Aicraft-0 SAT D1 for greener, safer and affordable More Electric (MEA) and Fly-by-Wire (FbW) Small Air Transport (SAT)

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Abstract

The ICAS 2024 proceedings focus on the innovative Aircraft-0 SAT D1 “iron bird” in the framework of Clean Sky 2 SAT TA project, aimed at advancing More Electric Aircraft (MEA) and Fly-by-Wire (FbW) Small Air Transport (SAT) systems. This research emphasizes integrating and validating technologies up to Technology Readiness Level (TRL) 5, covering Electrical Power Generation and Distribution Systems, FbW Flight Control Systems, and Electromechanical Actuators for landing gear and brakes. The project showcases significant advancements, including testing capabilities, industry accessibility, and reductions in fuel consumption and operational costs, paving the way for a sustainable future in aviation.

Keywords: More Electric Aircraft, Fly-by-Wire, Small Air Transport, Electromechanical Actuators, Electrical Power Systems

1. Introduction

The advent of More Electric Aircraft (MEA) represents a disruptive shift in civil aviation, transitioning all onboard power systems to electrical formats. This transition offers substantial benefits, including performance and life-cycle cost optimization and enhanced reliability and safety. However, adopting MEA involves overcoming challenges related to the replacement of traditional pneumatic, mechanic, and hydraulic systems with digital electrical alternatives.

This shift applies to both large and small aircraft, albeit with some key distinctions. For large commercial airplanes, replacing these legacy systems with electrically driven actuators offers significant weight and fuel consumption reduction benefits. Additionally, digital control systems enable more precise and efficient operation of flight control surfaces, landing gear, and other critical components.

In contrast, the focus for small MEA often leans towards operational simplicity and improved safety. Replacing hydraulics with electric motors eliminates the need for complex fluid management systems, potentially reducing maintenance costs. Digital control systems can also simplify pilot interfaces and improve automation capabilities. A key challenge for small electric aircraft remains the limited power density of current electrical power generation and distribution technology, that must be improved as key enabler. However, challenges arise in ensuring the redundancy and safety of these new, complex digital architectures for flight-critical functions.

The Clean Sky 2 Small Air Transport Transversal Activities (SAT TA) framework’s final demonstrator, namely Aircraft-0 SAT D1 “iron bird” is a pioneering initiative aimed at advancing MEA and Fly-by-Wire (FbW) technologies for SAT TA. The project’s primary objective is to validate the integrated maturity of MEA and FbW technologies up to Technology Readiness Level (TRL) 5, contributing to a greener, safer, and more economical general and business aviation future.

Piaggio Aerospace has selected the P.180 Avanti aircraft as its testbed for a comprehensive upgrade program centred on advanced digital electric systems. This research focuses on modernizing critical components to enhance overall aircraft performance and safety. The core upgrades include:

- A 28/270V DC Electrical Power Generation and Distribution System (EPGDS): This improved system optimizes power management for the modernized aircraft.

- A FbW Flight Control System (FCS) with Smart Air Data Sensors, Electro-Mechanical Actuators (EMAs), and Flight Control Computers (FCCs) on an Avionics Full-Duplex Switched Ethernet Network (AFDX™): This integrated system replaces traditional mechanical controls with a digital, electrically driven architecture for enhanced precision, manoeuvrability, and reduced pilot workload. The FCCs communicate via the high-speed AFDX™ for reliable data exchange.
- Electric Landing Gear (LG) and Brakes with EMAs: This upgrade replaces the hydraulic system with electric actuators, simplifying maintenance and potentially reducing weight.

The project adopts a systematic approach, addressing the integration aspects of each upgrade individually before evaluating their combined effect on the P.180. This ensures a thorough understanding of each modernization's impact and facilitates a smooth integration process.

However, the high level of system integration inherent to the project introduces a heightened risk profile. The interdependence between various aircraft systems necessitates meticulous control consistency to achieve complete flight safety. As error is inherent in the development of such complex systems, sophisticated testing activities using digital simulation and experimental facilities are essential to mitigate potential risks.

1.1 Electrical Power Generation and Distribution System

The primary functions of the Aircraft Electrical System include the transformation of mechanical energy into electrical energy and the distribution of power for equipment feeding. The anticipated increase in electrical power consumption, due to the adoption of more electric configurations for general and mission systems, necessitates the introduction of high voltage capability. This requirement leads to the development of a high-level functional architecture for power management.

Figure 1 EPGDS Architecture

Figure 2 EPGDS Lab Test Rig

The functional breakdown presented is implemented in the architecture, as depicted in the figures above, which comprises the following equipment:

- Human Machine Interface
- Low Voltage (LV) EPGDS, developed by INDRA in the scope of the CLEANSKY2 INDIS project
- Power Conversion
- High Voltage (HV) EPGDS, Distribution
- EPGDS-A/C Interface

HV/LV Distribution was developed by INDRA and HV Generation by Thales in the framework of CleanSky2.

The new EPGDS offers enhanced onboard electrical power with dual-engine generators, a dual voltage system (28VDC and 270 VDC) for diverse electrical loads, and improved aircraft safety through multiple power sources management.

It also enhances the electrical system's reliability, maintainability, and testability using new technologies and diagnostic software.

1.2 Fly-by-Wire

The Fly-by-Wire (FbW) system, a significant advancement in the Flight Control Systems (FCSs) of small aircraft, replaces traditional mechanical linkages with electrical signals. This system interprets pilot control inputs as desired outcomes and calculates the necessary movable surface position, obtained through dedicated actuators controlled by the FbW system itself.

Despite the advantages of FbW systems, such as flight envelope protection, advanced autopilots and automation, reduced maintenance costs, improved natural aircraft dynamic behavior, and reduced drag due to optimized trim setting of controls, the specific needs of Small Air Transport (SAT) have proven to be a significant issue in the scaling process of the system toward a small aircraft.

The FbW system has three main functions: surface actuation, sensing, and computation. These functions are performed by Electro-Mechanical Actuators (EMAs) for elevator, rudder and ailerons control surfaces, inertial sensor, and air data probes (ADPs), and flight control computers (FCCs), respectively. Each equipment has a redundancy level and multiplicity chosen in function of aircraft safety requirements.

