is presented in the following pages. The geometrical representation can be seen in Fig. 3.2 and the aircraft weights (actual and the resulting model values) are included in Table 3.2.



(a) Side View (b) Front View

(c) Top View (d) Isometric

Figure 3.2: APD Model Geometry

Table 3.2: Modeled and Actual Weights.

	APD (lb)	**Actual (lb)**
MTOW	35,800	35,800
MZFW	28,739	28,660
MRW	36,000	36,000
MLW	33,000	33,000
OEW	19,887	19,890
Max. Fuel	12,115	12,155
Single Dry Engine Weight	1,307	1,325

As already mentioned, calibration factors were applied to the aerodynamics and engine data inputs to create a model that had the same performance, i.e., the same payload-range diagram as the Falcon 2000.

The aircraft has a nominal range of 3,000 NM with a payload of 1,600 lb, at a speed of Mach 0.8. The blue line shows the range found by the model when a payload of 1,600 lbs is used. Figure 3.3 shows the payload-range diagram for the F2000. The blue line represents the calculated range for 1,600 lb of payload. The result is within 3% of the published data.

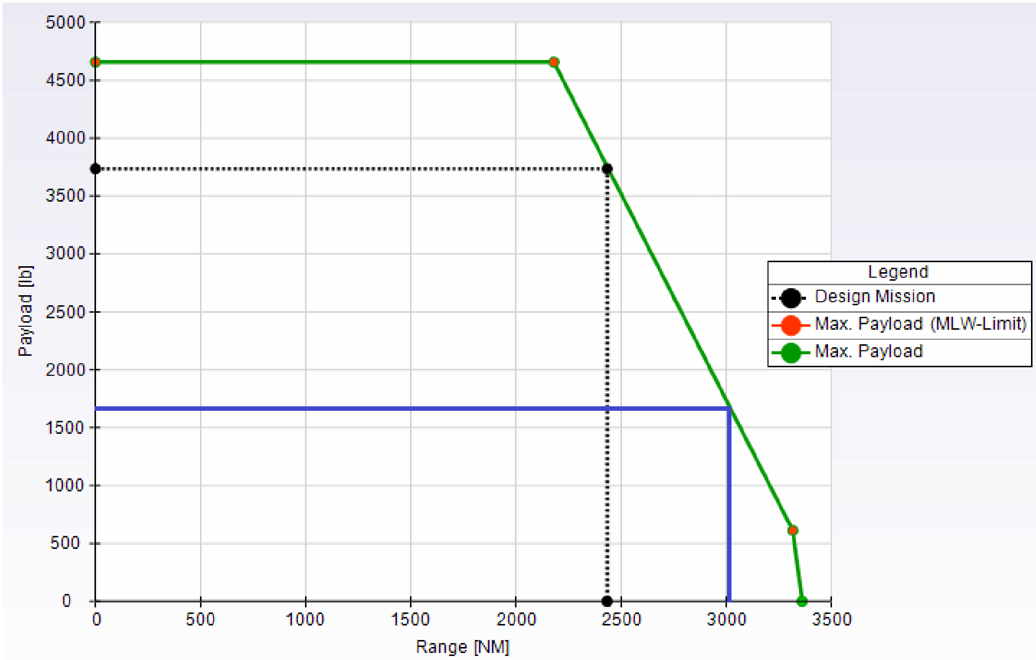


Figure 3.3: Payload-range diagram for Falcon 2000.

As shown in Figure 3.3 the range found by the model is approximately 3,100 NM; this means the percent error is 3%, demonstrating that the model is representative of the Falcon 2000's performance.

The engine model used is presented in Fig. 3.4. It is based on a generic engine deck for turbofans in APD©. It was calibrated to match thrust and specific fuel consumptions at two discrete points: sea level static and at 40,000 ft and Mach 0.8. At this condition the engine produces 1,310 lb of thrust (Dassault, 2004).

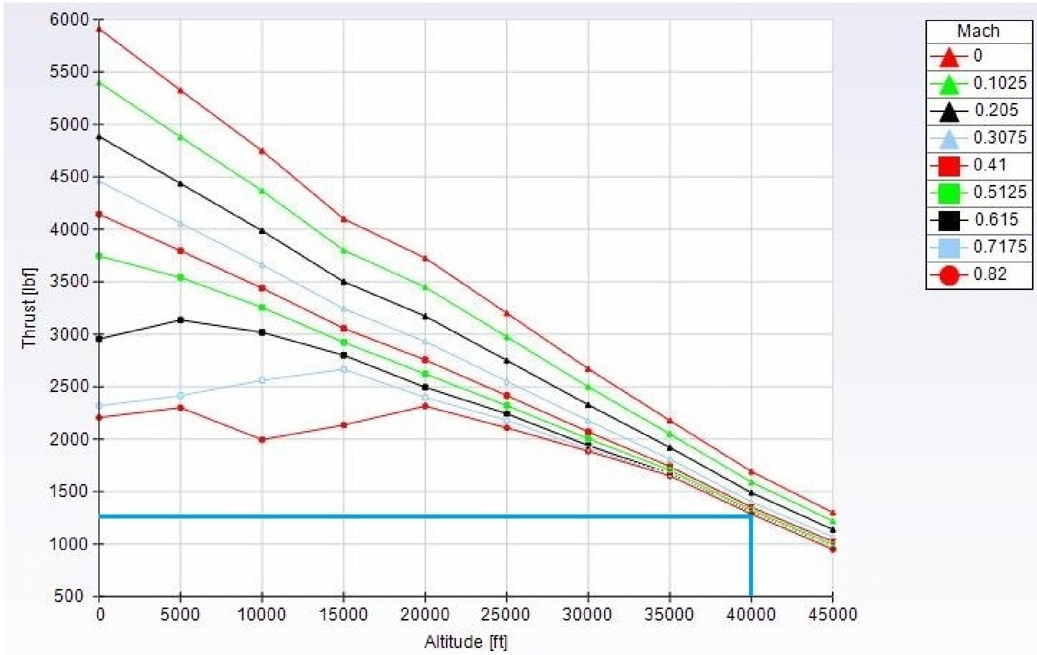


Figure 3.4: Engine thrust at cruise altitude.

4. Aircraft Systems Model in SysArc©

A model of the current Falcon 2000 was created in SysArc©. Three systems were included: electrical, pneumatic, and hydraulic. Information on the performance and characteristics of each system was obtained from the Falcon 2000 Series System Description (Dassault, 2008). The performance of the individual components of the electrical system was calibrated using an electrical load analysis provided by Dassault. The weights of some of those components was provided by Dassault on a private communication.

Figure 4.1 shows the placement of components and their "logical" connections in the APD© aircraft model. This shows which source provides power to which component, how power is distributed, and what the final usage of the power is.

The color helps to identify the components: compartments are red, sources are orange, electrical, hydraulic and pneumatic components are yellow, green and blue, respectively. There are two separate volumes in the fuselage that are pressurized: the cockpit and cabin.

The orange areas on the wing and tails are the primary flight controls: ailerons, rudder and elevator, whereas the green ones are the secondary flight controls: flaps, slats and air-brakes.

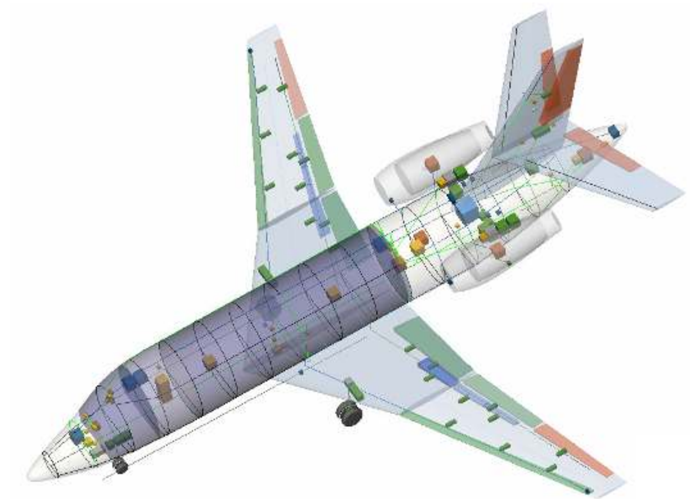


Figure 4.1: Baseline F2000 SysArc© model.

The diagram corresponding to the above model is displayed in Fig. 4.2. Each block represents a system component. This picture serves to give a global picture of the present systems architecture. More detailed views will be presented in subsequent sections.

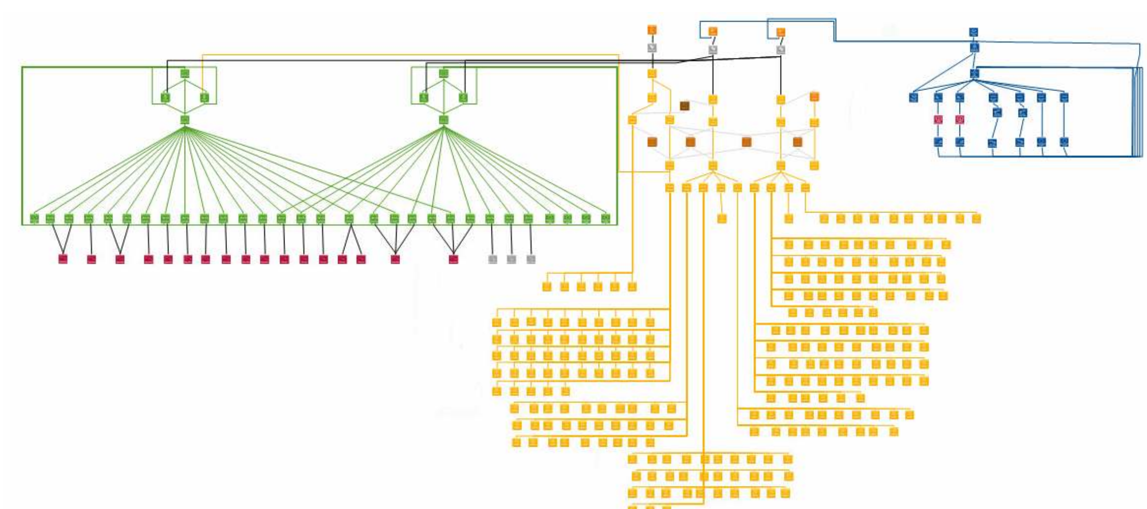


Figure 4.2: Current System Logical Map

4.1 Pressurization and De-icing System

Pressurization and de-icing are performed using bleed air, i.e., they are part of the pneumatic system. The pneumatic system has a maximum pressurization of 9.0 psi at 47,000 ft (Dassault, 2008). Figure 4.3 shows a schematic of pneumatic system and Fig. 4.4 shows a detailed view of the logical map of its model in SysArc©.

