FT2025
SWEDISH AEROSPACE AT THE CROSSROADS
The 12th Swedish Aerospace Technology Congress FT2025 in Stockholm October 14-15, 2025

A Review of Aircraft On-Board Systems in the Context of Energy and Power Management

Erick Espinosa-Juárez[1] **and Alessandro Dell'Amico**[1]

[1]*Division of Fluid and Mechatronic Systems (Flumes), Linköping University, Linköping, Sweden.*
E-mail:, erick.espinosa.juarez@liu.se

Abstract

On-board systems in fighter aircraft are expected to deliver high performance under extreme and hostile operational conditions. As technology advances, system architectures are shifting from traditional federated configurations toward integrated and electrified designs characteristic of more electric aircraft. A range of architectural configurations, combining different power sources and consumers, can each fulfil the required system functionalities, offering distinct advantages and drawbacks. To tackle this problem, energy and power management offers a solution-independent, agnostic framework for assessing on-board system architectural decisions with respect to their impact on top-level aircraft requirements. Nonetheless, a clear understanding of the state-of-the-art and design sensitivities of these systems are needed in early stages of design. This study describes on-board system architectures and their associated trade-offs to quantify and compare architectural options available to system designers. It reviews on-board systems from both federated and more electric aircraft architectures, linking them to the aircraft-level functions they fulfill and outlining key design trade-offs. The systems reviewed include flight control systems, hydraulic systems, fuel systems, electrical systems, pneumatic systems, environmental control systems, auxiliary power systems, emergency power systems, and landing gear systems. The review highlights that the interdependence and diversity of options of on-board systems require robust integration frameworks that assess them collectively, rather than in isolation, to achieve a balanced architecture at the aircraft level.

Keywords: Aircraft Systems, Energy Management, More Electric Aircraft, System Architecture

1 Introduction

On-board aircraft systems are a set of interacting parts, components, processes or functions connected together in an organized way that are combined to enable an aircraft to perform a particular function (Moir [1]). The present study reviews the on-board systems that impose the highest power requirements in military aircraft, outlines their architectures, and summarizes the key trade-offs they present.

To accomplish this, a basic functional description of a fighter aircraft is summarized in a Perform flight operations function, which is shown in Figure 1. The figure identifies the inputs and outputs of the functions (boxes) and shows how a fighter aircraft transforms external information, pilot commands and fuel (in bold) into aircraft movement, crew life support and deliver ordnance (underscored and in bold). The on-board systems architectures and trade-offs are now described.

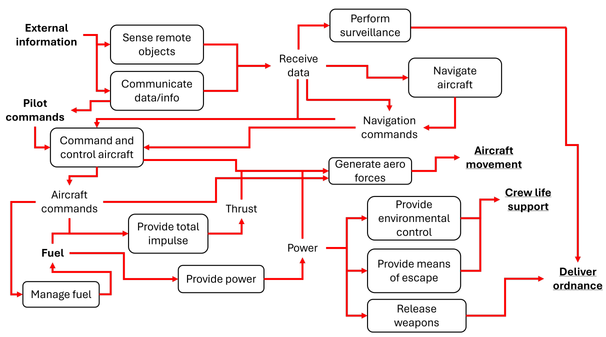

Figure 1: The Perform flight operations function of a fighter aircraft. Modified from Jackson [2].

2 Flight Control Systems

Flight Control Systems (FCS) enable an aircraft to achieve rotational motion around a fixed set of axes in a three-dimensional space. The major functions of the system are to transfer pilot and autopilot commands into control surface deflection and provide load feel. These are subfunctions of the Command and control aircraft function from Figure 1. The FCS also relieves the pilot's workload by providing automatic trimming in all axes and performing automatic limiting functions, such as the Carefree Maneuvering system used in the Saab Gripen (Hillgren [3]).

2.1 System Description

In conventional aircraft, flight control is accomplished through deflection of the primary control surfaces: ailerons control roll, elevators control pitch and the rudder controls yaw. In contrast, high-performance delta-wing aircraft like the Saab Gripen use elevons, canards, and rudder as primary control surfaces (shown in blue in Figure 2). Elevons provide combined pitch and roll control, with canard deflection offering supplementary authority. Secondary flight controls include high-lift devices such as slats and flaps, as well as airbrakes (shown in red in Figure 2). These surfaces manage functions such as lift augmentation, drag modulation or trim, which require slower and less frequent movements than primary control surfaces.

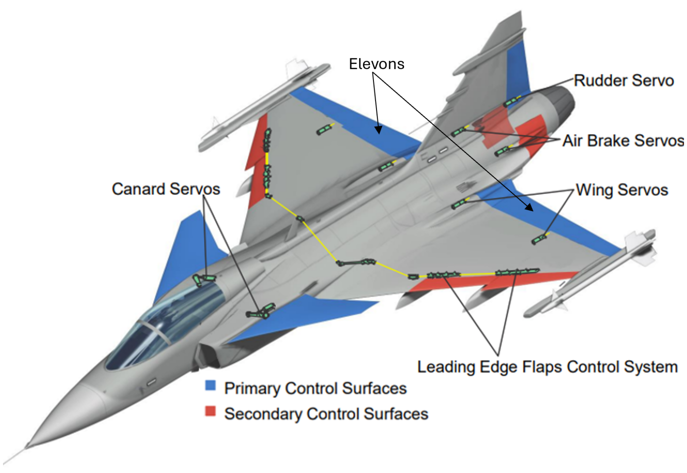

Figure 2: Saab Gripen's control surfaces. From Landberg [4].

Since loss of the FCS would mean imminent loss of control, it is considered a flight-critical system where a high level of redundancy is needed. Earlier versions of the Gripen therefore had a triple redundant asynchronous digital FCS that computed its control laws at 60 Hz and achieved a maximum deflection rate of approximately 60 °/s (Hillgren [3]).

2.2 System Trade-offs

Major trade-offs of the system include selection of actuator signaling method, actuation type and balance between redundancy vs. dispatch reliability (Jackson [2]).

One such signaling method is power-assisted mechanical signaling. It uses rods, cables, and pulleys, with optional hydraulic assist. This continuous mechanical connection provides robustness, immunity to *Electromagnetic Interference* (EMI), simpler tactile feedback and easy integration for mechanical reversion in case of failure. However, they also have increased weight and volume, susceptibility to jamming, difficult integration into the airframe, and limited routing flexibility.

In comparison, *Fly-by-Wire* (FbW) replaces the mechanical linkages with digital signaling through cables. FbW lowers weight, improves reliability, enables automation and lowers maintenance. However, their susceptibility to EMI and *High Intensity Radiated Fields* (HIRF) demand shielding and separation. The problem is amplified in aircraft with extensive use of composite material that lack the shielding properties of metal airframes. In contrast, *Fly-by-Light* (FbL) systems send command and feedback signals through optical-fiber lines which are lighter, more compact, have smaller bending radii, have less attenuation, are immune to EMI and HIRF and have higher bandwidth. Nonetheless, challenges such as waveform distortion, receiver saturation, thermal performance variations and expensive connectors and couplers have limited its application outside experimental aircraft (NASA [5]).

Wiring mass is reduced further in *Fly-by-Less-Wire* (FbLW) systems such as the one used in the Northrop B-2 Spirit shown in Figure 3. Pilot inputs and sensor data from the *Air Data System* (ADS) and *Attitude Motion Sensor Set* (AMSS) are sent to the *Flight Control Computer* (FCC), which transmits commands through a digital bus via *Actuator Remote Terminals* (ARTs) to the actuators (Britt [6]). FbLW retains wire in the most critical links and reduces integration complexity but introduces latency (delay), reduced bandwidth and higher local complexity to fit a transceiver. The ultimate concept, *Fly-by-Wireless* (FbWL), eliminates all physical links but has the greatest concerns regarding security, safety, and still has weight associated with the power distribution needed for each element.