The backbone of the system is the high speed AFDX™ bus that permits a fast and reliable communication channel between the FCS units. AFDX™ is a safety-critical and deterministic Ethernet network specified by the ARINC – Aeronautical Radio Inc. – 664 part 7).

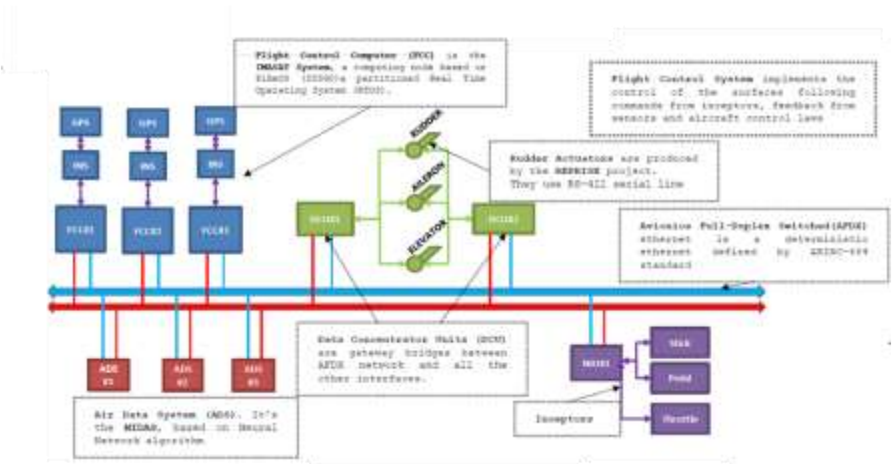

Figure 3 Fly-by-Wire Architecture

Electro-Mechanical Actuators

The A/C 0 Flight Control Demand System employs Electromechanical Actuators (EMAs) to provide control commands to the Flight Control Actuation System. Each actuator in this system comprises the following components:

- Electronic Control Unit (ECU): Provides the inner loop of actuation control.
- Electromechanical Actuator (EMA): Generates the actuation force.
- Kinematic Chain: Transfers the actuation displacement to the surface.

Rudder Actuation System

The EMA for the rudder is a fault-tolerant differential direct drive actuator, based on a unique system patented by UMBRAGROUP spa, developed in the scope of the CLEANSKY2 REPRISE project in collaboration with Università degli Studi di Bergamo and Zettlex. Its primary feature is the utilization of two independent electric motors and two independent electronics that drive fully redundant mechanical components, eliminating the need for differential gear boxes.

The ECU controls the EMA and, in addition to the power supply board, comprises two boards in a Control (CON) Monitor (MON) architecture that hosts software for controlling the EMA. Each board controls an electric motor, ensuring redundancy in the actuation system. The ECU communicates with the Flight Control System (FCS) via two RS-422 digital interfaces, one for each board, and with the EMA via analogue I/O.

Aileron Actuation System

The system incorporates the Commercial Off-The-Shelf (COTS) analogue servo actuator. This actuator plays a significant role in the aileron actuation system. For both the left and right ailerons, the system includes 2x EMA, providing redundancy, ensuring reliability and robustness in the system's operation.

Elevator Actuation System

Like the aileron actuation system, the elevator actuation system incorporates the same COTS servo actuators. The system includes 2 independent EMA per each movable surface of the elevator.

Horizontal Stabilizer Actuation System

The Horizontal Tail Trim Actuator (HTTA) is responsible for generating actuation displacement for the horizontal trim surface.

Air Data System

The Air Data System (ADS) has been developed in the CleanSky2 Modular and Integrated Digital probe for SAT aircraft Air Data System (MIDAS) project, by Selt srl, Politecnico di Torino and Istituto Nazionale di Ricerca Metrologica (INRiM). MIDAS integrates a Total Air Temperature (TAT) probe and a Pitot probe and provides an AFDX^{TM} interface for communicating with the FCC.

It is a synthetic sensor that computes the Angle of Attack (AoA) and Angle of Sideslip (AoS) by means a neural network, starting from pressure, temperature and dynamic information of the A/C received from FCC.The computing algorithms are implemented in the Field Programmable Gate Array (FPGA).

FbW Computing

The FbW computing is based on the Flight Control Computer (FCC) that has been developed within Clean Sky 2 Integrated Modular Avionics for SAT (IMASAT) Project, by Aertec Solutions sl and Clue Technologies sl.

The purpose of the project was to provide a Computing Node designed to guarantee reduced weight, volume, and power dissipation ready to be integrated within a dual Avionic Full Duplex (AFDX^{TM}) switched Ethernet Architecture.

Inertial/Position System

The system includes three Inertial Navigation Sensors (INS), each of which provides a six-degree-of-freedom (6-DOF) attitude and rates measurements. It also comprises three Global Positioning System (GPS) receivers. These receivers are responsible for providing geolocation data.

1.3 Landing Gear

Leg Actuation

The primary functions assigned to the Landing Gear Actuation Systems, which facilitate the alteration of the aircraft configuration through the extraction and retraction of the Nose and Main Landing Gear Legs, remain unaffected by the system revision proposed within the Clean Sky2 framework. However, the design solution has been comprehensively revised to meet the requirements of a fully electric power supply. The preceding functional architecture is realized in the subsequent equipment architecture:

- The Landing Gear Command (e.g., cockpit push button) executes the pilot's command for the extraction and retraction of the tricycle landing gear. The choice between the Landing Gear UP/DOWN command is determined by the manual command provided from the pilot's cockpit.

- Three Electronic Control Units (ECUs) are included, each implementing the actuation control function for the Left Main Landing Gear (MLG) leg, Right Main Landing Gear (MLG) leg, and Nose Landing Gear (NLG) leg, respectively.
- Three Electromechanical Actuators (EMAs) are incorporated, each performing the actuation function for the left-handed Main Landing Gear (MLG) leg, right-handed Main Landing Gear (MLG) leg, and Nose Landing Gear (NLG) leg, respectively.
- Three Kinematic Chains are employed to transform the LH MLG/RH MLG/NLG actuation into leg extraction/retraction movement.
- Two MLG Sensing Systems are used to monitor the weight on the wheel of the LH/RH MLG.
- The Avionics are designed to implement a dedicated MFD Synoptic Page that integrates visual feedback for the pilot's system control loop closure.