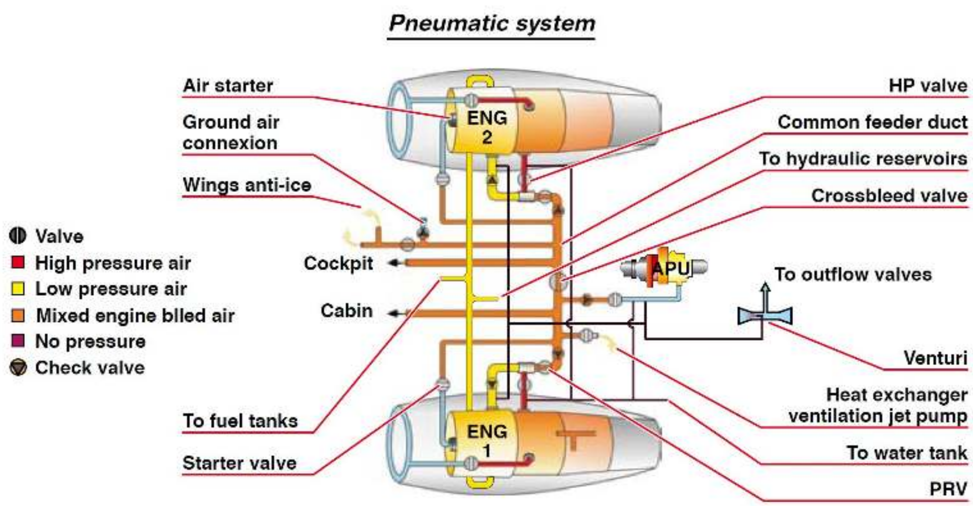


Figure 4.3: Falcon 2000 pneumatic system (Dassault, 2008).

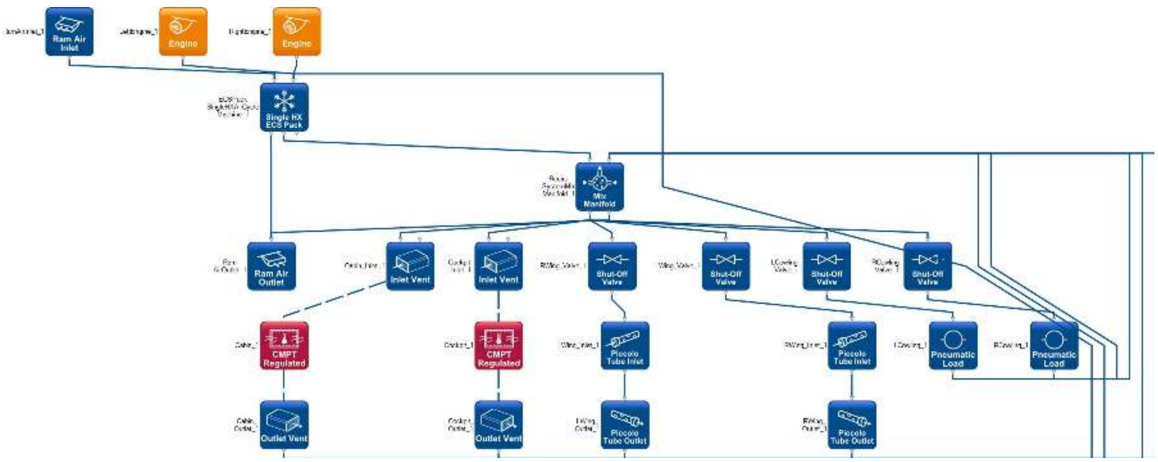


Figure 4.4: Original pneumatic system model

The first row of elements includes the engines (orange) and air inlet. The immediate level below is the Environmental Control System (ECS) and mix manifold, where recycled air is mixed with fresh air. This leads to the inlets and valves that control the airflow for pressurization and de-icing. All of this is returned to the mix manifold where some is recycled and part is vented from the aircraft.

The cabin and cockpit pressurization was modeled using climate controlled compartments with volumes equal to the pressurized volumes. The nose cone volume was added to that of the cockpit for simplicity; they are fed by the same duct. The wing de-icing was modeled using piccolo tubes that run the length of the leading edge. Piccolo tubes are simple representations of the ducts used to heat the leading edge.

4.2 Flight Controls and Hydraulic System

The flight controls model consists of the actuators for the primary and secondary flight controls and the rod and crank system that provides the input. The hydraulic system includes all the flight controls actuators and the steering, braking, thrust reversers, and landing gear actuation. The original hydraulic system is composed of two independent systems set up for redundancy. Each system has a single reservoir, two pumps and an accumulator. The system (Fig. 4.5) is pressurized at 21 psi and uses MIL-H-5606 mineral fluid (Dassault, 2008).

The logical map of the modeled hydraulic system can be seen in Fig. 4.6. The top level includes the engines with the gearboxes that power the hydraulic pumps or generate electricity to power a pump. Hydraulic fluid flows from the reservoirs to the pumps and into the accumulator where it is distributed into the individual actuators. From the actuators the hydraulic fluid flows back to the reservoirs.

This model includes the main components: pumps, reservoirs and accumulators. The components library in SysArc© does include an accumulator and, therefore, a reservoir element was used instead.

The electric pump is shown connected to the electric system. The primary flight controls actuator locations were provided in a private communication by Dassault. Since the exact location of the rest of the actuators was not readily available, in the model they were placed next to and centered on the surface or object they actuate on (see Fig. 4.1). Every actuator is attached to a flight control surface or a mechanical load such as steering. The thrust reversers were represented as loads because the simple actuator element does not represent the motion required for their motion.

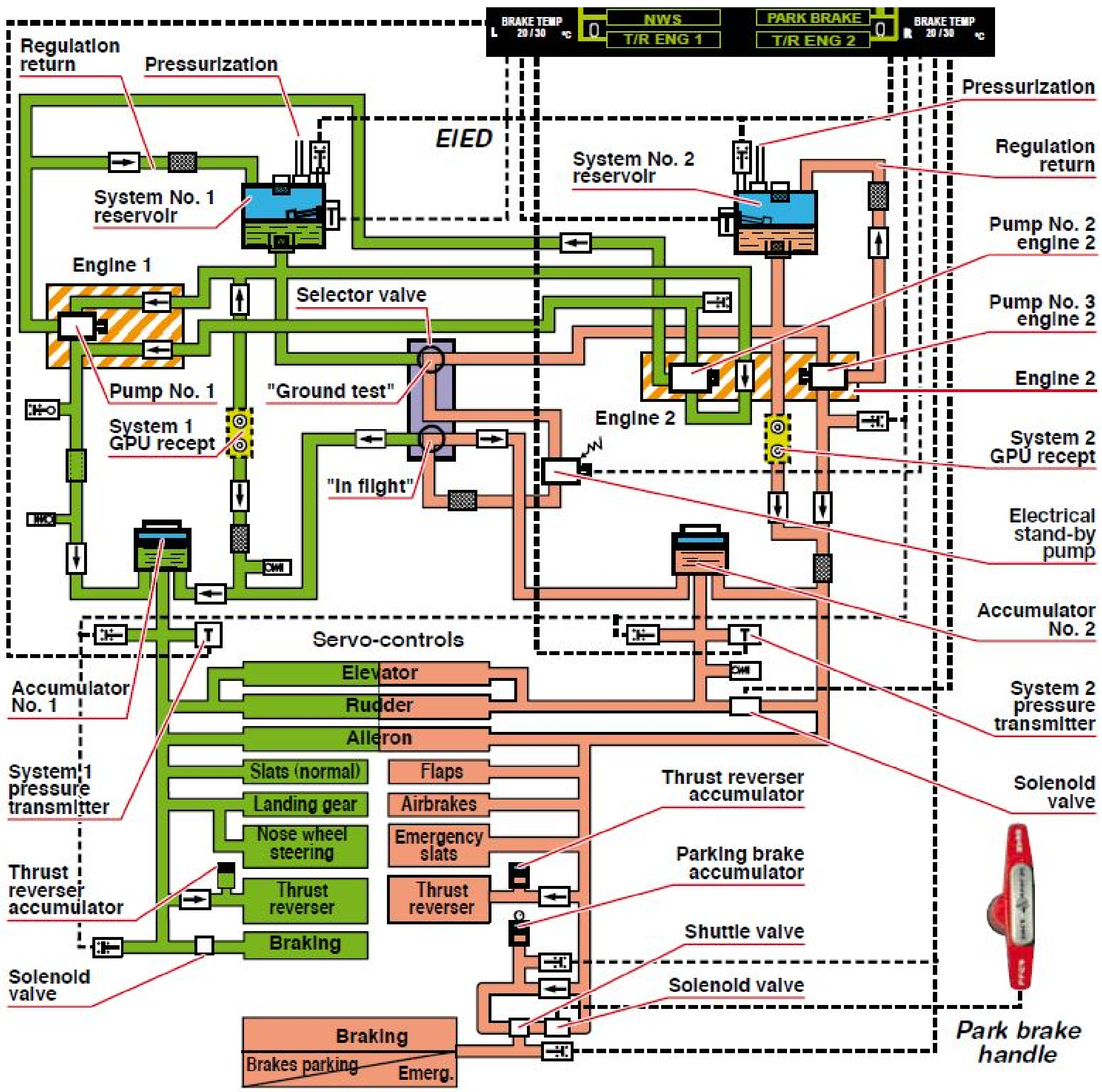


Figure 4.5: Falcon 2000 hydraulic system (Dassault, 2008).

