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FIRST FLIGHT OF THE ECARAVAN

magniX and AeroTEC's All-Electric Cessna 208B Technology Demonstrator

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ABSTRACT

In May of 2020, magniX and AeroTEC successfully flew the first all-electric Cessna 208B Caravan, referred to as the eCaravan. At the time, it was the largest commercial all-electric aircraft to fly, a milestone achievement in the flight testing of electric powered aircraft. The test team was a combination of magniX subject matter experts, with extensive flight test experience, and AeroTEC Flight Test Engineering, Instrumentation, Test Pilot, Maintenance and Operational staff. The program brought together several outsourced components and required strict configuration control to ensure the success and safety of the flight test program. Potential hazards were documented and discussed, and a thorough risk mitigation plan was put into place to address the complexity of the system, integration and new technology employed. Critical component failure during crucial phases of flight were taken into consideration and mitigation strategies were not only extensively practiced but actually employed during the flight. This paper will examine the methods and processes used to modify the platform, prepare for the flights, mitigate identified risks, and execute the flight test program.

I. THE ECARAVAN PROGRAM

The eCaravan project was initiated by magniX in early 2019 for the purpose of demonstrating the viability of electric powered flight though the retrofit of an aircraft of a size used in commercial applications, and to further develop the technology though integration and flight test that would form the basis of future development and eventual type certification. The eCaravan, Fig. 1, was based on a Cessna 208B Grand Caravan, a popular 9-seat single engine PT6 turbine powered aircraft used extensively around the world in commercial applications ranging from recreation and bush flying to serving remote communities. The core technology of the electric power train was the magniX "magni500" electric propulsion system and a 258 kwh lithium battery based energy storage system.

Fig. 1 The eCaravan

The integration work was performed by AeroTEC at their Moses Lake facility. AeroTEC also performed all the aircraft related structural design, integration, test build up and execution. magniX provided the electric power train (engine, controllers, flight deck display, systems, battery and battery management).

The eCaravan first flew in May 2020, and throughout the program achieved three flights, with the longest being 30min, flying to a maximum altitude of 8,000ft and successfully demonstrating that electric flight of a retrofitted aircraft of this type is feasible.

This paper will provide an overview of the technology at the core of the program and the integration into the Caravan, the challenges encountered, and lessons learned during the lead up to and execution of the first flight as well as discuss in detail the preparation undertaken to conduct this first flight successfully and safely.

II. ECARAVAN ELECTRIFICATION ARCHITECTURE AND INTEGRATION

A. MAGNIX ELECTRIC ENGINE SYSTEM ARCHITECTURE

1. SYSTEM OVERVIEW

The magniX magni500 electric engine was a propulsion system that converts electric energy into mechanical power for aerospace propulsion applications. The system combined a single electric engine (magni500) with four independent engine controllers (magniDrives), cooling system, governor oil feed system and Human Machine

Interface (HMI) into a single system. The motor, magniDrives and support equipment together are referred to as an electric engine or Electric Propulsion Unit (EPU) and was developed as a direct replacement for mid-size traditional Pratt & Whitney PT6 turbo prop engines for electric retrofits of existing aircraft, or clean sheet designs.

The motor and magniDrives were liquid cooled, with two independent cooling systems forming a split system that remain functional in the event of the loss of a single cooling system. The architecture of the EPU (Fig. 2) is comprised of four independent segments, each controlled by a separate magniDrive, with the remaining segments remaining fully functional in the event of the loss of a single or multiple inverters. This redundancy and graceful degradation of system performance in the event of the failure of a single component without loss of the entire EPU was a key design principle employed by magniX in the development of the system, with the intent to design a system with greater levels of safety and reliability then traditional propulsion systems for single engine applications.