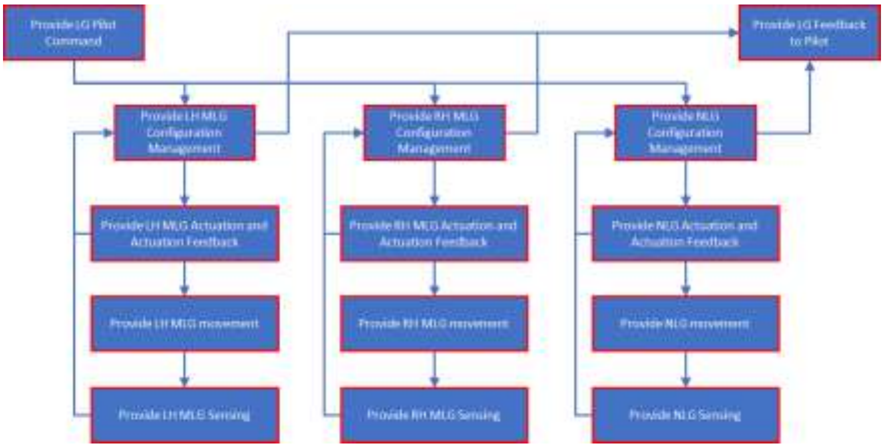

Figure 4: Landing Gear Test Bench and System Functional Breakdown

The project Fail-Safe Electro-mechanical actuation for Landing Gear (FASE-LAG) of ITD Systems provides the electrical solution for the landing gear system. The project was coordinated by UMBRAGROUP, supported by Fundación Tecnalia Research and Innovation, ItalSystem S.R.L, VGA srl, and Magnaghi Aeronautica SpA.

Brake

Actuation for on-ground aircraft handling, deceleration and steering (differential brake), is provided by the implementation of four EMAs on each main landing gear wheel brake managed by a Brake Control Unit (BCU) installed on baggage compartment.

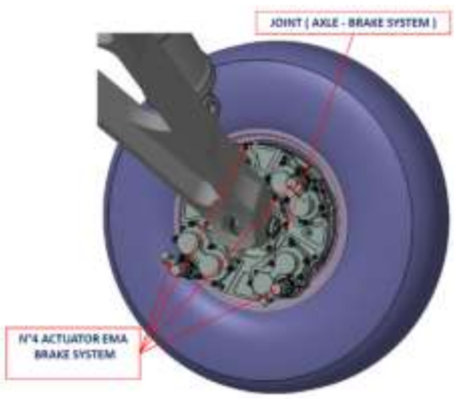

Figure 5: E-Brake EMAs installation details

This electro-mechanical braking system for small aircraft has been developed in the Project E-BRAKE of ITD Systems coordinated by Mare Group spa (MARE) in partnership between University of Perugia, Umbria Aerospace Systems spa (UAS), Magnaghi Group spa (MA) and CRDC Tecnologie scarl (CRDC).

1.4 P180-based Aircraft Flight Simulator

The 6Dof P180-based Aircraft Flight Simulator encompasses all the requisite facilities to drive, control, and display the simulated segment of the FbW “Hardware-In-the Loop” (HIL) testing, and circulate the necessary data to close the simulation loop. In conjunction with the FbW Test Rig development program, the P180-based Aircraft Flight Simulator was developed with the objective of providing an integrated and advanced simulation environment, thereby enabling the execution of soft real-time closed-loop tests. The core software of the Flight Simulator is predicated on a customized version of the FlightGear Flight Simulator (FGFS). This hosts the JSBSim 6DoF dynamics simulation and incorporates an embedded code in the PXI RT controller to acquire input signals from the Cockpit Simulator (CS) and feedback signals from the Test Rig and propagate them back to the simulation environment. The 6Dof P180-based Aircraft Flight Simulator, which can be connected via an Ethernet local area network to the FbW Test Rig to share the simulated flight parameters, is predicated on a suite of software tools. These tools manage the data of the simulated environment (geographic area, environmental conditions) in a realistic and consistent manner to ensure the correlation of sensor images and data, thereby providing the crew with an integrated and coherent mission simulated environment. The P180-based Aircraft Flight Simulator can reproduce malfunctions or introducing noises on the simulated systems under test, and realistically simulating the system behaviors and the scenario in terms of weather conditions and the geographic database. In the remainder of this document, the use of the 6Dof P180-based Aircraft Flight Simulator will be referred to as “SIMPAS”. The subsequent figure illustrates the architecture of the SIMPAS Flight Simulator.

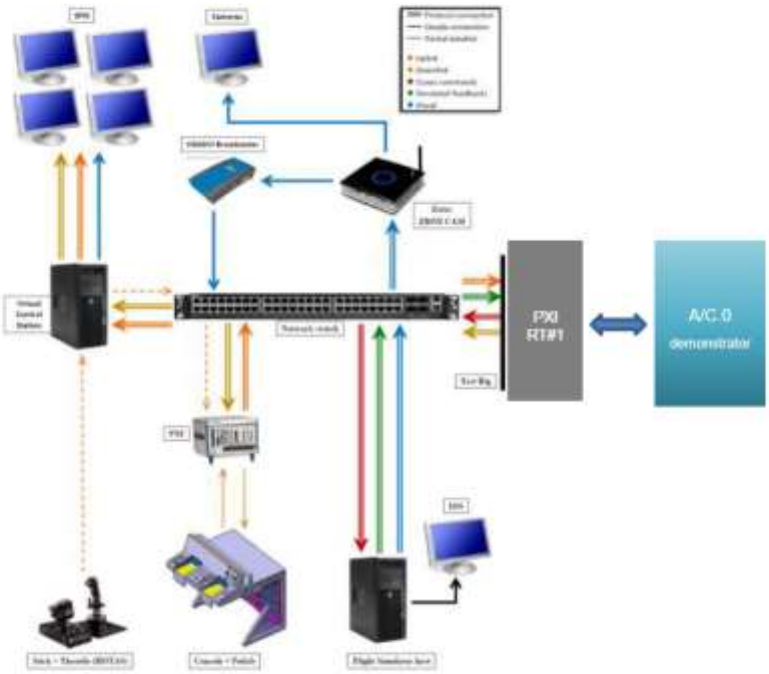

Figure 6 Flight Simulator hardware architecture in FbW HIL setup

1.5 Test results

A complete flight simulation on the Iron Bird aircraft offered the opportunity to replicate the real flight conditions with remarkable precision, providing invaluable insights for design validation and verification of the systems, hardware, and software in a representative environment.