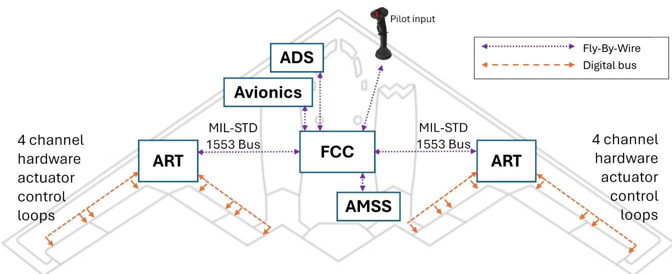

Figure 3: Northrop B-2 Fly-by-Less-Wire (FbLW) flight control architecture. Based on Britt [6].

While signal routing governs how commands are delivered, actuators determine how those commands are physically executed. Actuators deflect control surfaces by transforming electric or hydraulic energy into mechanical linear (or rotary) displacement and are one of the major consumers of auxiliary power. Actuators must operate with high precision at displacements below 1% of stroke, while meeting strict requirements for stability, accuracy, response time, stiffness, and disturbance rejection of aerodynamic loads (Lopes Junior [7]). Figure 4 illustrates the three main primary actuator types used for aerospace applications. The usage of one or the other results in different power supply systems, control strategies and system architectures.

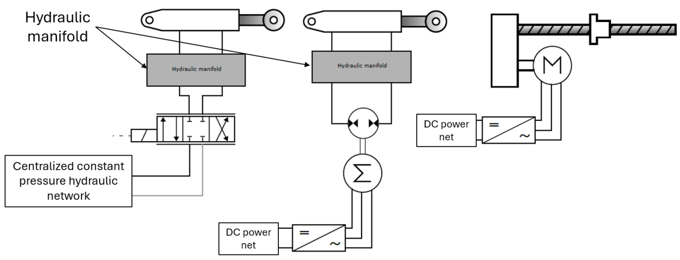

Figure 4: Most common actuator technologies in aerospace. From left to right, these are the Servohydraulic Actuator (SHA), Electrohydrostatic Actuator (EHA), and Electromechanical Actuator (EMA). From Dell'Amico [8].

Servohydraulic Actuators (SHA) are regarded as the most reliable solution for flight control actuation and are shown left in Figure 4. These are driven by a servo valve supplied with hydraulic pressure from a centralized system. Since SHA are controlled through electric signals, they fall in the category of *Signal-by-Wire* (SbW) systems. SHA possess excellent power density, low heat emission, can use oil as coolant, flight heritage, better handling of failure modes, low inertia and higher acceleration capabilities. Nonetheless SHA require a complex hydraulic system, have non-linear control behaviour and low efficiency product of leakage in the servo valves and its throttling control. Damping is also needed for failure modes that allow control surfaces to float freely, as it helps avoid excessive fluttering. This is an important feature that hydraulic actuators provide by default.

More Electric Aircraft (MEA) such as the Lockheed F-35 (Figure 5) use *Electrohydrostatic Actuators* (EHA), which are self-contained units including an electric motor, fixed-displacement pump, valve manifold, and cylinder (centre in Figure 4). Powered by the DC electrical network, EHAs enable a *Power-by-Wire* (PbW) architecture that is easier to install and maintain (plug and play), draws power only when needed, improves survivability, and allows energy exchange with other on-board systems. Efficiency is increased through displacement control and reduced leakage. Limitations include high motor inertia, mechanical friction, susceptibility to EMI, temperature-dependent performance, and complex local heat management. Load holding (maintaining a given position under external load) may require constant power unless a brake is used, unlike other actuator types that can use hydraulic blocking or mechanical locking.

The *Electromechanical Actuator* (EMA) uses an electric motor to drive a ballscrew, with or without a gearbox, to generate motion (right in Figure 4). It eliminates hydraulic fluid reducing leak risks, and allows easier fault monitoring and isolation. However it has low power density, high mechanical resistance, risk of jamming, and susceptibility to bearing failure. As a result, EMAs are mainly used in non-flight-critical secondary control surfaces or thrust-reversers where rapid actuation is not imperative and a fail-frozen mode is desirable. In general while SHAs offer higher actuator-level power density, EHAs and EMAs provide superior power density at the system supply level. (Dell'Amico [8]).

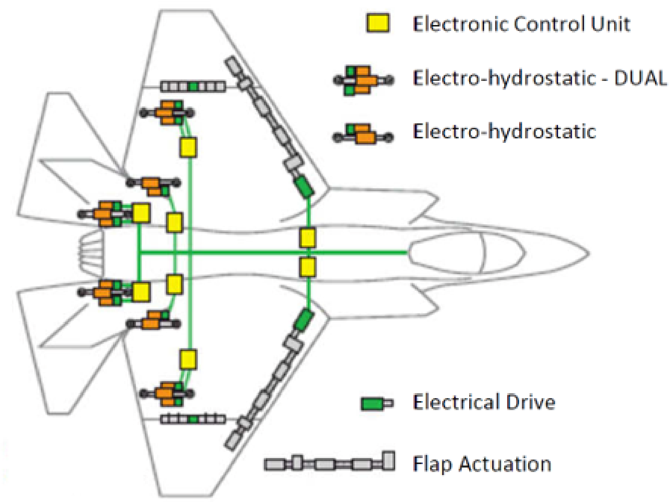

Figure 5: The F-35 actuation system employing Electrohydrostatic Actuators (EHA) and Electromechanical Actuators (EMA) for flap control. From Delbecq [9].

Regarding redundancy, SHA must maintain full performance after the loss of one hydraulic circuit or an engine, as noted in Section 3. This is achieved using independent hydraulic systems, multiple pumps with separate power sources, or dual source actuators, as demonstrated on the Northrop YF-23 (Vieten [10]). Electrical architectures employing EHAs achieve redundancy through multiple generators, at least two dissimilar standby sources and duplicate actuators. In all cases, signal paths should be physically separated, and mechanical reversion is desirable for control if SbW is lost (Roskam [11]). More-electric aircraft such as the Airbus A380 use SHA along with *Electrohydrostatic Backup Actuators* (EHBA), combining two distinct power sources for redundancy (Maré [12]).

3 Hydraulic Systems

The primary function of the hydraulic system is to supply auxiliary power through uncontaminated pressurized fluid. The system fulfills the Provide power function shown in the Perform flight operations diagram of Figure 1.

3.1 System Description

Figure 6 shows the Saab Gripen federated centralized hydraulic system. The system comprises two independent and separated hydraulic circuits (red and green), with redundant supplies to the FCS, landing gear and brakes. The system also provides power to some of the fuel transfer pumps and the aerial refueling receptacle. Accumulators support the pump during peak flow demands, maintain pressure when engines are off, and provide emergency hydraulic backup. The *General Electronic Control Unit* (GECU, shown in Figure 9) monitors reservoir level, fluid pressure and temperature as well as activates shut-off valves in case of leakage detection.

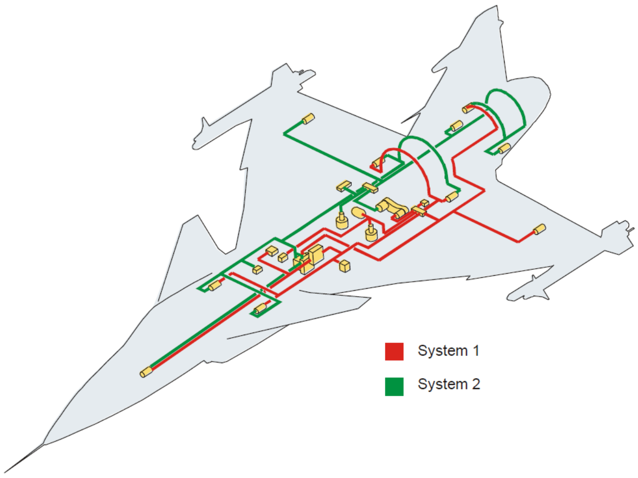

Figure 6: The Saab Gripen's double redundant hydraulic system. From Landberg [4].