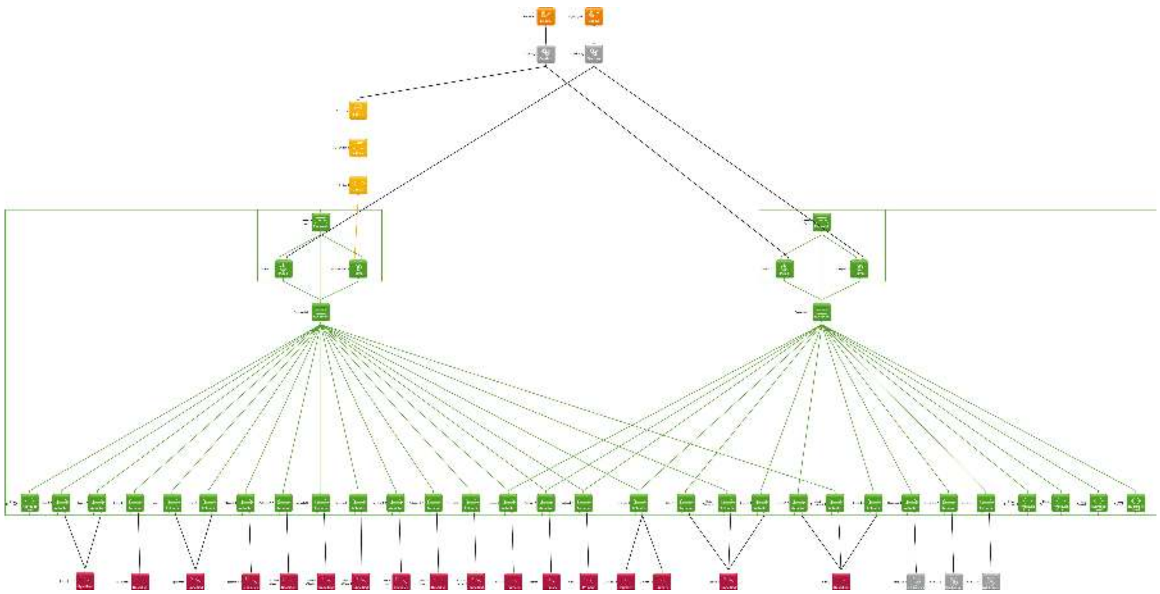


Figure 4.6: F2000 hydraulic system model.

4.3 Electrical System

The electrical system provides power to all equipment in the cockpit, such as navigation and engine control units and the "Arthur Q" artificial feel system, and to the pitot probes, static ports, and windshield de-icing.

The electrical system is powered by the two engines, an APU for use at start-up, and a battery for emergencies. Buses are employed to distribute the power to the required loads. The system runs on 28V DC (Dassault, 2001).

A diagram of the electric system is shown in Fig. 4.7 and Fig. 4.8 shows the logical map of the model. A more detailed view of the power distribution is shown in Fig. 4.9.

All the yellow blocks in Fig. 4.8 represent the aircraft electrical loads. The blocks at the top of the diagram are the engines and the APU with their gearboxes, which are attached to the electrical generators. The intermediate layers represent the buses that distribute the power and with their switches above them.

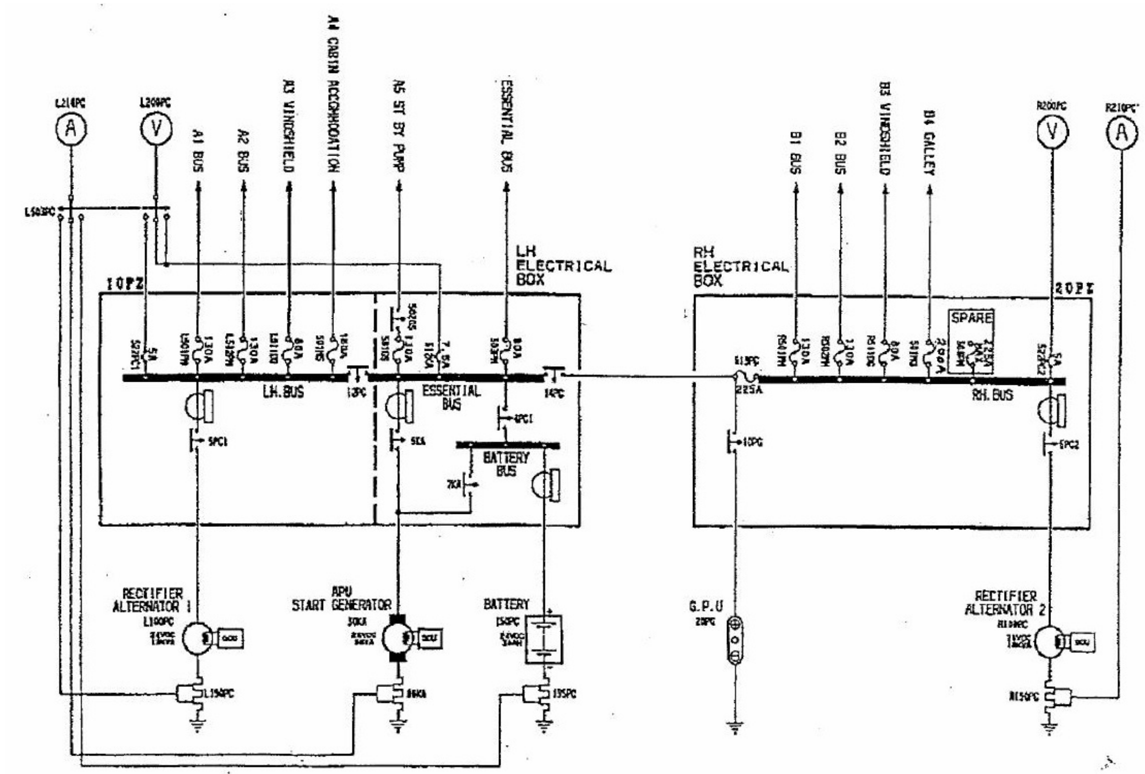


Figure 4.7: Falcon 2000 electrical system (Dassault, 2001).

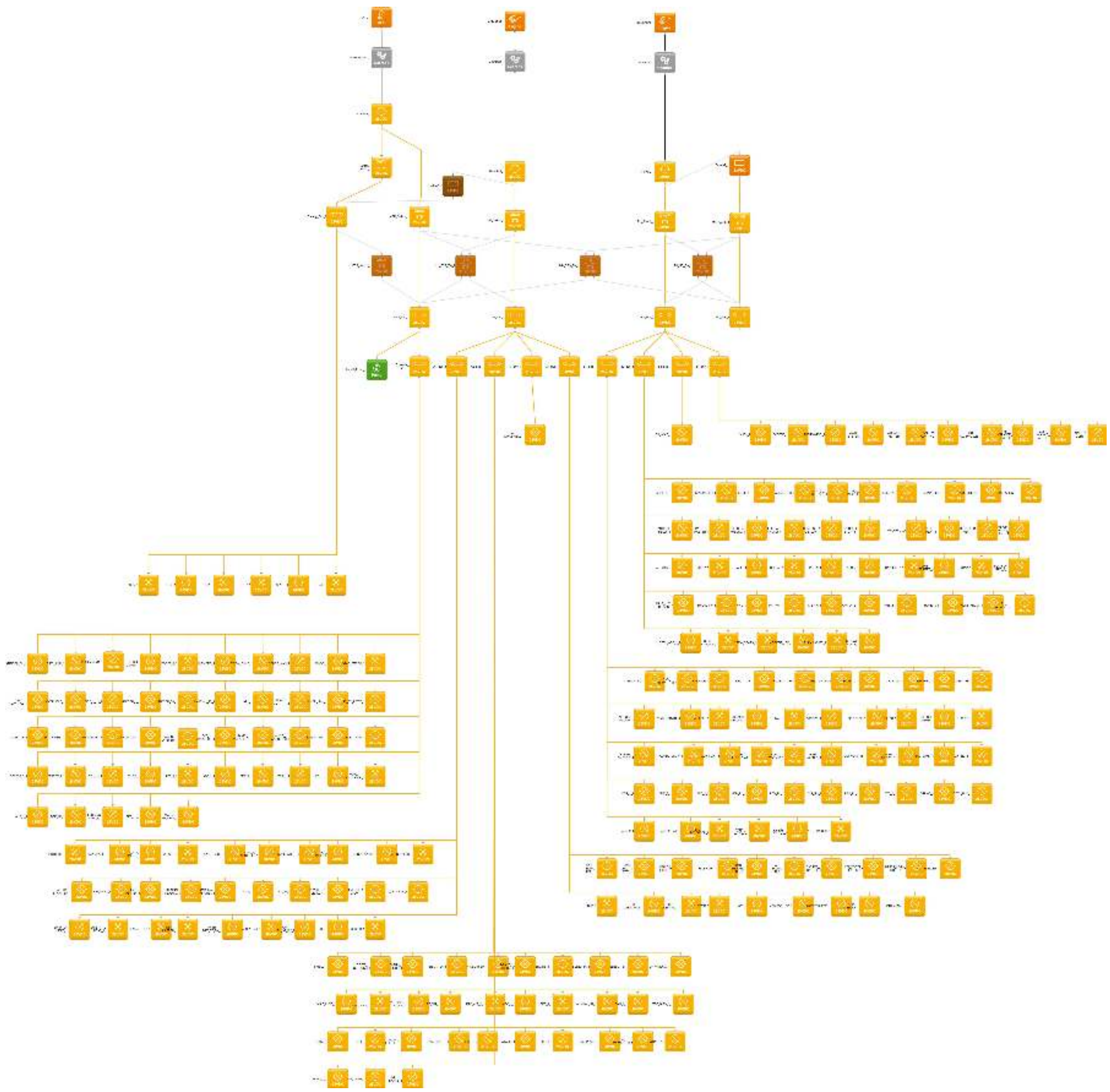


Figure 4.8: F2000 electrical system model.