Nominal Flight Condition

The first flight testing campaign has been performed simulating a nominal flight without failures (take off, climb, cruise, descent, and landing) to check the proper functionalities of the systems: FbW, EPGDS and Landing Gear, that represent the hardware in the loop. Meanwhile, the other onboard systems have been simulated through virtual loads: Environmental Control system , pressurization, avionics, de-ice system, etc. The main recorded parameters during the flight have been: Calibrated

Air Speed (CAS), GPS height, Landing gear position and weight on wheel percentage.

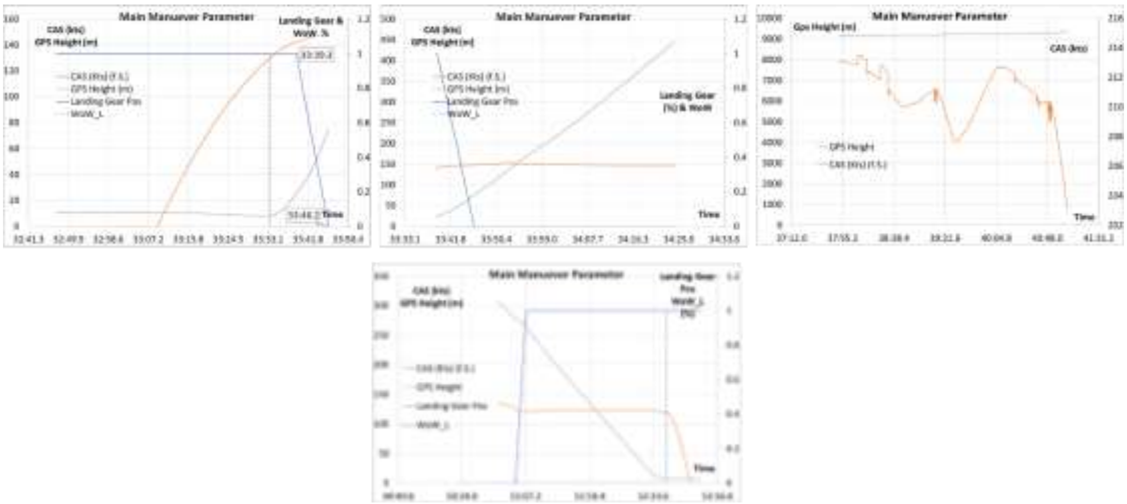

Figure 7 Main maneuvers parameter for nominal flight phases (Take-off, Climb, Cruise, Descent, Landing)

Failure Condition

The second Simulated Flight Test Campaign was executed by injecting failures into the EPGDS, Fly-by-Wire (FbW), and Landing Gear System to assess the resilience of the aircraft to potential malfunction occurrence.

Electrical Power Generation and Distribution Failures

Specific Failures were injected on the electrical generation or distribution and for both high voltage and low voltage side.

Two test cases are illustrated in the follow as example.

Low Voltage Failure

A failure was simulated on the left side of the low voltage primary distribution unit. This failure was expected to result in a loss of nominal power feeding lines for all equipment, without any functional degradation on the system behavior.

The test case was performed during a straight and level cruise. Graphs were provided to illustrate airspeed and altitude traces. Prior to the execution of the doublet, an electrical failure was simulated that resulted in zero voltage on the left buses. During the maneuver, several doublet commands were executed on the three axes. The graphs illustrate that the longitudinal doublet command corresponds to a damping in 5 periods of pitch. The roll command is damped in 4 periods, and the yaw command in 5 periods. It was observed that the yaw command induces a predictable delayed roll oscillation, which is damped in accordance with that of the yaw. The graph displaying the control surface positions (RVDT) confirms their complete operation during the flight. The body accelerations graph indicates that the aircraft's maneuverability was maintained, as were the contained values of the angle of attack and sideslip, and the body angles. Aircraft stability and handling were confirmed in the presence of the voltage primary power distribution unit section failure. A second prescribed cruise condition was flown to verify the retraction of the landing gear. As observed in the graph, the extension of the landing gear was successfully performed. The test was therefore classified as passed.

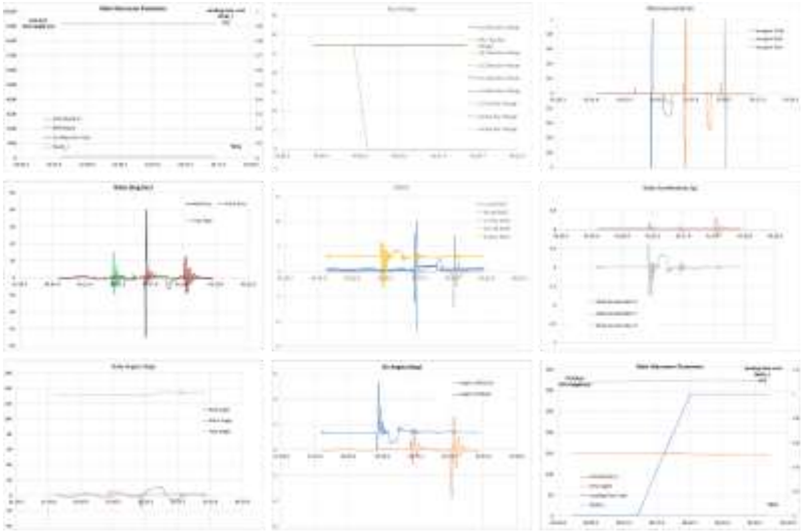

Figure 8 Results for LV Power Distribution Failure test

High Voltage Failure

A second test on Electrical System was conducted during a low-speed and low-altitude cruise to verify availability of opening/closing function of the landing gear. A failure was simulated by turning off one High Voltage generator, after which the opening and closing of the landing gear were tested. The test conditions were met as evidenced by the graphs. The altitude and speed were as expected, and the extraction and retraction of the landing gear were completed in approximately 7 seconds. Despite the single HV power feeding for the Landing Gear EMA, the landing gear functioned correctly. The test was therefore classified as passed.