Figure 7 illustrates the hydraulic power generation in the Saab Gripen. During normal operations, shaft power from the engine is used to drive the *Airframe Mounted Accessory Drive* (AMAD), which contains one independent pump for each hydraulic system and an electric generator. The system automatically reconfigures during emergencies to maintain hydraulic power supply, as discussed in Section 9.1. The Gripen's hydraulic power supply is in the vicinity of 54 kW, which is significantly less than the reported 107 kW of the F-16A, 142 kW of the F/A-18C/D or the 215 kW of the F/A-18E/F (Landberg [4]).

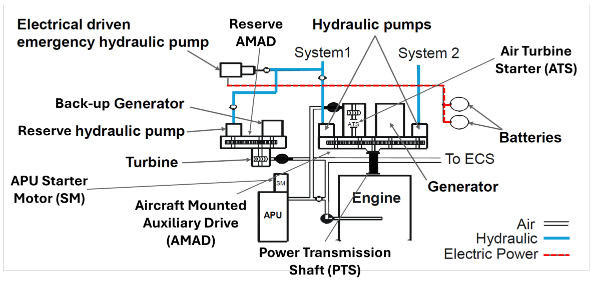

Figure 7: Saab Gripen hydraulic and auxiliary power generation layout. Modified from Landberg [4].

3.2 System Trade-offs

Trade-offs can be analyzed from the perspective that a centralized hydraulic system is only one possible solution to cover the Provide power functional requirement. More electric aircraft are moving towards electrified de-centralized power-by-wire solutions. The selection of one or the other will determine the type of actuator used as discussed in Section 2.2. Centralized hydraulic systems remain the most common systems due to their proven reliability, simple construction, low heat output, and ability to use hydraulic fluid as a thermal sink. Nonetheless, they have low network-level power density, higher maintenance and fluid conditioning needs, require line separation to prevent common-cause faults, and are less energy efficient in engine power use. An electric power distribution system (like the one used in the Lockheed F-35 shown in Figure 5) offers potential benefits such as reduced cost and weight, simplified maintenance, improved survivability, and the use of a common power source for all on-board systems, enabling advanced power management and load sharing among consumers. Its drawbacks include costly new developments in hardware, thermal management of its components and transformers as well as updated qualification and flight testing methods (Robbins [13]).

Operating pressure is another important trade-off, with typical aircraft operating at 3000, 4000 or 5000 psi (Moir [1]). High-pressure systems are common in modern fighters due to their ability to reduce actuator size and weight while offering faster response and higher stiffness. However, they impose greater mechanical stress on components, require tighter tolerances, reject more heat and demand more complex maintenance. Conversely, low-pressure systems are simpler, less costly, and easier to maintain but require bigger components and provide less responsive control.

Pump selection is another key trade-off since hydraulic power requirements change with flight conditions. Low-Mach, low-altitude flight have low hinge moments but high surface deflection rates. This requires high flow and low pressure, favoring variable-displacement pumps which, while not more efficient themselves, save power on a system level. They also have reduced power draw, despite their mechanical complexity and cost. High-Mach, high-altitude conditions have high hinge moments but low surface deflection rates and therefore demand low flow and high pressure. Fixed-displacement pumps would be preferable for those conditions since they offer simplicity, higher efficiency at the pump level and lower cost but are heavier, less efficient at the system level and extract more continuous engine power (Landberg [4]).

4 Fuel Systems

The fuel system performs several key functions, including tank pressurization and inerting, ground and aerial refueling and defueling, supplying fuel to the engine and APU, fuel transfer and jettison, quantity measurement and indication, prevent condensation, and overall fuel management and control (Langton [14]). These are related to the Manage fuel function in the Perform flight operations function diagram (Figure 1). Because fuel is highly combustible, the system's design, placement, operation, and accessibility are all highly critical. The system must be capable of isolating and reconfiguring itself in the event of a fire.

4.1 System Description

Fighter aircraft fuel tanks often have irregular shapes, since regular volumes are usually taken up by other subsystems, as seen in the Gripen's tanks shown in Figure 8. Figure 9 shows another perspective of the architecture of the fuel system. Fuel from the outer tanks is transferred to tank T1, which is maintained full at all times. T1 supplies fuel through a booster pump in the *Negative-G Tank* (NGT), with fuel flow regulated by the *Full Authority Digital Engine Control* (FADEC). Fuel is displaced through the transfer pump, which is located in the *Forward Refueling/Transfer Unit* (FRTU). The *Aft Refueling/Transfer Unit* (ARTU) controls transfer from the wing tanks and the *Drop Tanks* (DT) to the FRTU. The whole system is controlled by the *General Electronic Control Unit* (GECU), which also manages the hydraulic and the environmental control systems described in Sections 3.1 and 6.1 (Holmberg [15]).

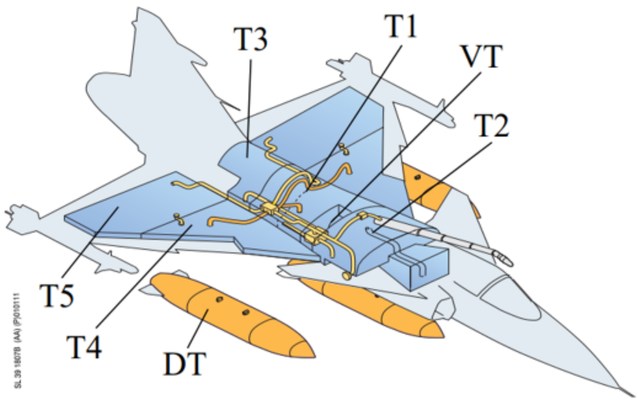

Figure 8: Saab Gripen's fuel system and three optional drop tanks (left, centre and right). Left wing tanks receive numbers T6 and T7. From Kensing [16].

Bleed air coming from the pneumatic system (Section 7) is used to pressurize the tanks. Bleed air is regulated through the *Controlled Vent Unit* (CVU) which adds or vents excess air through the ejector in the *Vent Tank* (VT). Tanks T1 and the NGT are not pressurized to facilitate fuel transfer thanks to the favorable pressure gradient (Kensing [16]).

4.2 System Trade-offs

Fuel system synthesis is directly linked to aircraft-level synthesis and therefore should be considered on early conceptual design (Jackson [2]). A key trade-off lies in the conflicting requirements of minimizing fuel tank volume to reduce weight and cost, while also needing larger tanks to extend range and improve operational flexibility. The tanks should also be located away from elements that could damage them in an otherwise survivable crash and separation should be kept from elements prone to producing sparks. External tanks can extend range but reduce aerodynamic performance, maneuverability, and increase radar signature. Conformal tanks lessen these drawbacks, but cannot be jettisoned by the pilot.