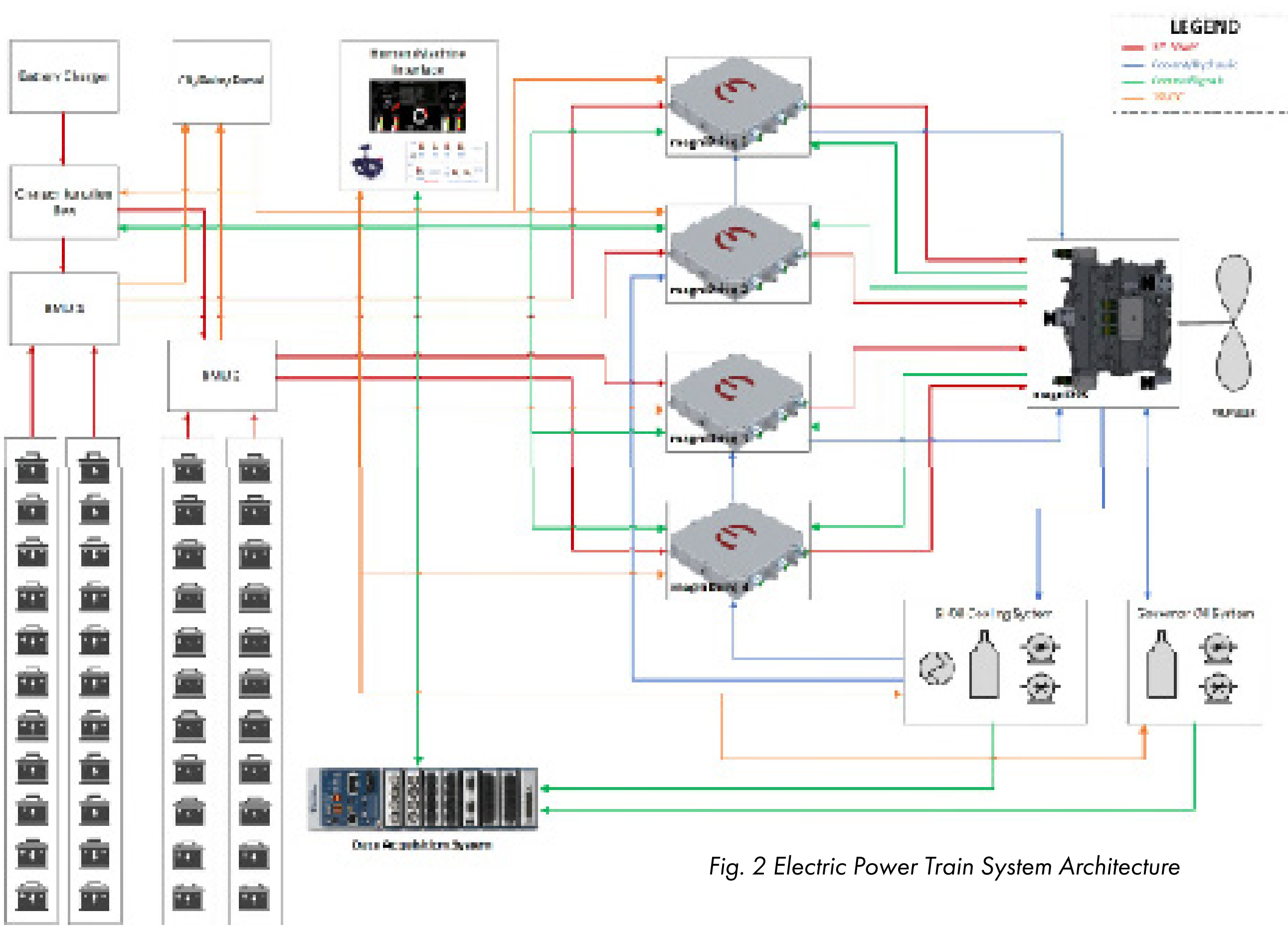


Fig. 2 Electric Power Train System Architecture

2. ELECTRIC ENGINE

The magni500 was a liquid cooled direct drive permanent magnet synchronous machine with four independent three phase groups, with each group controlled by a separate three phase variable frequency inverter and controller (magniDrive).

The engine is capable of continuously delivering up to 560kw (750 shp) of shaft power at 1900RPM to drive a conventional constant speed propeller. Speed control was achieved via a traditional hydromechanical propeller governor mounted to an accessory gearbox on the rear, with high pressure oil fed through the shaft to the propeller hub.

Control of the engine was achieved via a torque command received from a position sensor on the aircrafts power lever read by the magniDrives which regulate the current delivered to the motor, thereby regulating the torque on the shaft.

3. ENGINE CONTROLLERS

The magni500 based EPU was controlled by four independent controllers (magniDrives). Each magniDrive is a single 3 phase variable frequency drive, converting DC power to AC current regulated by the torque command from the aircrafts power lever.

In addition to the core function of torque control, each magniDrive also acted as a health and performance monitor, with an independent condition motoring unit, which ensured the accurate control of the motor, provided error detection and communication to the HMI display, and could act to shut down its segment in the event of a detected failure. This dual command and monitoring functionality was a core element of system safety built into the magniX EPU.