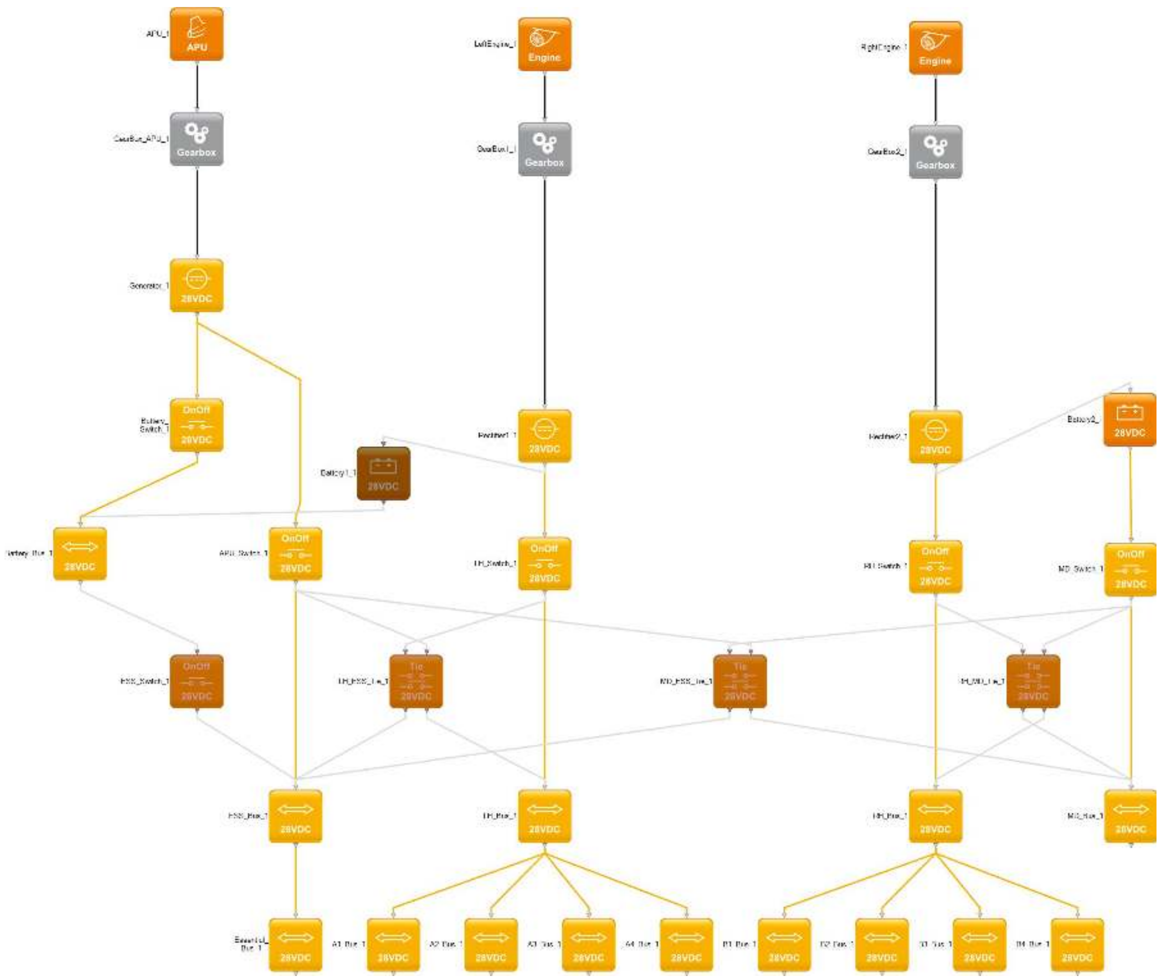


Figure 4.9: Detailed view of electrical system in Fig. 4.8.

4.4 Model Summary

SysArc© models the system weight and power consumption and includes power required-component weight scaling laws. The following are the weights calculated by the program:

- Electrical system, 810 lb.
- Hydraulic system, 647 lb.

- Pneumatic system, 1,255 lb.

The electrical system weight includes the batteries, buses, switches, wiring, and generators. It does not include the loads weight, which are part of the equipment and furnishing weights in the APD© model. The weight of the hydraulic system includes the pumps, reservoirs, actuators, accumulators and pipes. And, finally, the pneumatic system weight includes the ducts, mix manifold, compressor and heat exchanger.

The APU weight was accounted for separately (182 lbs). The engine weight was set in the initial APD© model and is not changed by SysArc© .

The program can display the power consumption in two ways: total maximum power consumed and the maximum power utilized at different flight stages, e.g., taxi, take-off, climb, cruise, descent, and landing, Fig. 4.10.

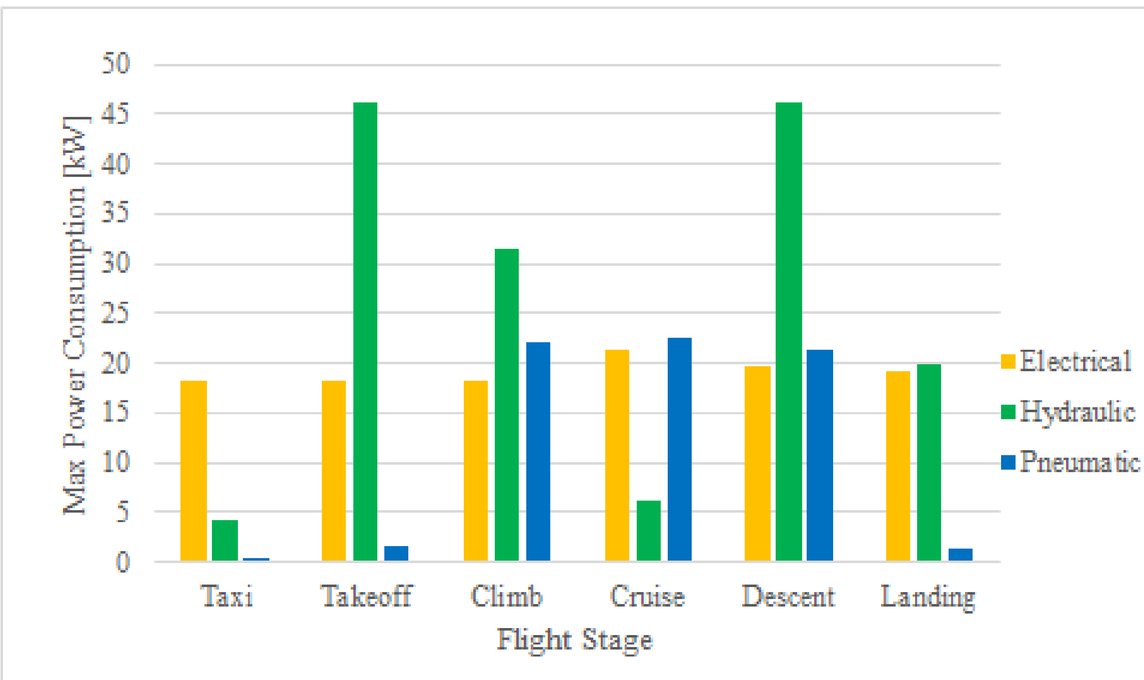


Figure 4.10: Modeled F2000 systems power consumption per flight phase.

5. Proposed MEA Systems Model

In the previous chapter the existing aircraft was modeled. Here the systems will be replaced with electrical components that are currently available or that are considered to be ready within five years. It is clear that it would not be efficient to simply substitute individual components but rather entire systems. In other words, a new architecture for the aircraft systems is needed. To differentiate it from the baseline model, the new aircraft will be identified as F2000MEA.

In the architecture proposed for the F2000MEA pressurization is carried out by electrically driven air compressors. Wing de-icing is accomplished using electric heat mats, while the engine cowling de-icing is still done with bleed air. The hydraulic system is partially electrified. The flight controls actuators(aileron, rudder, elevator, flaps, slats and airbrakes) are replaced with EMAs and EHAs. The remaining systems that use hydraulic power are not modified, i.e., landing gear, steering, braking and thrust reversers. Obviously, this new architecture requires significantly more electrical power and has to be designed to the same level of safety. Thus redundancy is necessary, which requires additional buses, batteries, and a ram air turbine (RAT).