Fuel transfer can be achieved through various pump configurations. An in-line pump setup uses a centrally located pump to draw fuel from multiple tanks. While simple and compact, this design is prone to cavitation, pressure loss, and may

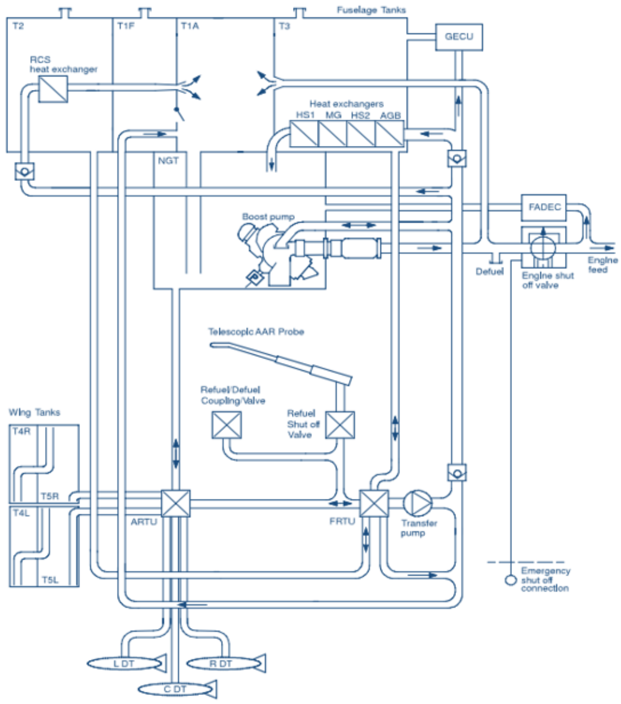

Figure 9: Saab Gripen's fuel transfer system. From Kensing [16].

require an oversized pump to meet system demands. In contrast, distributed pumps are positioned throughout the tanks for fuel transferring, minimizing cavitation risk. Jet pumps enable inter-tank fuel transfer without mechanical or electrical power, but they remain ineffective at low pressures or flow rates. Siphon systems rely on pressure differences to passively transfer fuel, making it the simplest, lightest and cheapest system. However, it is dependent on both atmospheric conditions and the fuel tank pressurization system. Bleed air is the most common source used for tank pressurization but demands high power loads from the engine. Finally, inert gas could be used but requires additional storage cylinders that need replenishment, add mass and take away volume. Because of the high fuel demand of the powerplant, boost feeder pumps may require substantial power to operate. These pumps can be either variable or fixed-displacement, each with trade-offs similar to those of the hydraulic pumps described in Section 3.2. Their energy source is also a trade-off, as they can be driven electrically, pneumatically, or hydraulically.

The refueling process is also subject to major design decisions. Most common receptacles need pressurized feed to function which allows fast refueling but demands either a fuel truck or a ground fuel pump. Gravity-fed systems would allow refueling from austere locations but it is a much slower process that cannot be performed while the aircraft is running or if the fuel system is pressurized. Operational range extension can be achieved through *Air-to-Air Refueling* (AAR), with the associated challenges of adding a telescopic AAR probe that increases drag, complexity, adds weight and reduces space (Gavel [17]).

5 Electrical Systems

The electrical system provides power for avionics, flight controls, weapons systems, and life-support equipment. The system fulfils the Provide power top-level function in the Perform flight operations diagram from Figure 1.

5.1 System Description

The electric power generation and distribution system of the right engine from the Lockheed F-22 Raptor is shown in Figure 10.

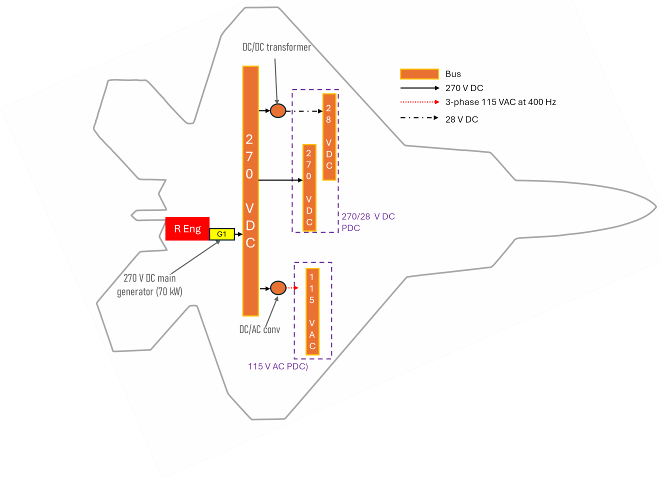

Figure 10: Electric power generation and distribution system in the Lockheed F-22. A mirrored system operates for the left engine. Based on Moir [18].

Electric power is produced by the generator, which transforms mechanical energy into electric energy through electromagnetic induction. The Raptor has one 270 V_{DC} switched reluctance 70 kW generator per engine. Still, many legacy subsystems and components require the conventional 115 V_{AC} and 28 V_{DC} to operate. Therefore, DC/AC converters and DC/DC converters are used to convert power needed in the eight *Power Distribution Centers* (PDC).

In comparison, Figure 11 presents the more electric architecture of the Lockheed F-35's *Electric Power System* (EPS), which is considered part of the *Power and Thermal Management System* (PTMS) discussed in Section 6.1. The EPS is divided between the *Electric Power Generator System* (EPGS) and the *Electric Power Management System* (EPMS).

The F-35 features an element that combines the starter and generator functionalities into a single electromechanical device called the *Engine-Mounted Starter/Generator* (ES/G). The ES/G cranks the engine in its starter mode, and once the required rpm is reached, the ES/G switches from starter to generator delivering two independent 80 kW outputs of variable frequency AC. *Inverter/Converter/Controllers* (ICCs) then convert power into regulated 270 V_{DC}. A DC/AC inverter transforms part of the power into 115 V_{AC} for weapons and the wing folding system (F-35C version). The other branches of electrical power are transformed to 28 V_{DC} by two voltage converters for low-power critical loads. Three *Converter/Regulators* (C/R) provide uninterrupted 28 V_{DC} to each branch of the triple-redundant flight control system's EHA (see Section 2.2). These C/Rs can accept power from three sources: the 28 V_{DC} distribution system, a 28 V_{DC} Li-Ion battery and two *Permanent Magnet Generators* (PMG) (Wiegand [19]).

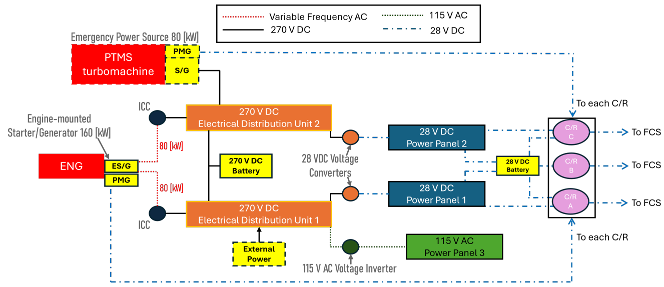

Figure 11: Lockheed F-35 Electric system. Based on Robbins [13].

To balance the dynamically changing loads, *Energy Load Management* (ELM) systems are used to regulate energy flow among providers, consumers, and storage elements. These systems ensure that the power generated at any moment is always equal to or greater than the power consumed. The two main elements that the ELM uses to control electrical loads are a *Generator Control Unit* (GCU), which controls generators' circuit breakers in case of a non-tolerable load and a *Bus Power Control Unit* (BPCU), which manages bus tie breakers between bus bars (Schlabe [20]).

5.2 System Trade-offs

Electric power generation and distribution systems are heavily constrained with conflicting requirements, yet they offer significant flexibility in architectural design.

The type of electric power generation, whether DC or AC, is a top-level requirement that influences the entire architecture. DC power generation has the ability to operate in parallel configuration, allowing for a "no-break" effect. Conversely, AC generators generally have higher power density. Modern aircraft use two main ways to produce constant-frequency AC from variable engine speed. *Integrated Drive Generators* (IDGs) achieve this mechanically via a hydromechanical constant speed drive, which possesses high mechanical complexity, low efficiency and requires constant maintenance. *Variable Speed Constant Frequency* (VSCF) systems use cycloconverters or DC links to regulate output frequency digitally. AC generation's challenges include difficulty of parallelizing and choosing of the proper frequency (Madonna [21]).

The type of power distribution will determine in part the number of power transformations needed. High AC voltages require lower current and have high 400 Hz frequency which lowers voltage drops as well as transformer and motor weight. *More Electric Aircraft* (MEA) have introduced a renewed shift toward high voltage DC distribution systems of 270 V_{DC}. This reduces cable diameter and overall weight but needs increased insulation and separation to mitigate the risk of arcing. Future advancements explore the implementation of *High Voltage Direct Current* (HVDC) distribution networks operating at 540 V_{DC} (Tarisciotti [22]).