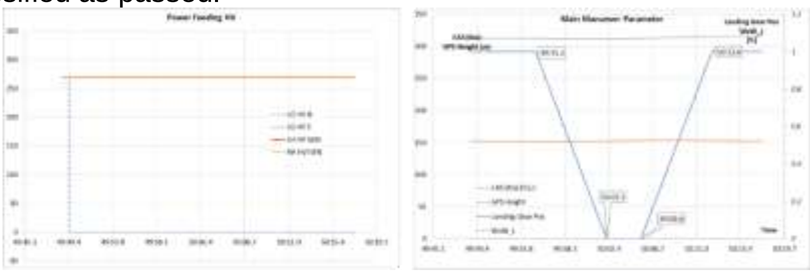

Figure 9 Results for HV Distribution Failure test

Each test was successfully passed, confirming the adequacy of the architecture of the EPGDS, as already demonstrated on SYS projects, and the proper integration with FbW and Landing Gear Systems.

Fly by Wire Failures

Failures were also introduced on the FbW side on single actuation of the primary control surface (Rudder, Aileron, and Elevator) and on a single component of the Computing Platform (DCU, AFDX Switch) during landing simulation with external disturbance (crosswind and/or step command) to assess the stability and handling qualities in such a demanding maneuver.

The test consists in the verification of effects induced by a failure with impact on FCS redundancy power feeding.

Test is performed separately for longitudinal, lateral and directional actuation control and for computing platform equipment, the malfunction of which causes transversal impacts on redundancy of commands on the three axes actuation.

Some examples are provided in the following paragraphs.

Actuation Failure - Rudder case

The test consists of a descent in the presence of crosswind during which a failure was simulated. The aircraft was prepared for landing with the extraction of the flaps and landing gear. The maneuver also included the flare and decrab to manage the crosswind. The landing run concluded with a deceleration until the aircraft came to a stop. The maneuver started at a height of 300 meters and at 150 knots, which were maintained throughout the descent phase. The landing gear was extracted at 220 meters. The flare phase can be observed by the height trace. The first change in CAS trace slope indicates the touchdown and removal of residual thrust, while the second is related to the application of the brake. Despite the failure injected on the rudder control surface redundancy and the crosswind introduced during descent, the FCS allowed adequate aircraft control both in flight and on-ground. The graph on rudder surface deflection shows that the failure on ECU1 did not affect the actuator functionality, leaving complete control authority. The good handling qualities were confirmed by the small pilot command excursion and smooth variation in acceleration, air angles, body angles, and rates. Complete deceleration from first touchdown to complete stop was performed in 980m. The test was therefore classified as passed.

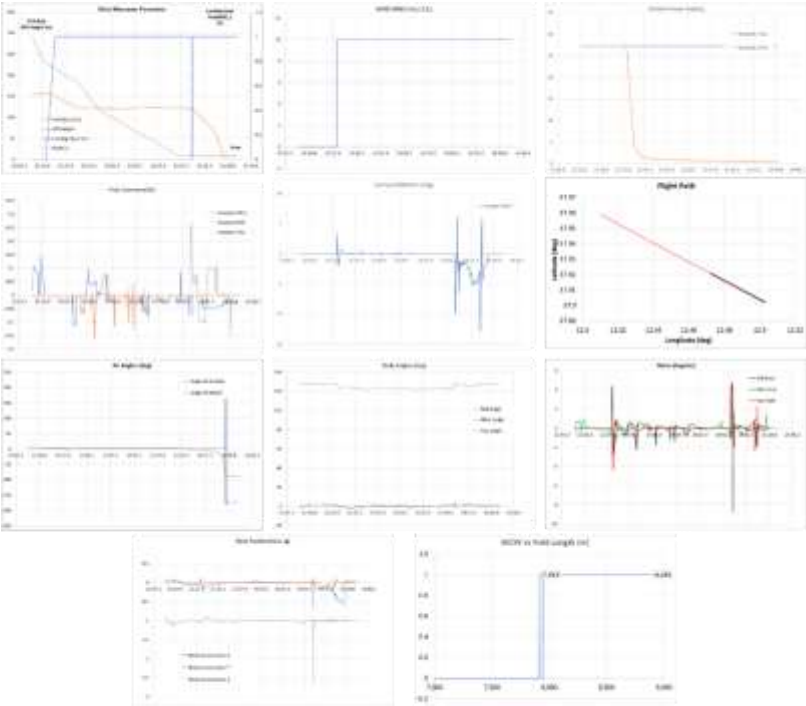

Figure 10 Results for Rudder Failure

Actuation Failure - Elevator case

The manoeuvre began at a height of 290 meters and at 150 knots. The crosswind was applied first, and the failure was injected after the landing gear was deployed (indicated by the red dotted line) at an altitude slightly above 250 meters and at a speed slightly below 130 knots. The loss of the left elevator had two noticeable effects.

The aircraft underwent a pitch down because the failing elevator positioned itself with the trailing edge downwards. This is due to the elevator control surface not being balanced, hence, in the absence of aerodynamic forces, it goes down in case of power feeding interruption. This is clearly seen in the air data graph that measures an instantaneous loss of two degrees of pitch, to which the pilot reacts with a command to pitch up to restore the descent angle.

The aircraft underwent a roll as a secondary aerodynamic effect due to the asymmetry of elevators deflection.

After this manoeuvre, the aircraft appeared perfectly controllable for the rest of the flight, similar to what was verified for the failure of the aileron described above. The landing gear was extracted almost immediately. The flare phase can be appreciated by observing both the height trace and the speed trace, which undergoes a first change in slope. The second change in slope indicates the application of the brake. Please note that after the flare, there was a brief floating of the aircraft due to the less than optimal piloting skill of the test engineer at the controls. The surface deflection graph shows the correct application of the failure. The test was therefore classified as passed. Given the symmetry of the aircraft and actuation system, the test is also considered valid for a failure on the right elevator.