The characteristics (weight and power) of components in the original model that were not replaced were retained unaltered. The new electrical components were sized to match the performance of the original component or subsystem they replace. This ensures that the aircraft response to the pilot is unchanged, the flight controls can provide the same amount of movement at the same speed, and that the aircraft meets all of the same requirements.

The components placement is shown in Fig. 5.1 and Figure 5.2 shows the diagram of its logical connections. These can be compared to Figs.4.1 and 4.2. The new electric components were placed in the same location as the components they are replacing. The prevalence of the yellow color in the diagram gives an indication of the level of "electrification" of this aircraft. The details of the new architecture are described in the following sections.

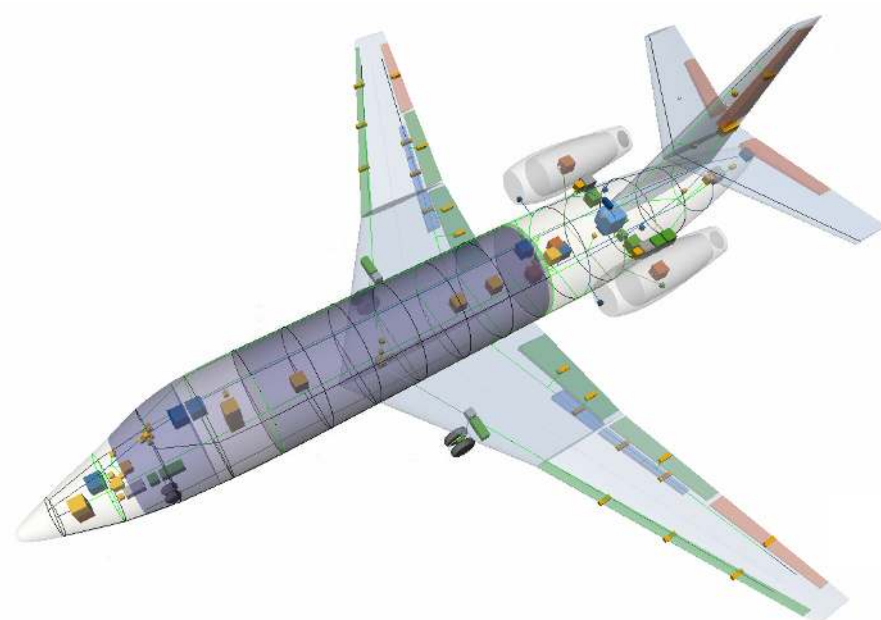


Figure 5.1: F2000MEA systems components placement.

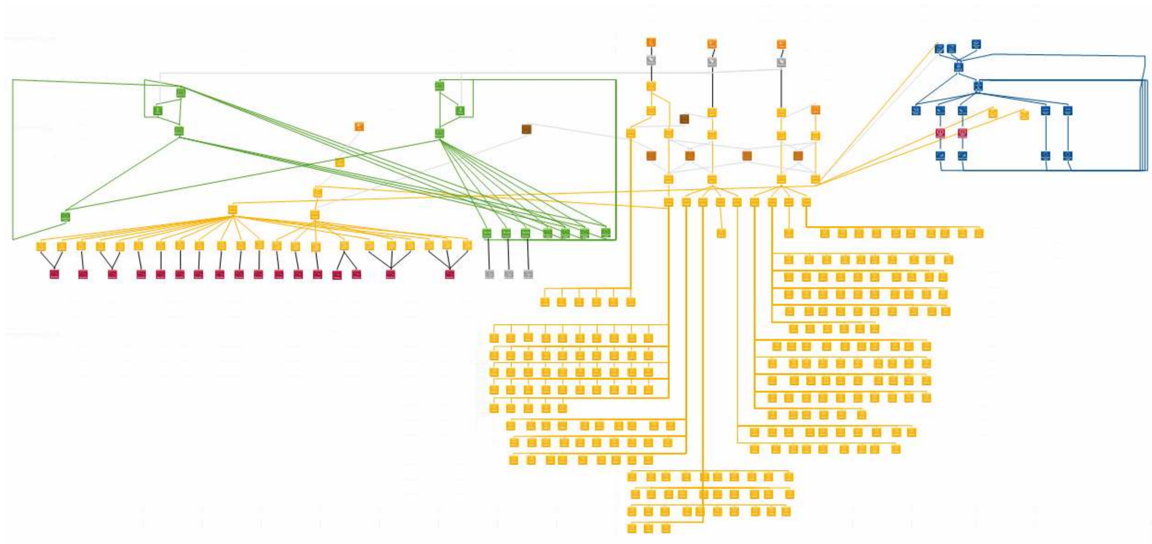


Figure 5.2: F2000MEA systems logical map.

5.1 Pressurization and De-icing

Bleed air was significantly reduced (only used for de-icing of the engine cowlings). De-icing of the wing is performed using heat mats and two electrical compressors are used for pressurization, following the example of the Boeing 787. Figure 5.3 shows the logical map of the remaining pneumatic system in the new architecture. The compressors and the de-icing components will be shown in the electrical system diagram.

For safety, redundancy is required and, therefore, two compressors are used. A single compressor can pressurize the cabin to 6 psi. Failure of one of the compressors allows the aircraft to retain certain pressure while a lower altitude is reached. The electric generators are sized to still provide power to both compressors for the case of one engine failure. Thus the aircraft remains pressurized even in the case of one engine failure. If both engines fail the compressors will not be powered.

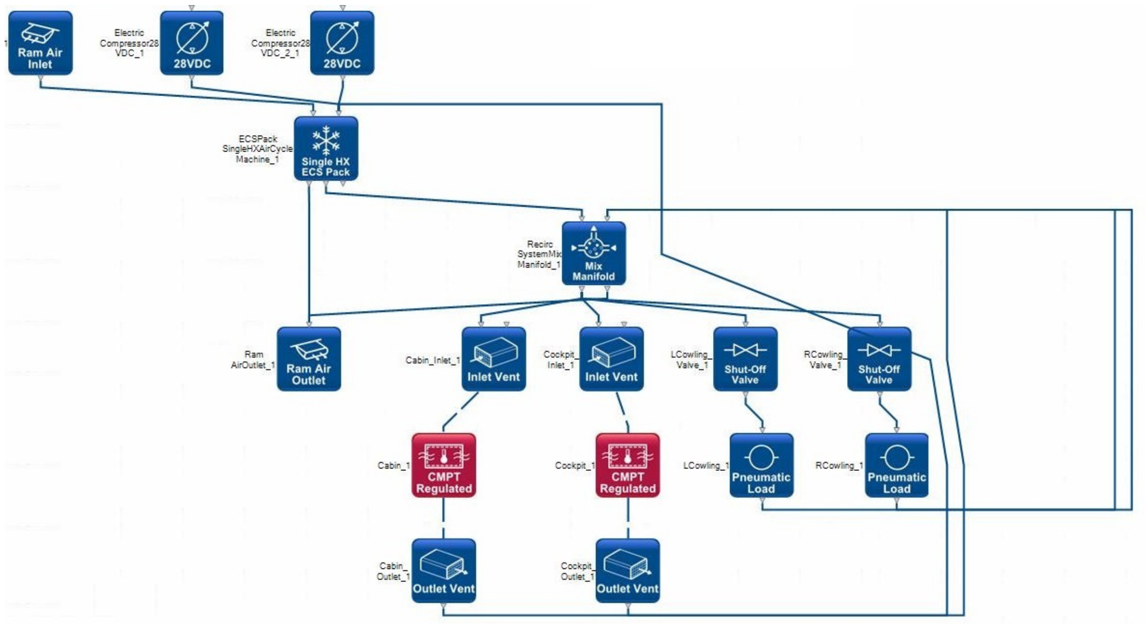


Figure 5.3: F2000MEA pneumatic system.

The compressors were sized to provide pressurization to 9 psi at altitude with air recycling every 2 minutes. Based on these requirements each compressor will draw up to 24 kW if only one is operational. The compressors would need to be placed in the aft maintenance bay behind the baggage compartment. The exact size of the compressors is not known but it is estimated that they would fit in the aft maintenance bay. However, this would create maintenance difficulties accessing all the needed components in the bay. It is likely that larger inlets would be required for feeding the compressors. The only location available for these inlets would be on the side of the fuselage, next to the compressors and they would need to be low on the fuselage to prevent interference with the engines. Clearly these considerations require detailed studies that are beyond the scope of the present study.

5.2 Flight Controls and Hydraulic Systems

In the F2000MEA only the flight controls actuators, originally hydraulic, are replaced with electric components; either EMAs or EHAs. The primary flight controls are replaced with EHAs, due to its higher reliability and faster actuation; EMAs have had problems with jamming. The secondary flight controls actuators are replaced with EMAs. The remaining functions remain hydraulic: landing gear steering and actuation, braking, and thrust reversers. It was considered that the level of technology maturity is not the one necessary for implementation at the present time. This results in the removal of two of the pumps, one from each of the independent hydraulic systems.

Figure 5.4 shows the diagram for the hydraulic system in the F2000MEA. The flight controls system, now electric, will be shown and further discussed as part of the electrical system.