Another trade-off lies on how power is distributed and converted. Centralized/bus-level power distribution systems employ large *Transformer-Rectifier Units* (TRUs) and common buses to supply power to loads operating at the same voltage level (see Figure 10). This keeps the buses in maintainable locations and reduces component complexity. Even so, centralized TRUs reject high heat quantities, pose single-point failure risks, and need high-current feeders, resulting in heavy wiring. Another method is the Distributed or *Point of Load* (PoL) system where voltage gets converted very close to each consumer. This generally produces lower cable distribution weight, graceful degradation (no single point of failure) and less voltage transformation steps. Nonetheless, these come with higher maintenance complexity, more difficult EMI shielding and disperse transformation modules that reject heat close to the components.

An important trade-off regarding ELM systems is stability vs. usable generator capacity. Conventionally, each load has a fixed, predefined priority when it comes to reconnection sequencing. If the ELM controls the system to a very narrow and precise operating current that maximizes generator usage, instabilities could arise where the system connects and reconnects frenetically.

6 Environmental Control Systems

The *Environmental Control System* (ECS) conditions atmospheric air to the required temperature, humidity, and pressure before distributing it for cooling purposes. Heat sources include avionics, the propulsion system, power electronics and the cockpit. Heat sinks include ram air, hydraulic fluid, oil, fuel, the structure and *Polyalphaolefin* (PAO). The system performs the Provide environmental control function in the diagram shown in Figure 1.

6.1 System Description

Bleed air is the main air source for cooling in federated systems despite its significant performance penalty on engine efficiency (see Section 7). The amount of heat rejected when conditioning bleed air to be used for cooling has a power-to-removed-heat ratio of around 10:1 [1]. The most common cooling architectures are the air cycle cooling machine/bootstrap systems, which use a compressor-turbine set, a condenser, multiple heat exchangers, valves and a controlling software.

Figure 12 shows a simplified diagram of the bootstrap system from the Saab Gripen. Hot air is extracted from the engine's compressor section or from the APU and rejects heat in the primary heat exchanger, where ram air serves as the cooling medium. The bleed air then enters the compressor stages of the bootstrap unit assembly, referred to as the *Cold Air Unit* (CAU). The air is conditioned and distributed to cooling consumers controlled by the *General Electronic Control Unit* (GECU), which also manages the hydraulic and fuel systems (see Sections 3.1 and 4.1).

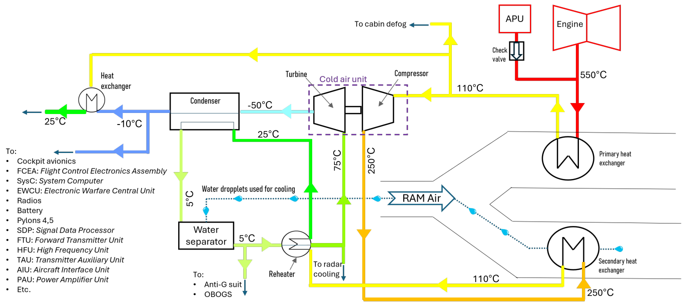

Figure 12: Environmental Control System (ECS) of the JAS Saab 39 Gripen. Temperatures at ∼ 33,000 ft of altitude and Mach 1.5. Based on Hällqvist [23].

In contrast, an example of a MEA closed-loop ECS architecture is the *Power and Thermal Management System* (PTMS) used in the Lockheed F-35 (shown in Figure 13). The PTMS combines into a single system the functions traditionally performed by the *Auxiliary Power Unit* (APU), the *Emergency Power Unit* (EPU) and the ECS. The PTMS covers both the Provide power & Provide environmental control functions from the Perform flight operations function in Figure 1 in a single integrated architecture.

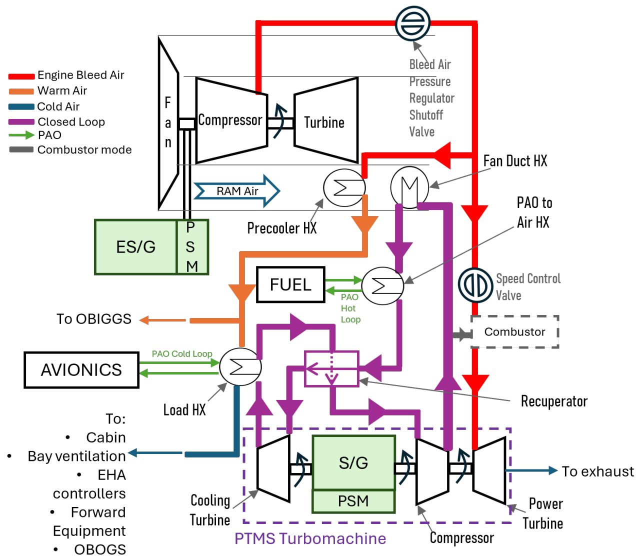

Figure 13: Schematic of the F-35's Power & Thermal Management System (PTMS). Based on Robbins [13].

The PTMS operation centers on its turbomachine, which runs at high rotational speeds to minimize weight and size. This turbomachine is energized by bleed air from the main engine's compressor stages which spins the power turbine. Therefore, this is not a bleedless architecture. The power turbine drives a closed-loop air cycle (purple arrows) through a compressor that is mounted on its same shaft. The cold, low-pressure air that exits the turbomachine's cooling turbine gets conditioned through a series of heat exchangers and then directed to other heat sources. PAO cycles interact with the system to cool down avionics (Robbins [13]).

6.2 System Trade-offs

A key system trade-off involves selecting the air supply source, which can come from a compressor, engine bleed air (see Section 7) or from a ram air intake. Ram air systems offer simple construction and avoid drawing power directly from the engine, but they increase fuel consumption due to drag, perform poorly at low airspeeds, and are dependent on ambient conditions. Routing of air inside the aircraft also takes space and adds piping weight, while the placement of heat exchangers and distribution of heat loads among sinks are also design decisions.

Another key trade-off is selecting between an open-loop or a closed-loop ECS system. Open-loop systems, such as the ECS in the Gripen, use air once and then dump it overboard. While less efficient because of constant engine air extraction, they provide strong instantaneous cooling and require fewer components, resulting in lower maintenance, weight, volume, and cost. Closed-loop systems like the F-35's PTMS recirculate air to conserve energy and decrease engine power off-take. However, the system involves more components and requires complex ducting, making it heavier, costlier, and less compact. This increase in weight may offset the bleed air power off-take from federated systems (Matuschek [24]).

Another design compromise is whether to keep systems separate or integrate them into a single unified architecture. Federated setups with distinct APU, EPU and ECS units result in simpler designs that are easier to maintain, better at isolating faults and easier to re-size independently. However, each power source must be oversized to meet reliability requirements, as systems require separate redundancies, have lower overall efficiency and cannot share power between them. On the other hand, coupled systems like the PTMS improve overall efficiency and reduce weight by sharing hardware and reallocating power as needed. However, they offer poorer fault isolation, greater complexity, higher component costs, and any redesign affects other coupled systems (Renz [25]). For example, the PTMS has struggled to meet cooling demands without placing excessive load on the engine (Norris [26]).

Future solutions for fighter aircraft cooling demands between 62 and 80 kW in the 2029–2032 timeframe are under investigation, as well as electrically driven *Vapor Compression Cycle* (VCS) technologies (Giuffre [27]) and its efficiency increase using precise compressor speed control.

7 Pneumatic Systems

The pneumatic system's primary function is to regulate and maintain air pressure for power and thermal management needs. It is associated with the Provide power function of the Perform flight operations function in Figure 1. The system uses bleed air from the engine compressor to power on-board systems including anti-icing/de-icing, cockpit pressurization, the *Environmental Control System* (ECS), fuel tank pressurization, engine starting among others.

7.1 System Description

An example of a federated pneumatic system is the F/A-18E/F layout shown in Figure 14. Engine pressure and temperature in the compressor stages varies greatly depending on the engine operating conditions. A *High Pressure Shut-Off Valve* (HPSOV) takes air from the *High Pressure Compressor* (HPC) at low engine powers and from the *Low Pressure Compressor* (LPC) at high rpm. Air is then conditioned and routed to the pneumatic power subscribers.