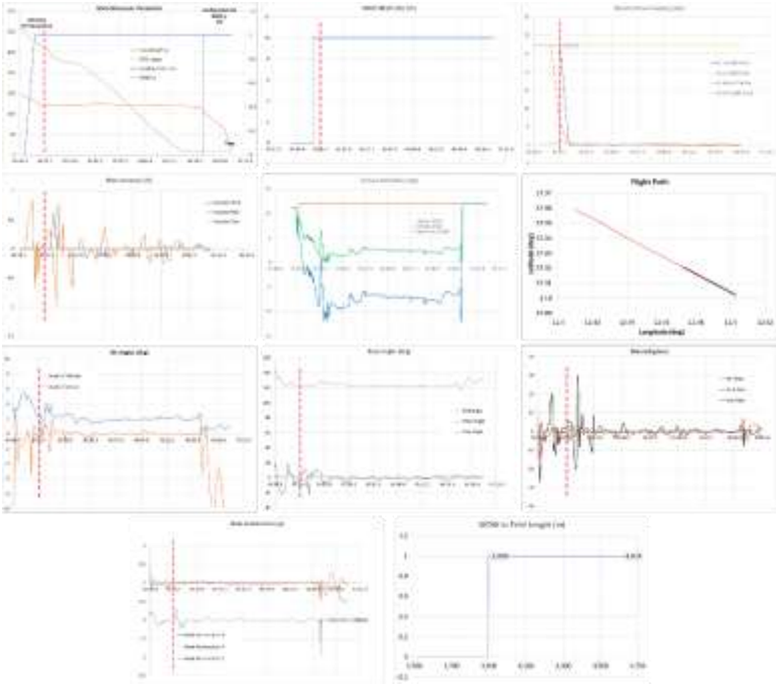

Figure 11 Result for Elevator Left Failure test

Computing Failure - Data Concentrator #1 case

The failure of DCU1 led to the deactivation of one of the servo actuators for both the right and left ailerons (due to a lack of command), as well as the left elevator connected to it. However, the second aileron servo actuators and the right elevator, which are linked to DCU#2, remained operational. The manoeuvre was like the previous ones. From the subsequent graphs, no significant issues were evident. The application of crosswind, as shown in the wind speed graph, was noticeable in both the application of roll control and in the aircraft's accelerations (body accelerations). There was also an oscillation of the sideslip angle at the beginning of the flight. Pilot commands and aircraft angles and rates remained within acceptable limits throughout the entire flight. The manoeuvre did not substantially differ from the previous tests. The power feeding, along with the actuation DCU (both #1 and #2) graphs, indicated the correctness and timing of the failure injection. The test was therefore classified as passed.

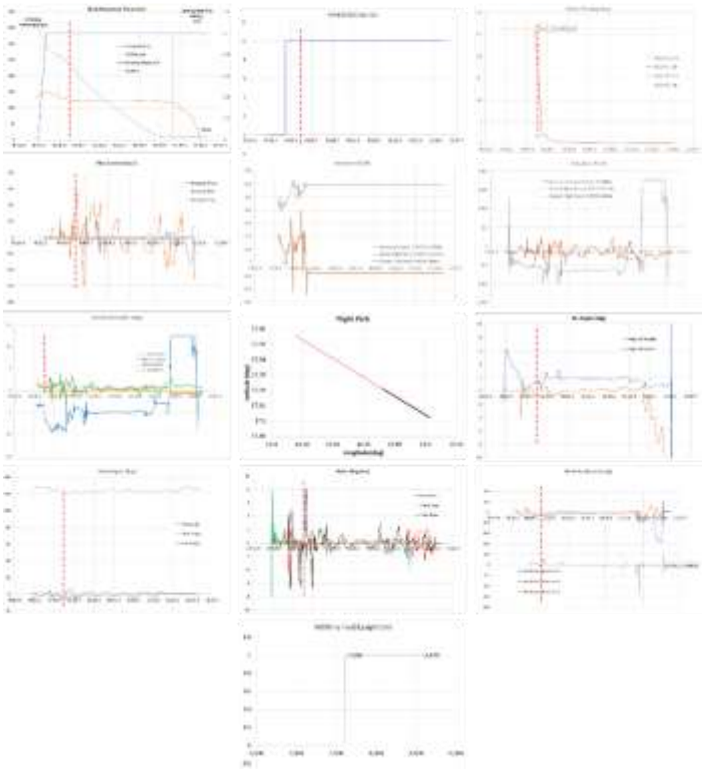

Figure 12 Result for DCU#1 Failure test

Computing Failure - Data Concentrator #2 case

The failure of DCU2 led to the deactivation of the second right and left aileron servo actuators, as well as the right elevator actuator connected to it. However, the first aileron servo actuators and the left elevator actuator, linked to DCU#1, remained operational. The manoeuvre was like the previous ones. From the subsequent graphs, no significant issues were evident. The application of crosswind, as shown in the wind speed graph, was noticeable in both the application of roll control and in the aircraft's accelerations (body accelerations). There was also an oscillation of the sideslip angle at the beginning of the flight. Pilot commands and aircraft angles and rates remained within acceptable limits throughout the entire flight. The manoeuvre did not substantially differ from the previous tests. The power feeding, along with the actuation DCU (both #1 and #2) graphs, indicated the correctness and timing of the failure injection. The aircraft's handling with a single failure on the Flight Control System was demonstrated. The test was therefore classified as passed.