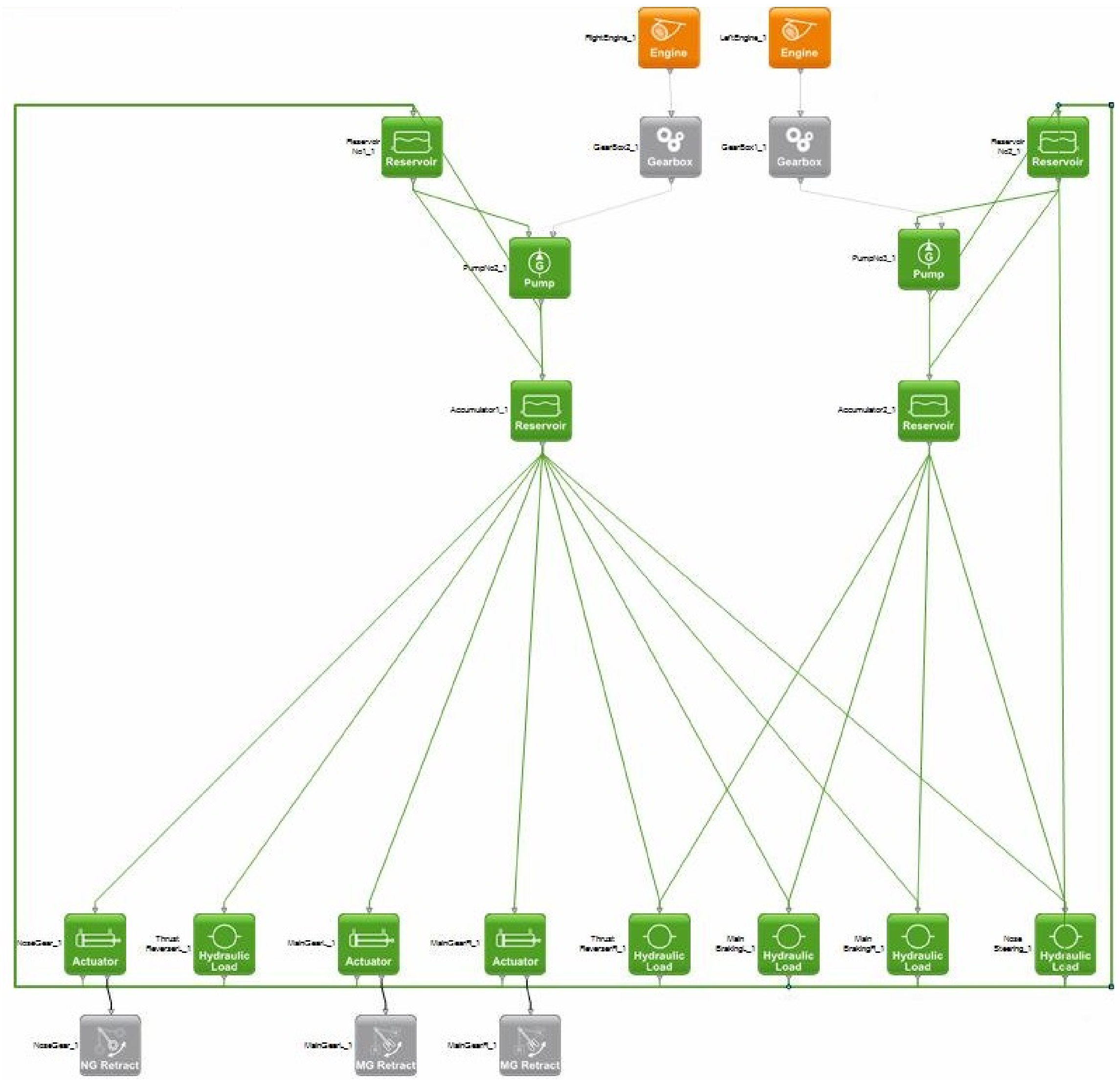


Figure 5.4: F2000MEA hydraulic system.

The original hydraulic system had two independent systems and this remains the case to ensure the same reliability. However, the entire system has been significantly reduced by the removal of the flight controls system. Therefore, one pump from each system is removed and a single pump has sufficient capability to pressurize the entire system. Each system controls certain functions. One controls the landing gear and steering, while the other controls braking and the thrust reversers. Each engine provides the power for one of the independent systems. If one engine fails the other system can support all of the hydraulic loads. In the case of both dual engine failure, an accumulator is associated with the braking system to provide the sufficient hydraulic pressure for braking. The accumulator associated with each system would provide some hydraulic power to help with the other loads although not at full power. A single engine will still provide the shaft power needed for a single pump to pressurize the hydraulic system.

5.3 Electrical System

In the new architecture, the flight controls system, pressurization and de-icing are now part of the electrical system. Specifically, two compressors for pressurization, plus heating mats for the wings de-icing plus EHAs and EMAs for the flight controls actuators are the extra loads. This requires larger generators to provide the additional power and another battery is also necessary. Due to their higher energy density, the original batteries in the systems and this new one are proposed to be lithium-ion.

A ram air turbine (RAT) is added for emergency power in case of engine failure. The distribution of the electrical power now requires two more buses. In summary, the original system architecture is not changed; all the new components are additions but no old components are re-routed through the new buses or attached to the new battery. Figure 5.5 shows the new electrical system, highlighting the new components (the red boxes). The box on the left covers all the flight controls system elements together with the RAT and the additional battery A detail view is presented in Fig. 5.6. The box on the right corresponds to the pressurization and de-icing systems, i.e., the electric heating mats and the compressors.

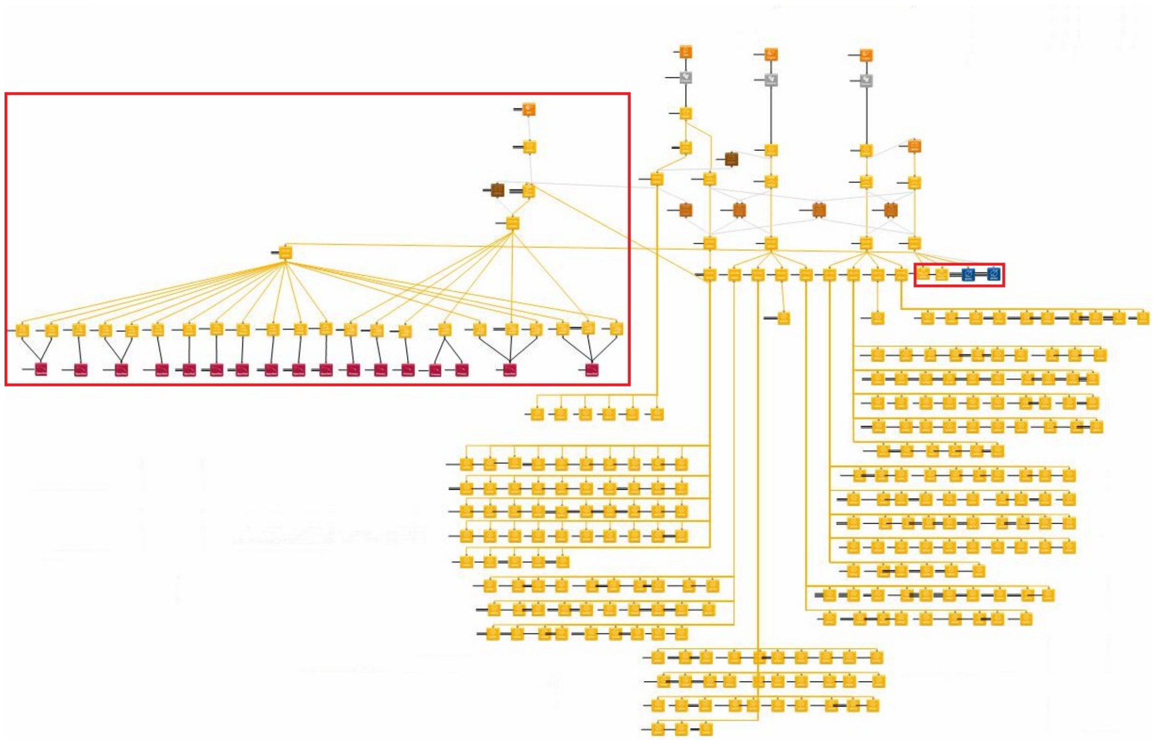


Figure 5.5: Diagram of the F2000MEA electrical system.

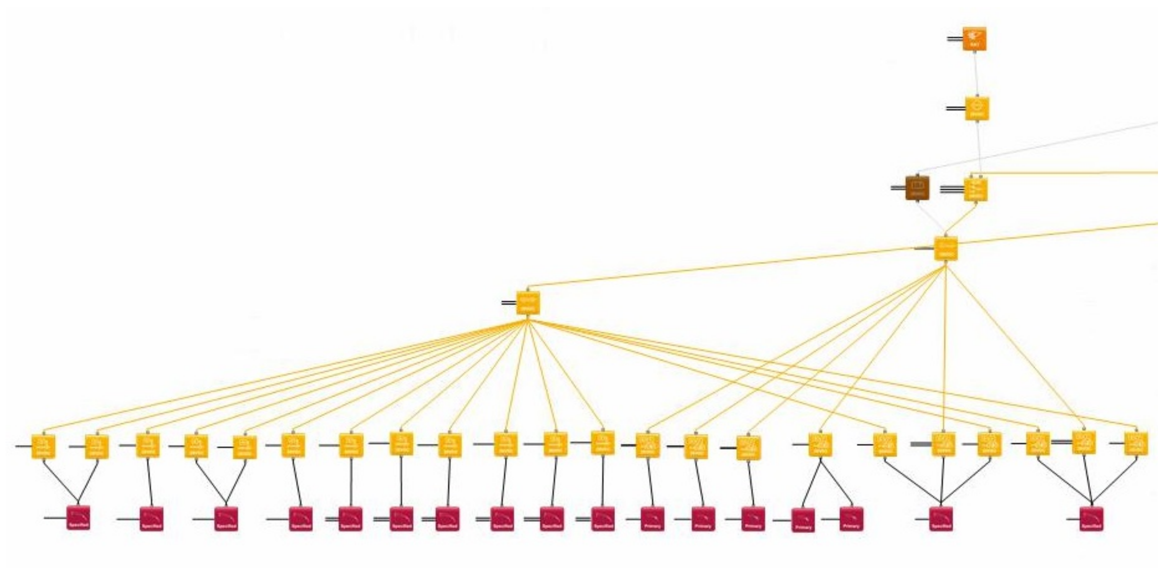


Figure 5.6: F2000MEA flight controls in the electrical system.