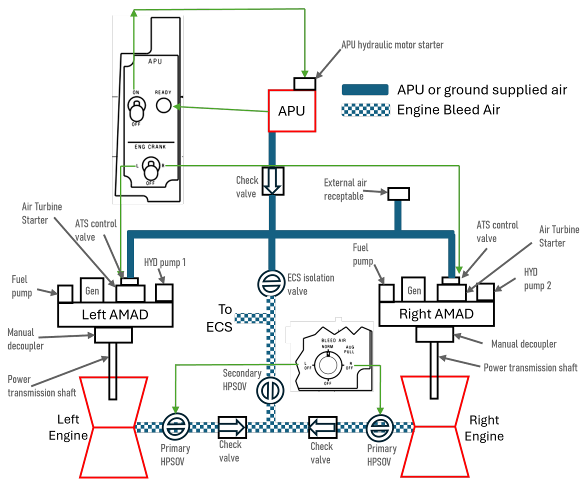

Figure 14: F/A-18E/F pneumatic system diagram. Based on U.S. Naval Air Systems Command [28].

If one engine fails in-flight, cross-bleed air may be used to rotate the non-operating engine's AMAD and retain partial fuel, electrical, and hydraulic system output. Rain protection and defogging are provided by routing hot engine bleed air through the system. Notably, the de-icing systems in combat aircraft such as the F/A-18 and A-10 have no dedicated measures for leading-edge de-icing [28] [29].

7.2 System Trade-offs

Conventional pneumatic systems are versatile and backed by decades of flight-proven experience. Nonetheless they often extract more air than necessary which results in energy losses, reduced engine thrust, and are prone to duct leaks. Bleed air extraction from compressor stages is inefficient since ground tests on an F/A-18 showed that a 1% increase in bleed air extraction can lead to a 2% loss in thrust (Yuhas [30]).

To address this, MEA technology is moving toward bleedless pneumatic architectures, such as the one used in the Boeing 787 where electric compressors replace engine bleed air. Power-by-Wire architectures like this one possess benefits like more efficient power distribution, precise airflow control, and lower maintenance. Nonetheless, these systems are generally heavier, costlier to install, and more complex to make redundant (see Sections 2.2 and 3.2). It is still unclear if the efficiency gains offset the added weight or if bleedless systems are heavier or lighter than federated ones (Cavalcanti [31]).

8 Landing Gear

The landing gear allows shock absorption during touchdown, steering, ground maneuvering, gear retraction and extension and ensures its position is maintained. These functions are related to the Command and control aircraft function from the Perform flight operations diagram in Figure 1.

8.1 System Description

Due to their high operational speeds, the vast majority of modern fighter aircraft are equipped with landing gear retraction mechanisms. The function of these mechanisms is to deploy and stow the landing gear while maintaining its correct position, even under adverse loading conditions. The actuator that performs this must overcome diverse resisting moments, which include the aerodynamic drag produced by the legs after take-off, varying position of the CoG and pre-lock uplock. The uplock is a positive lock that functions independently of the auxiliary power system which retains the landing gear in the retracted position even in case of power loss. The actuator delivers maximum moment at the very end of the cycle (Berry [32]).

The most common type of retraction system uses hydraulic power with alternate sources available in case of failure. This is the case for the Fairchild Republic A-10 Thunderbolt II, whose landing gear system schematic is shown in Figure 15.

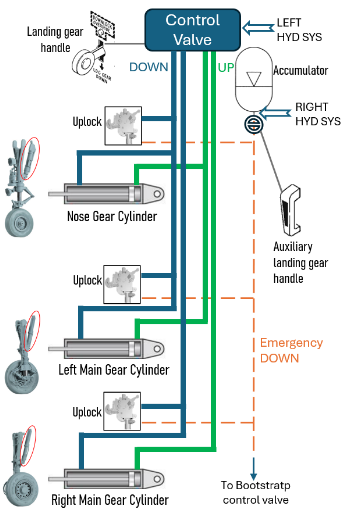

Figure 15: Landing gear system schematic of the Fairchild Republic A-10 Thunderbolt II. Based on USAF [29].

Bound for landing when the landing gear handle is set to "Down" position, the blue system becomes pressurized from the left hydraulic system, the uplocks are released and the hydraulic cylinders are extended. Restriction valves are used to control the speed of landing gear movement. After take-off when the handle is set to UP, the green system retracts the cylinders to the stored position. Just before the end of the cycle, pressure is applied to the braking system to stop rotation before they become stowed on the bays.

In case of control valve or left hydraulic system failure, the auxiliary landing gear extension handle can be pulled, so the right hydraulic system energizes the auxiliary gear extension system (orange dashed). Its main function is to release the uplocks and deploy the gear using gravity and aerodynamic forces. If no hydraulic power is present at all, an accumulator provides the power needed for uplock release. The landing gear doors operate in a coordinated sequence managed by door sequence valves. These valves are hydraulic devices that synchronize the door positions on gear deployment and retraction. Pneumatic landing gear systems are most commonly used as emergency override features, such as the one present in the Lockheed F-22 Raptor (Eaton [33]).

8.2 System Trade-offs

Hydraulic systems offer high energy density and proven reliability, but require piping lines, fluid maintenance and conditioning, and are prone to leaks. Pure electric or EHA retraction systems simplify plumbing, reduce piping weight, and allow distributed placement, yet add motor weight and power electronics heat that can struggle with peak power demands. Fast retraction cycling minimizes drag but drives up actuator power, thermal load, and structural stress. Slower cycles save energy and decrease component wear, though they are detrimental for take-off performance. Designing for high sink rates improves survivability on austere fields and hard landings but requires heavier shock-strut structure and are more expensive. Centralized hydraulic steering guarantees damping and in a light and compact construction, but adds long fluid lines. Local EHA or EMA steering has been investigated by Duval [34] which shortens plumbing, improves modularity, and eases installation, yet must address low power density, shimmy control, heat management and new electrical failure modes.

The braking system is housed in the landing gear and its primary function is to decelerate the aircraft after landing, assure directional control and acting as a parking brake. Selecting the brake actuation method is another key design choice. SHA brakes dominate for their high force density, proven reliability, and straightforward control, but they add plumbing, leakage risk and frequent maintenance. *Electric Brake Actuators* (EBAs) are a type of EMA solution now used on MEA aircraft like the Boeing 787. These eliminate hydraulic piping, have ease servicing, and improve *Foreign Object Damage* (FOD) tolerance, but bring local thermal and size issues. Brakes with EHA can locally supply hydraulic power to components such as bay doors, further reducing the need for piping. However, EHA are heavy, limited by heat dissipation, and requires high-voltage power conversion (Dinca [35]).

9 Auxiliary Power and Emergency Systems

Auxiliary power generation systems provide power to all consumers not directly related to propulsion generation. The need to supply power during failures links these systems to emergency systems whose main function is to assure that flight-critical functions in the Perform flight operations function of Figure 1 continue to be fulfiled should a failure occur. If not possible, emergency systems are responsible for the Provide means of escape and other miscellaneous functions beyond the scope of this study.

9.1 System Description

The Saab Gripen's system serves as an example of a federated auxiliary and emergency power architecture and is shown in Figure 16. The system is divided into three groups: *Aircraft Gear Box & Power Transfer System* (AGB/PTS), the *Auxiliary & Emergency Power System* (AEPS) and the *Auxiliary Power and Engine Starting System* (APESS). These are mainly located on the left side of the aircraft's rear fuselage.