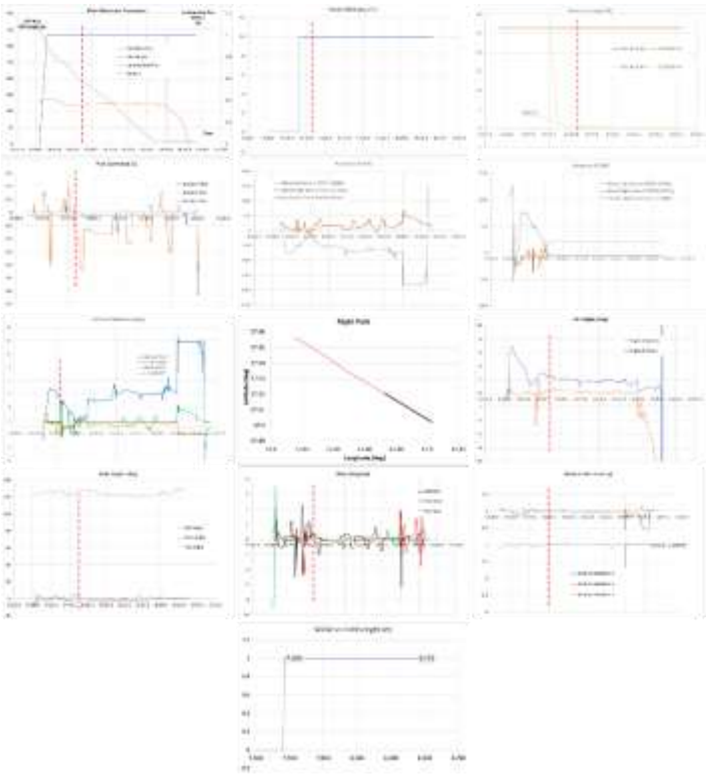

Figure 13 Result for DCU#2 Failure test

Landing Gear Failures

Testing on Landing Gear and Brakes with failure occurrence was performed at the end to complete the ground campaign. For this purpose, the effect of failures on Landing Gear Systems was reproduced as power feeding loss on one of the redundant channels, verifying that the redundancy allows for tolerating a failure without safety effect both on Landing Gear Actuation and Brake, as expected for certifiable systems.

Actuation System

The capability of retraction of the landing gear followed by extraction in a degraded condition due to loss of power feeding to the ECU was demonstrated during a nominal take-off maneuver. The maneuver was properly performed as shown in the subsequent graph. The aircraft rotation started at 125 knots. The landing gear retraction was performed at 65 ft, after the failure injection shown in the LG ECU Power Feeding graph. The proper extraction and retraction of the Landing Gear followed after 7 seconds. The test was therefore classified as PASSED.

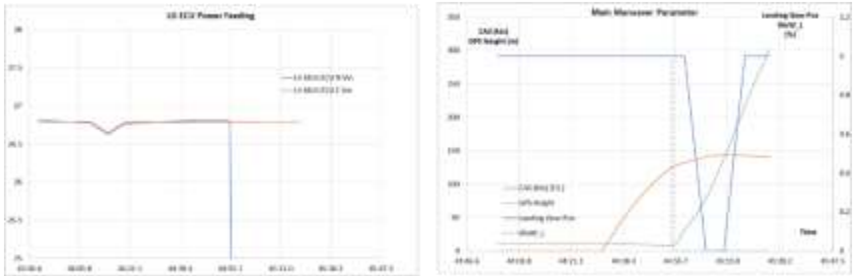

Figure 14 Results for Landing Gear Handling functionalities with single failure test

Brake System

The verification of braking functionality was performed during a rejected take-off manoeuvre, starting with a latent failure active from the beginning of the test. The manoeuvre was essentially a "Rejected Take-Off". At a speed of 114 knots, a braking action was performed using only the emergency power feeding to the brake actuator. It was observed that the braking did not cause any handling issues, allowing the aircraft to hold the runway centreline and come to a stop at a distance entirely comparable to that in the absence of a failure. The test was therefore classified as passed.

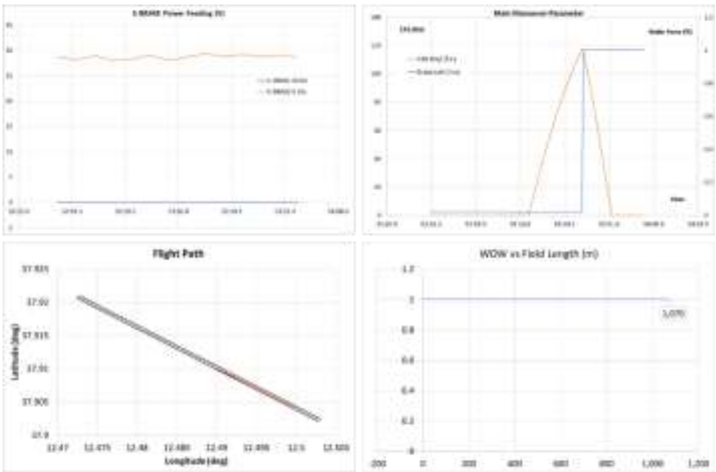

Figure 15 Results for Brake functionalities with single failure on controller

Failure with effects on multiple systems - Single Engine Failure

The effects of failure that lead to simultaneous degraded of functions on several Systems have been also assessed as in the case of single engine failure, where the behavior of the Integrated A/C System has been tested in a critical flight condition induced by asymmetry of thrust and loss of redundancy of the HV/LV electrical system during take-off.

A standard take-off was executed, the aircraft accelerated to approximately 130 knots, took off, and the left engine failure was simulated. The pilot managed the asymmetry, retracted the landing gear, and maintained a positive climb gradient. The Electrical Power Generation and Distribution System (EPGDS) reconfigured to provide full power to all systems despite the engine failure. Despite the loss of thrust, the aircraft maintained its climb rate to gain altitude. Despite thrust asymmetry, the aircraft's trajectory was largely maintained during the climb with no increase in controllability effort. All control surfaces remained powered and available due to the redundancy of the electrical distribution system. The test confirmed no power supply discontinuities to any electrical equipment on the aircraft. The aircraft's stability and handling were deemed acceptable, even with asymmetric thrust, due to the availability of all control surfaces. The test was classified as passed.