The new electrical system represents a minimum change with respect to the original one; only the modules pertaining to the functions now electrified are added. Therefore, the safety of the system can be assumed to be assured with supplementary redundancy and fail-safe measures added to these new modules. However, since the entire system is interconnected, some of the original components need resizing. In particular, a new emergency power bus is added. This bus routes power to all of the critical flight controls, primary flight controls and the slats, that must function, even in the case of engine failure. The slats are included here; their actuation was already redundant in the original system. In the F2000 the slats have hydraulic actuators and an emergency actuator. The new emergency bus and corresponding loads is powered now with a RAT (not present in the F2000) in the case of engine failure. An additional battery is added to provide power during landing, when the RAT becomes less effective due to lower airspeeds.

Safety is also ensured by components segregation and redundant wiring. Both sets of wires providing power to critical systems have separate paths to minimize the probability of total power failure in case of events such as an uncontained rotor burst. SysArc© modeling does not allow for multiple inputs and outputs to the system blocks. However, in the model, the impact of redundant wiring is taken into consideration by doubling the total wiring weight.

The EHAs and EMAs for the flight controls are sized to match the performance (force, torque, etc.) of the hydraulic actuators they replace. The actuator required force capacity is based on the control surface and, since these were not changed, the force capacity is unaltered. The modeled weight of the new electrical actuators for the flight controls is 391 lb.

The area of de-icing mats, placed along the leading edge of the wing is the parameter needed to determine their power consumption. The leading edges of the flaps are also de-iced. As modeled, their total weight is 131 lb, consuming 2.5 kW.

The existing batteries are replaced with lithium-ion batteries of the same power capacity. An extra battery is now added for emergency purposes. The power requirements of the emergency bus dictate the power of this battery. The weights of the total battery set are based on typical commerical values, giving a total weight of 111 lb and a power output of 24 kWhr.

The RAT is sized based on the power requirements for an engine failure case. The weight of the RAT was here scaled from the Falcon 7 RAT with respect to the power needs.. It produces 40 kW and weighs 60.0 lb.

The most significant change is the substantial growth required of the generators. They are sized to provide the maximum power consumed, which is determined for the different flight phases. The new generators were assumed to have the same power-to-weight ratio. They have a 60 kW capacity and weigh 393 lb.

6. Evaluation

After the two models, the F2000 and the F2000MEA, were created in SysArc©, it is possible to make a direct comparison and determine the overall aircraft impact of the new architecture. For clarity, the total effect is presented in this chapter broken into different aspects. The deltas between the two architectures in terms of systems weights, effect on engine SFC, power consumed and aircraft range are presented and discussed.

Tables 6.1 and 6.2 show the weight breakdown comparison for both system architectures. The first one uses the familiar ATA chapter classification and, the second one, a detailed component breakdown. It is evident that the F2000MEA is significantly heavier with, as expected, the largest increase belonging to the electrical

Table 6.1: F2000 and F2000MEA systems weights.

		System Components Weight [lbs]	
ATA Chapter	Title	Original	MEA
21	Pressurization	1,062.8	1,186.4
24	Electrical Power	810.0	1,669.6
27	Flight Controls	448.9	332.8
29	Hydraulic Power	197.7	112.8
30	De-Icing	192.0	131.3
36	Pneumatic	1,062.8	1,186.4

system. However, the weight reductions in the other systems are not sufficient to compensate the former's increase. The pressurization system does not show a significant weight reduction since most of the ducts to the cabin and cockpit are unaffected. The benefit of this change comes from the engine efficiency boost, given that the bleed has been practically eliminated. It should be noted that the entry for ATA 36 for MEA in Table 6.1 includes the electrically driven compressors. The values for each component are included in Table 6.2. The compressors weight (but not the demands on the electrical system to power them) are still displayed under the Pneumatic System heading.

Table 6.2: Component weights for both architectures.

		Original [lbs]	MEA [lbs]
APU		181.96	181.96
RAT		NA	60.00
Electrical System		810.02	2133.63
	Wiring	611.70	775.00
	Actuators	NA	332.80
	Batteries	81.79	111.30
	Generators	116.53	392.71
	Actuators	NA	390.57
	De-icing	NA	131.26
Hydraulic System		646.66	112.79
	Pipes	126.30	60.45
	Actuators	448.93	NA
	Pump	31.00	11.90
	Reservoirs	14.95	14.95
	Accumulators	25.49	25.49
Pneumatic System		1254.81	1186.42
	Ducts	720.02	545.05
	De-icing	192.00	NA
	ECS	342.79	291.37
	Compressor	NA	350.00
Total		2711.49	3432.84

Figures 6.1 to 6.3 present the power demands of the different systems for each flight segment for both architectures, original (F2000) and proposed (F2000MEA). Their aggregate (i.e., the total aircraft power consumption at each mission stage) is shown in Fig. 6.4.

Unexpectedly, the F2000MEA requires more power for all flight phases, with the exception of taxiing, and, therefore, for the overall aircraft mission. The large increase in power required for landing is due to how much more electrical than hydraulic power is required for actuation.

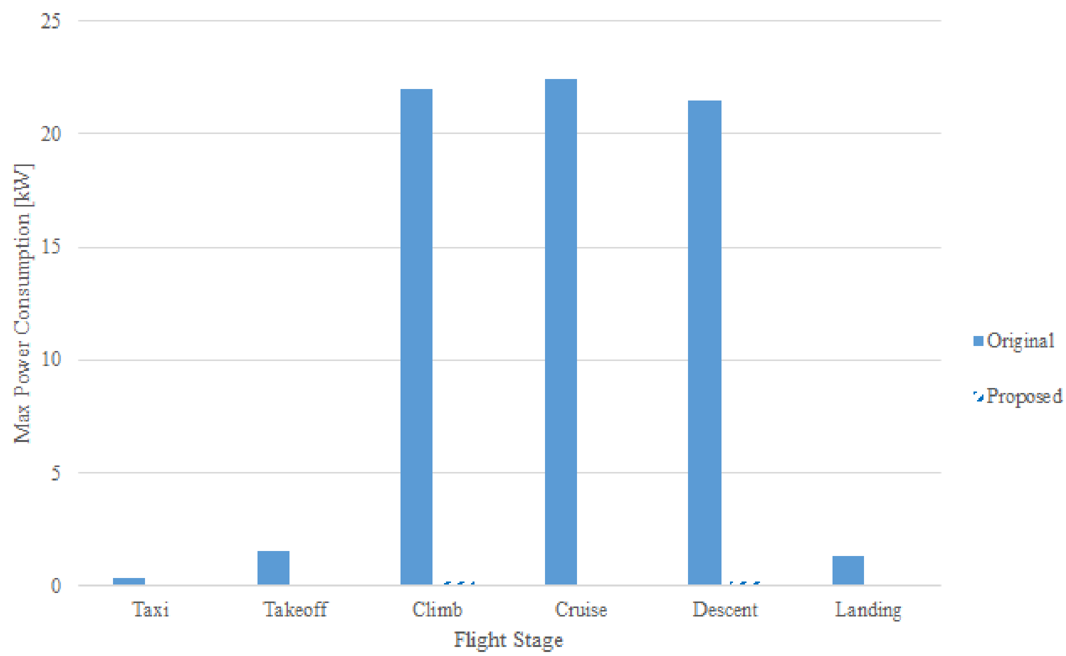


Figure 6.1: Pneumatic system. Calculated max power consumption per flight phase for F2000 and F2000MEA.

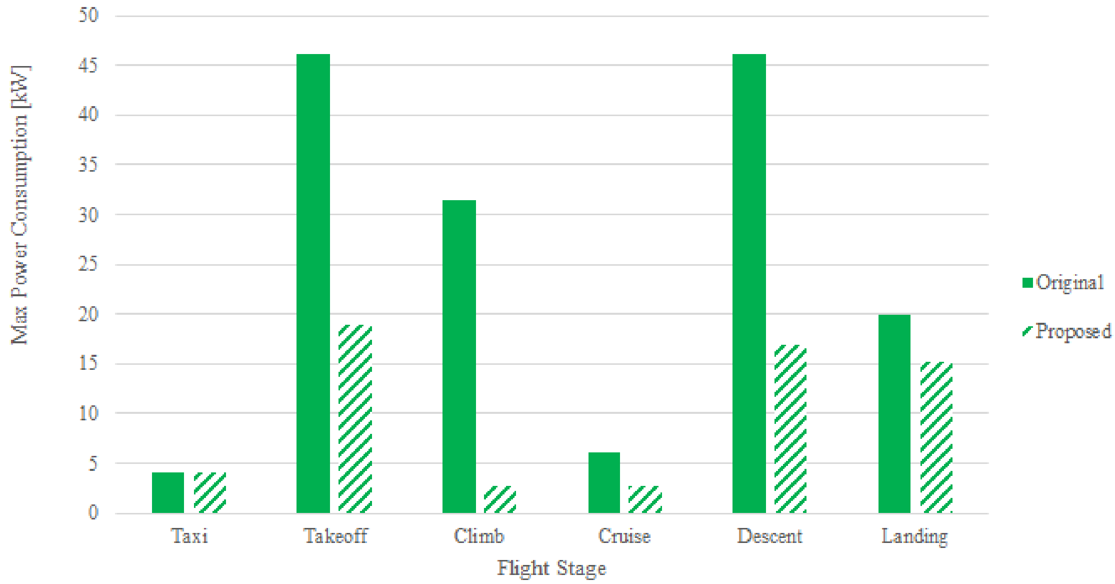


Figure 6.2: Hydraulic system. Calculated max power consumption per flight phase.