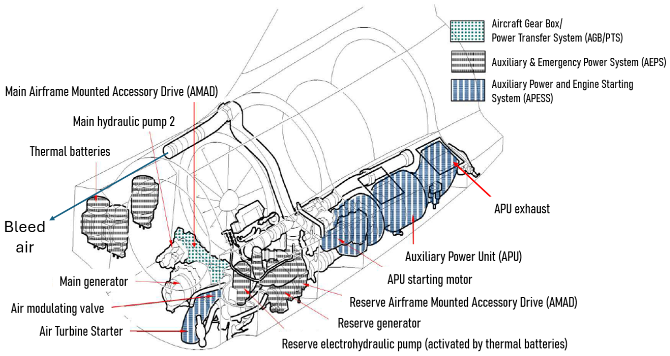

Figure 16: Auxiliary power generation of the Saab Gripen. Translated from Eilersten [36].

The AGB/PTS provides non-propulsive power to the electric and pneumatic systems through the *Airframe Mounted Accessory Drive* which is mechanically operated from the engine through a power transmission shaft (Sections 3, 5 and 7). The *Auxiliary Power & Engine Starting System* (APESS) is shown in Figure 17.

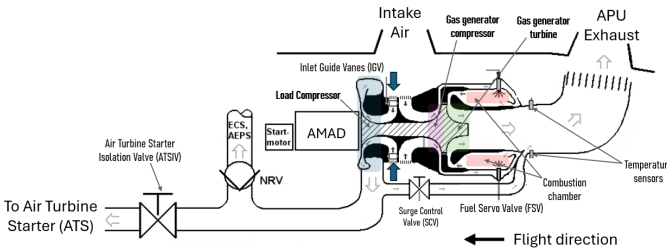

Figure 17: Auxiliary Power & Engine Starting System (APESS). Translated and modified from Andersson [37].

It is centered around the *Auxiliary Power Unit* (APU) and aids in engine starting and supplying electrical, pneumatic, and hydraulic power when the engine is inactive. It provides compressed air for the *Auxiliary & Emergency Power System* (AEPS), the ECS and other consumers. The APU, illustrated in Figure 17, consists of two main sections: the gas generator and the load compressor. The gas generator is made up of a centrifugal compressor (purple) and a radial turbine (green) where air entering the system through the intake is routed to the compressor and then to the combustion chamber (red) where fuel is injected and controlled by the *Fuel Servo Valve* (FSV). The hot, high pressure air expands on the gas generator turbine (green) and exits through the exhaust. The gas generator powers the load compressor (blue), which shares the same inlet and shaft. While the gas generator produces the motive energy, the load compressor has the function of delivering pressurized air from the APU to various auxiliary power consumers. During engine start, air from the APU is directed to the *Air Turbine Starter* (ATS), which is a radial turbine that when driving the main AMAD spins the main engine for startup (see also Figure 7).

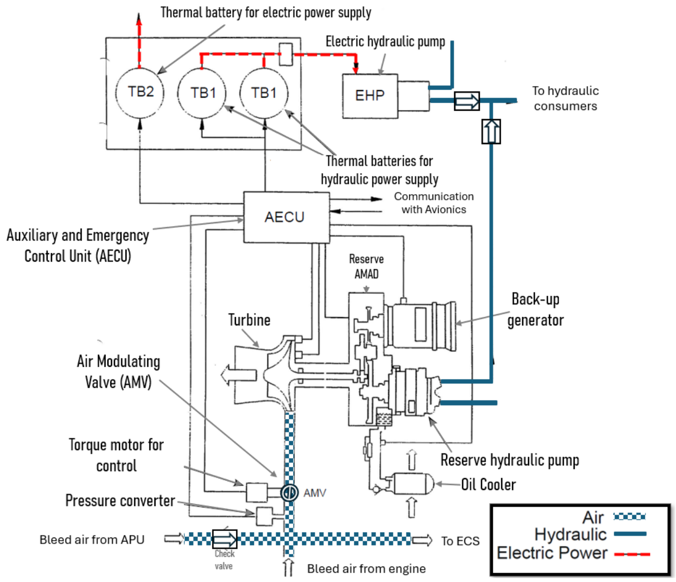

Figure 18: Auxiliary & Emergency Power System (AEPS). Translated and modified from Eilertsen [36].

Another section of the system is the *Auxiliary & Emergency Power System* (AEPS, illustrated in Figure 18) whose main function is to serve as a backup source of secondary auxiliary power. The AEPS has a single-stage radial turbine that is driven by airflow from the main engine or the APU. The turbine connects to a reserve AMAD, which energizes an auxiliary hydraulic pump and a back-up generator if the main AMAD fails. If both the engine and APU fail, thermal batteries supply emergency electrical power and operate an *Electrical Hydraulic Pump* (EHP) to partially pressurize the hydraulic system 1 (also shown in Figure 7). Accumulators are also used for limited emergency hydraulic power provision. The system is controlled by the *Auxiliary and Emergency Control Unit* (AECU) which communicates with other avionics to determine whether auxiliary or emergency power should be utilized. The AECU also executes various system tests during initialization and operation (Eilertsen[36]).

9.2 System Trade-offs

An early design decision is whether to include an emergency capability or not, typically based on risk and safety analyses. Those analyses determine if failure modes have sufficient probability of occurrence and if the consequences could be sufficiently severe (Moir [1]). To prevent failure propagation, the emergency system must be physically, digitally, and logically isolated from other systems, thus affecting its physical location on the airframe. Aircraft may include a deployable *Ram Air Turbine* (RAT) as an additional emergency power. This adds redundancy and safety enhancement but consumes space, adds weight, has higher integration complexity and needs maintenance.

Other top-level trade-offs include choosing the type of the emergency power sources. Federated APU and EPU systems remain simpler, easier to isolate in case of fault, and certifiably robust. However they have increased weight, need redundant plumbing and routing, and minimal power source sharing hydraulic and electrical busses are typically isolated. Integrated auxiliary and emergency architectures, such as the F-35's PTMS (described in Section 6.1) offer higher power density and adaptive load management. They also enable power sharing and reduce overall system weight, yet they introduce tight coupling, thermal hotspots, common-failure modes and more challenging certification.

Lithium polymer battery technology development has evolved heavily since the 2010s as reported in Fu [38]. The higher energy density capacity can be used in auxiliary power systems both as an energy source for engine starting (as the AEPS from Figure 18) or as limited backup power. The battery can also be used as an accumulator, storing energy during overvoltage events and supplying power during peak demand in *Vehicular Management Systems* (VMS). Nonetheless, they still have low energy density, degrade after a limited number of cycles, and are expensive. Other chemical compositions that promise a much higher physical limit on their pack-level energy density are being explored such as lithium sulfur, lithium metal and even lithium air (Liu [39]).

10 Conclusion

The study described diverse on-board system architectures and their associated trade-offs to quantify and compare options available to system designers. It identified the top-level aircraft function they fulfil and related it to current research in future trends. The study showed that there exists an interdependence and diversity of options of on-board systems that can grow rapidly and become complex. This highlights the need for robust, systematic, and holistic power management integration frameworks that evaluate system architectures collectively rather than in isolation. There is also a clear need to develop methods for supporting decisions at the on-board system level and understand their impact on aircraft top-level requirements. These studies at the on-board system level would support aircraft conceptual design, ultimately enabling the development of a balanced overall system architecture.

References

[1] Ian Moir and Allan Seabridge. *Aircraft Systems: Mechanical, Electrical, and Avionics Subsystems Integration*. Wiley, Chichester, UK, 3rd edition, 2008.

[2] Scott Jackson. *Systems Engineering for Commercial Aircraft*. Ashgate Publishing, Aldershot, UK, 2015.

[3] Robert Hillgren. Carefree manoeuvring and automatic return to normal flight envelope jas 39 gripen. volume 40, pages 115–120, 06 2007.

[4] Magnus Landberg and Birgitta Lantto. Aircraft hydraulic system technologies. *Presented at the 3rd Workshop on Innovative Engineering for Fluid Power (WIEFP2016)*, 2016.

[5] NASA Langley Research Center. Electromagnetic interference considerations in composite aircraft structures. Technical Report NASA TP-1997-035035, NASA, 1997.