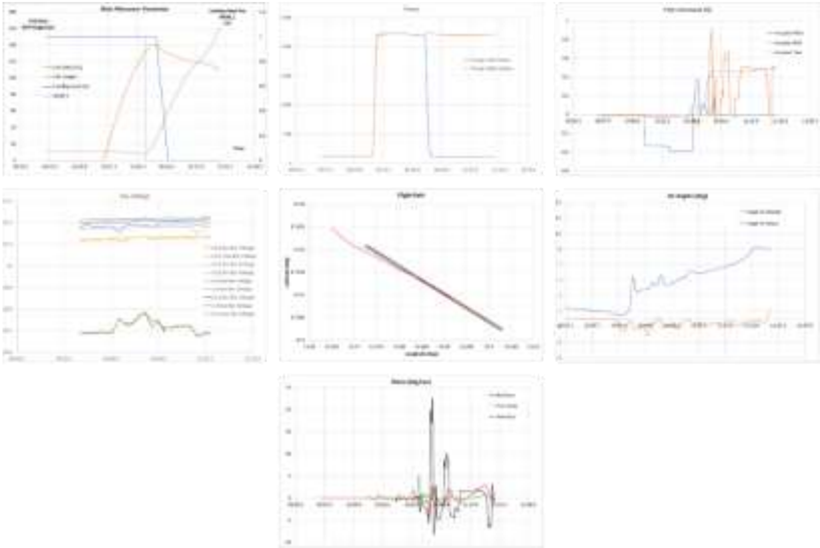

Figure 16 Results for Engine failure test

1.6 Conclusion

The increasing demand for sustainable and safer air mobility has rendered System Electrification and Digitalization as pivotal drivers for the competitiveness of future Small Aircraft. Traditional civil aircraft, characterized by a combination of mechanical, hydraulic, pneumatic, and electrical systems, have been developed over decades by system suppliers adhering to a philosophy of segregated system architecture. This design approach, however, has inherent limitations in terms of safety, efficiency, and operating costs.

The research presented in this paper explores the feasibility of overcoming these limitations through the development and integration of innovative and cost-effective Fly-by-Wire Flight Control Systems (FCS) and Electrical Technologies. Technological advancements have led to the replacement of simple mechanical and analog equipment, which require direct pilot control, with more complex, electric, electronic, and digitally controlled systems such as Flight Control Computers (FCC), Avionics Full-Duplex Switched Ethernet (AFDX) networks, Electro-Mechanical Actuators (EMA), Solid State Power Controllers (SSPC), and Adaptive Dynamic Programming Neural Networks.

While system design has become more complex to provide enhanced capabilities, software controls have been introduced to maintain, or even simplify, crew management. However, the use of software introduces an additional level of complexity. Furthermore, the interdependence of aircraft systems (e.g., electronics depending on electrical power supply) necessitates coherent design and control, as all systems jointly contribute to safe flight.

The Aircraft-0 test bench, developed for the validation of these new technologies, provides clear evidence of the improvements achieved through the adoption of these technologies. It also highlights the increased level of synergy and interdependency and offers a crucial means to test the integrated architecture. The risk of error is inherent in the development of such complex and highly integrated aircraft.

It is important to note that the current P-180 FCS is purely mechanical and thus solely dependent on pilot command on flight controls. The introduction of Digital-FCS (D-FCS) based on EMA provides unparalleled capability while making the System dependent on Electrical Generation and Distribution, Computing Platform (FCC, DCU, Switches, etc.), Inertial System, and so forth.

The specification of requirements and subsequent design, manufacturing and/or coding of such a complex architecture can introduce errors or other unforeseen side effects. Several steps must be undertaken to mitigate this risk and enhance the capability for early and effective detection of potential problems. Conducting the development in a structured and rigorous manner will ensure that the number of deficiencies or errors in a system design are minimized.

Best practices such as analysis, independent review, or comprehensive simulation can support the definition phase of a project, while laboratory testing of real items, from item level to aircraft level, provides the most effective way to verify the project after realization. This is the purpose of the Aircraft 0, a full-scale model of the P-180 used to test the integration of each of the aircraft systems — software and hardware-in-the-loop, electrics, flight controls, actuators, avionics, wiring, landing gear, and brakes — to ensure their proper functioning together.

The Aircraft 0 has been used to verify both nominal conditions and intentionally induced failure scenarios, also assessing human factors with pilots virtually handling the aircraft on-ground and in-flight. The first ground test campaign covered the verification at the aircraft level of the proper integration between D-FCS, Electrical Power Generation and Distribution Systems (EPGDS), and Electrical Landing Gear Systems during start-up/shut-down and during flight and ground maneuvers (take-off, climb, cruise, descent, land, and rejected take-off) without system failures and with possible external disturbances (induced oscillation, cross-wind, etc.).

The systems were monitored to highlight potential malfunctions or unexpected behavior leading to unexpected issues with flight control handling. Maneuvers were successfully executed, and tests passed without any major remarks, with acceptable handling grading.

The second Ground Test Campaign was executed by injecting failures into the EPGDS, Fly-by-Wire (FbW), and Landing Gear System to assess the resilience of the aircraft to potential malfunction occurrence. Failures were simulated on the electrical side for generation and distribution. The simultaneous Low Voltage/High Voltage (LV/HV) generation failure was tested, assuming the possible case of a single engine failure, to assess the behavior of the system in a critical flight condition induced by the asymmetry of thrust and where the redundancy of the electrical system was demonstrated to be adequate to support necessary flight controls.

Specific failures were injected on the distribution side to assess proper fault-tolerant logics for both LV and HV boxes. Each test was successfully passed, confirming the adequacy of the architecture of the EPGDS, as already demonstrated on SYS projects, and the proper integration with FbW and Landing Gear Systems.

Failures were also introduced on the FbW side on single actuation of the primary control surface (Rudder, Aileron, and Elevator) and on a single component of the Computing Platform (DCU, AFDX Switch) during landing simulation with external disturbance (crosswind and/or step command) to assess the stability and handling qualities in such a demanding maneuver. The possibility to land also in this critical condition was confirmed, and the tests were passed.

Testing on Landing Gear and Brakes with failure occurrence was performed at the end to complete the ground campaign. For this purpose, the effect of failures on Landing Gear Systems was reproduced as power feeding loss on one of the redundant channels, verifying that the redundancy allows for tolerating a failure without safety effect both on Landing Gear Actuation and Brake, as expected for certifiable systems. This part was positively carried out, completing the laboratory test campaign.

Overall, the tests confirm the Technology Readiness Level 5 (TRL5) and the adequacy of the architecture and integration of the systems, demonstrating the aircraft's capability to handle critical conditions and failures effectively. This research thus contributes significantly to the advancement of sustainable and safer air mobility.

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