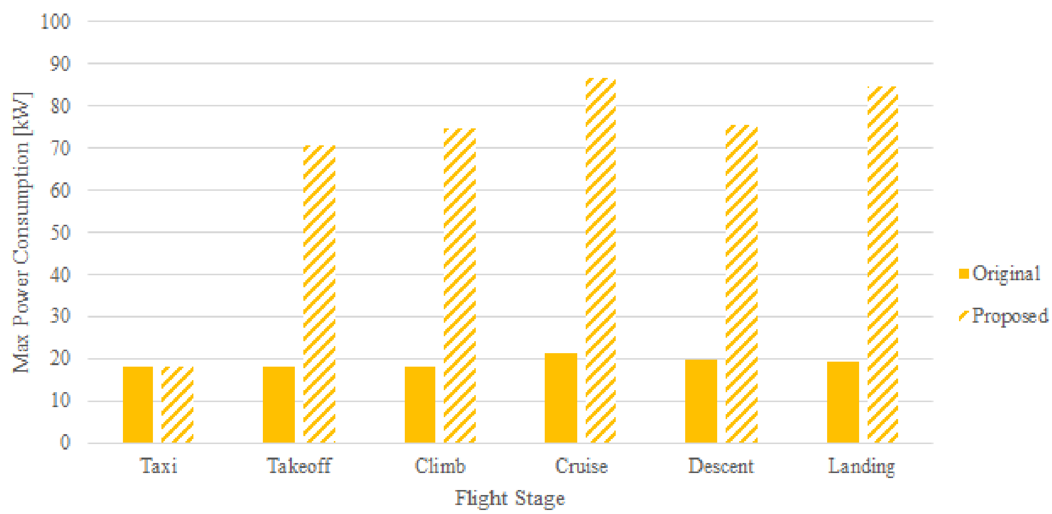


Figure 6.3: Electrical system. Calculated max power consumption per flight phase.

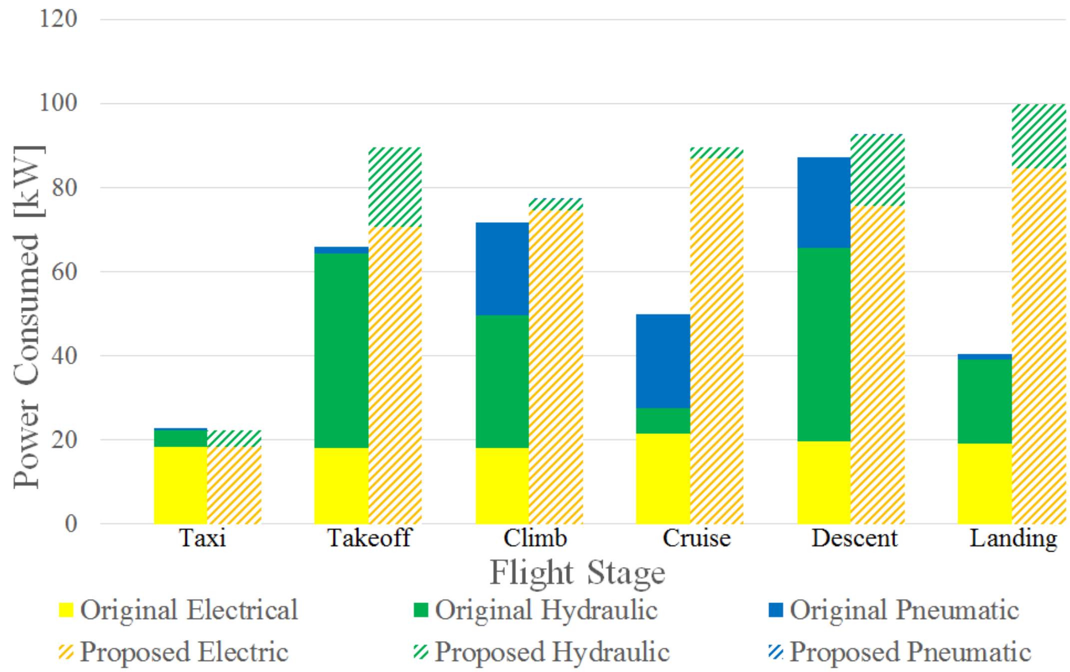


Figure 6.4: Comparison of total system power consumption per flight phase between F2000 and F2000MEA.

The Falcon 2000 engine specific fuel consumption, at cruise, Mach 0.8 at 40,000 ft, is 0.752 lb/hr/lb. The elimination of the engine bleed for pressurization and de-icing reduces that number to 0.748 lb/hr/lb for the same flight conditions.

The Falcon 2000 has a range of 3,275 NM, at Mach 0.8, at 40,000 ft. After all the effects are taken into consideration, the F2000MEA would have a range of 2,600 NM for the same flight conditions.

7. Conclusions and Recommendations

The results of the previous section indicate that, with the current level of technology, direct replacement of systems by electrical ones is detrimental to the aircraft performance. The proposed more-electric architecture increased the weight by over 700 lb., i.e., 27%, in part as a consequence of the increase in power consumption of the electrical components. The decrease in specific fuel consumption, produced by the elimination of the bleed-air, was not sufficient to offset the weight penalty and, therefore, the range was significantly reduced, specifically by 20%. Other potential penalties were not captured in this analysis such as the effect on drag that the intakes for the air compressors would have. Also, a more detailed study is needed to determine whether there is enough space for an easy installation and maintenance of the new components, specifically the air compressors for pressurization, but also the larger generators or batteries.

It is estimated that, at the present time, the cost of an MEA would be greater than that of a current system because, typically, the electrical components are costlier than their hydraulic or pneumatic counterparts. Also, a compressor is more expensive than the relatively simple ducts and valves required for bleed-air.

The reliability of some of the electrical components has not been fully demonstrated to be equivalent to that of proven and mature hydraulic and pneumatic technologies.

However, the promise of the MEA remains intact because of the trends that the development of power electronics and electrical components displayed in recent years. Progress is constant and power densities and efficiencies have not yet reached their maximum. There are two specific recommendations drawn from this thesis. One is to re-assess the same concept with improved components, i.e., when the new technologies under development in the areas of batteries, generators, power distribution reach higher technology readiness levels. The second one is to focus on the assessment of the more-electric architecture not on an existing platform but, rather, incorporate it already at the aircraft conceptual and preliminary design stages. This would allow to reap the full benefits that new components would bring.

REFERENCES

Anonymous. (2001). Advances in more-electric aircraft technologies. *Aircraft Engineering and Aerospace Technology*, *73*(3), 294.

Cao, B., W., & Atkinson, D. (2012, September). Overview of electric motor technologies used for more electric aircraft. *IEEE Transactions on Industrial Electronics*, *99*(9), 3523-3531.

Chakraborty, I. e. a. (2014). A requirements-driven methodology for integrating subsystem architecture sizing and analysis into the conceptual aircraft design phase. In *14th aiaa aviation technology, integration, and operations conference, aiaa aviation.*

Dassault. (2001, February). *Falcon 2000 electrical load analysis* (Tech. Rep.). Dassault Falcon Jet.

Dassault. (2004, June). *Falcon 2000 technical specifications* (Tech. Rep.). Dassault Aviation.

Dassault. (2008, June). *Falcon 2000 series systems description* (Tech. Rep.). Dassault Falcon.

Dassault. (2011). *Focus no 3. the electric aircraft* (Tech. Rep.). Dassault Aviation.

Derrien, J. (2012). Electromechanical actuator(ema) advanced technologies for flight controls. In *28th international congress of the aeronautical sciences.*

Faleiro, L. (2005). Beyond the more electric aircraft. *Aerospace America*, *43*(9), 35.

Goraj, Z. (n.d.). An overview of the de-icing and anti-icing technologies prospects for the future. In *24th international congress of the aeronautical sciences.*

Jones, R. I. (2002). The more electric aircraft, assessing the benefits. *Proceedings of the Institution of Mechanical Engineers*, *216*(5), 259-269.

Karimi, K. (2007). *Future aircraft power systems - integration challenges.* Web. (Boeing)

Kirby, M. (2008, October). *Bombardier flight tests all-electric braking system.* Web. (Flight Global)

Moir, I., & Seabridge, A. (2008). *Aircraft systems: Mechanical, electrical, and avionics subsystems integration* (3rd ed.). John Wiley and Sons, Ltd.

Naayagi, R. (2013). A review of more electric aircraft technology..

Pacelab. (2016a). *Pacelab apd.* Web. (https://www.pace.de/products/pacelab-apd/)

Pacelab. (2016b). *Pacelab sysarc.* Web. (https://www.pace.de/products/pacelab-sysarc/)

Rosero, O. J. A. E., J.A., & Romeral, L. (2007). Moving towards a more electric aircraft. *IEEE A and E Systems Magazine*.

Spencer, K. (2013, December). *Investigation of potential fuel cell use in aircraft* (Tech. Rep.). Institute for Defense Analyses.

Tarter, J. (1991, April). Electric brake system modeling and simulation. *SAE Technical Paper Series*.

Whyatt, G., & Chick, L. (2012, April). *Electrical generation for more-electric aircraft using solid oxide fuel cells* (Tech. Rep.). US Department of Energy.