[6] R. T. Britt, J. A. Volk, D. R. Dreim, and K. A. Applewhite. Aeroservoelastic characteristics of the b-2 bomber and implications for future large aircraft. In *RTO AVT Specialists' Meeting on Structural Aspects of Flexible Aircraft Control*, number ADP010486. Northrop-Grumman Corporation, Defense Technical Information Center, October 2000.

[7] Rodrigo Simoes Lopes Junior, Dimitri Oliveira e Silva, Marcos Paulo Nostrani, Alessandro Dell'Amico, Petter Krus, and Victor Juliano De Negri. A comparative analysis of innovative digital hydraulic actuators for primary flight control. In *The 33rd Congress of the International Council of the Aeronautical Sciences (ICAS 2022)*, Stockholm, Sweden, September 2022.

[8] Alessandro Dell'Amico and Christopher Reichenwallner. A conceptual comparison of hydraulic and electric actuation systems for a generic fighter aircraft. *Aerospace*, 12(1), 2025.

[9] Scott Delbecq. *Knowledge-Based Multidisciplinary Sizing and Optimization of Embedded Mechatronic Systems - Application to Aerospace Electro-Mechanical Actuation Systems*. PhD thesis, 11 2018.

[10] K. W. Vieten. High performance fighter fly-by-wire flight control actuation system. In *1992 Aerospace Design Conference*, Irvine, CA, February 3-6 1992. AIAA. AIAA 92-1123.

[11] Jan Roskam. *Airplane Design: Part IV: Layout Design of Landing Gear and Systems*. DARcorporation, Lawrence, Kansas, USA, 2000.

[12] Jean-Charles Maré and Jian Fu. Review on signal-by-wire and power-by-wire actuation for more electric aircraft. *Chinese Journal of Aeronautics*, 30(3):857–870, 2017.

[13] Drew Robbins, John Bobalik, David De Stena, Ken Plag, Keith Rail, and Ken Wall. F-35 subsystems design, development & verification. In *2018 Aviation Technology, Integration, and Operations Conference*, Atlanta, Georgia, June 25-29 2018. AIAA.

[14] Roy Langton, Chuck Clark, Martin Hewitt, and Lonnie Richards. *Aircraft Fuel Systems*. Wiley, Chichester, UK, 2009.

[15] Anders Holmberg and Per-Erik Eriksson. Decision support system for fault isolation of jas 39 gripen: Development and implementation. Master's thesis,

Linköping University, Department of Electrical Engineering, Linköping, Sweden, June 2006. Reg nr: LiTH-ISY-EX– 06/3839–SE.

[16] Vibeke Kensing. Black-box model development of the jas 39 gripen fuel tank pressurization system: Intended for a model-based diagnosis system. Master's thesis, Linköping University, Department of Electrical Engineering, Division of Automatic Control, Linköping, Sweden, December 2002. Reg. nr: LiTH-ISY-EX-3294-2002.

[17] Hampus Gavel. *On aircraft fuel systems: conceptual design and modeling*. PhD thesis, 01 2007.

[18] I. Moir and A. G. Seabridge. In *Military Avionics Systems*, pages 399–401. Wiley, New York, NY, USA, 2006.

[19] Chris Wiegand, Bruce A. Bullick, Jeffrey A. Catt, Jeffrey W. Hamstra, Greg P. Walker, and Steve Wurth. Key air vehicle and propulsion system technologies of the lockheed martin f-35 lightning ii. Atlanta, Georgia, June 2018. AIAA.

[20] Daniel Schlabe and Jens Lienig. Energy management of aircraft electrical systems - state of the art and further directions. 10 2012.

[21] V. Madonna, P. Giangrande, and M. Galea. Electrical power generation in aircraft: Review, challenges, and opportunities. *IEEE Transactions on Transportation Electrification*, 2018.

[22] L. Tarisciotti, A. Costabeber, C. Linglin, A. Walker, and M. Galea. Evaluation of isolated dc/dc converter topologies for future hvdc aerospace microgrids. In *Proceedings of the IEEE Energy Conversion Congress and Exposition (ECCE)*, pages 2238–2245, October 2017.

[23] Robert Hällqvist, Magnus Eek, Ingela Lind, and Hampus Gavel. Validation techniques applied on the saab gripen fighter environmental control system model. pages 199–206, October 2015.

[24] Tomasz Matuschek, Jannik Häßy, Marc Schmelcher, and Alexander Görtz. Performance assessment of an engine-integrated closed-air cooling thermal management system in a next generation fighter configuration. In *ICAS 2024 Conference*, German Aerospace Center (DLR), Linder Höhe, 51147 Köln, Germany, 2024.

[25] David D. Renz. Intersociety energy conversion engineering conference paper, 1992. NASA Technical Memorandum 105852.

[26] Steve Trimble. An insider's view of options to fix the f-35's cooling crisis. *Aviation Week Network*, June 2023. Accessed: 2025-05-13.

[27] Andrea Giuffre, Federica Ascione, Piero Colonna, and Carlo De Servi. Integrated design optimization of environmental control systems for next-generation aircraft. *Journal of Aircraft*, 62:1–18, 02 2025.

[28] Naval Air Systems Command (NAVAIR). *NATOPS Flight Manual, Navy Model F/A-18E/F (165533 and Up Aircraft)*. United States Navy, 2008.

[29] United States Air Force (USAF). *Flight Manual: USAF Series A-10A Aircraft (Serno 75-00258 and Subsequent)*. McClellan AFB, CA, USA, change 8 - 15 march 1988 edition, 1983. Accessed: March 18, 2025.

[30] Andrew J. Yuhas and Ronald J. Ray. Effects of bleed air extraction on thrust levels of the f404-ge-400 turbofan engine. NASA Technical Memorandum NASA TM-104247, NASA Dryden Flight Research Facility, Edwards, California, 1992.

[31] Victor Cavalcanti and Claudia Andrade. A trade-off study of a bleedless and conventional air conditioning systems. 10 2008.

[32] Patrick Berry. *Aircraft Conceptual Design Methods*. Licentiate thesis, Linköping University, Department of Mechanical Engineering, Fluid and Mechanical Systems, Linköping, Sweden, 2005. ISSN 0280-7971; Licentiate thesis (composite); In English.

[33] Eaton's Aerospace Group Product Capabilities. Lockheed martin/boeing f/a-22 raptor component systems overview, 2014.

[34] M. N. Duval and T. Lammering. Development of an electro-hydrostatic nose landing gear actuation and steering system. In *The 34th Congress of the International Council of the Aeronautical Sciences (ICAS)*, Florence, Italy, September 2024.

[35] Liviu Dinca, Jenica-Ileana Corcau, and Dmitrii Condrea. Modern aircraft braking systems. In *Scientific Research and Education in the Air Force – 24th AFASES International Conference*, Brașov, Romania, May 2023. Henri Coandă Air Force Academy.

[36] Adrian Eilertsen. Modellering av reserv- och nödkraftsystem i jas 39 gripen. Master's thesis, Linköping University, Department of Electrical Engineering, Linköping, Sweden, 2010. Reg. nr: LiTH-ISY-EX–10/4438–SE.

[37] Karl Andersson. Modellering av kontrollenhet för jas39 gripens hjälpkraftsystem. Master's thesis, Tekniska högskolan i Linköping, Linköping, 2007. Reg. nr: LITH-ISY-EX–07/3951–SE.

[38] P. Fu, L. Zhao, X. Wang, J. Sun, and Z. Xin. A review of cooling technologies in lithium-ion power battery thermal management systems for new energy vehicles. *Processes*, 11(12):3450, 2023.

[39] Yaolong Liu, Siyu Zhang, Jiechao Zhang, Kexuan Yao, Mingqiang Luo, and Yantao Liu. Towards the design of future aircraft: A critical review on the tools and methodologies. *Aerospace Research Communications*, 3(2):13096, 2024.