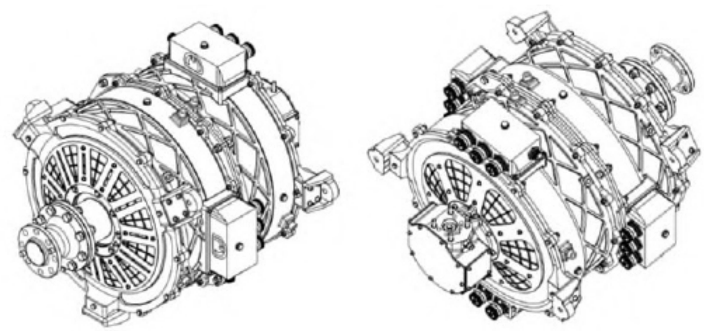


Fig. 3 magni500 Electric Engine

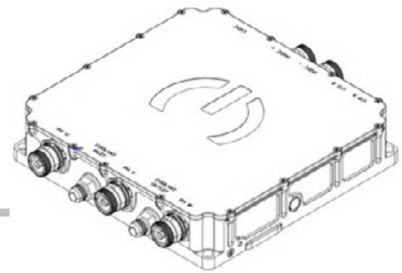
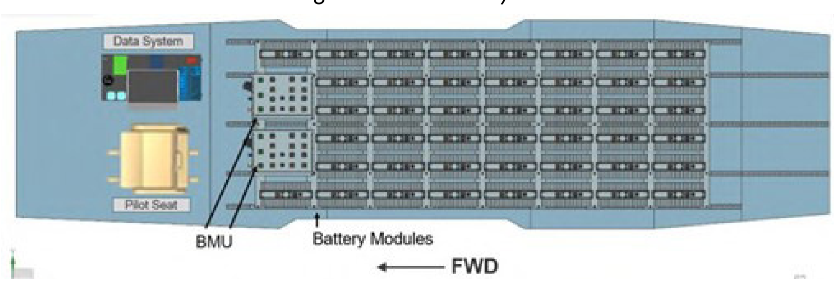


Fig. 4 magniX magniDrive

4. ENERGY STORAGE SYSTEM

The Energy Storage System (ESS) replaced the conventional fuel system and supporting subsystems on the aircraft with electrical energy storage, in this case batteries. The ESS in the eCaravan consisted of two independent battery systems, each with a dedicated Battery Management Units (BMU) and two strings of 11 battery modules for a total of four battery strings in the aircraft. The batteries and BMUs were located along the inside of the fuselage in what is normally the passenger cabin. Due to the experimental nature of the program, the battery system was designed for safety and practical integration rather than optimized for weight or volume, and as such filled most of the available cabin space and useful payload, shown in Fig. 5. The selection of the battery cell used in the project was also focused on the maturity and experience from the supplier with that specific cell model.

Fig. 5 ESS Cabin Layout



The ESS was selected and contracted in order to deliver sufficient capacity to support demonstration flights with a design that would guard against cascading thermal runaway events in flight and be able to operate under all expected flight campaign operational environment without the necessity of liquid thermal management system. The result was a robust lithium-based battery system designed to meet the specification shown in Table 1.

The BMU included all the functions typically related to the battery systems such as managing the charging, balancing and monitoring of the battery modules. It also included the high voltage distribution system with necessary hardware to connect the battery strings to the magniDrives. Each BMU also provided 28V LVDC to power the aircraft systems and magniDrives via DC-DC converters, a feature included due to the loss of the original alternators driven by the engines in the conventional architecture. The independent nature of the architecture provided for redundancy in the event of a single BMU failure or shutdown.

Table 1 Energy Storage System Design Details and Specifications

PARAMETER	NAME/VALUE
Cell Manufacturer	Samsung SDI
Cell Model	INR 18650-30Q
Module Configuration	16 Series 34 Parallel
Number of Modules per string	11 Series
Number of strings	4
Maximum System Voltage	739.2 V
Nominal Voltage	633.3 V
Minimum Voltage	440.0 V
Nominal Capacity	258.5 kWh
ESS Weight	3850 lbs

B. AIRCRAFT INTEGRATION

1. MECHANICAL INSTILLATION

Design modifications to the Caravan motor mount were developed to interface directly into the existing Caravan engine mount support truss for the PT6 engine being replaced. In addition to the electric engine, the new mount also provided mounting for the drives, cooling system, governor oil system, sensors and pumps, as shown in Fig. 6. No modifications to the Caravan firewall truss were needed. A stress analysis was performed on the new mounting bracket system and all components, including the original Caravan firewall truss, showing positive margins for all loading cases. The engine mount analysis followed a 14 CFR part 23 approach noting that electric engines have some unique loading conditions that are not issues on traditional gas turbines owing to the ability to spin up very quickly. The magniDrives limit start up torque in an effort to prevent high stress levels in the motor mount and blades during start up.

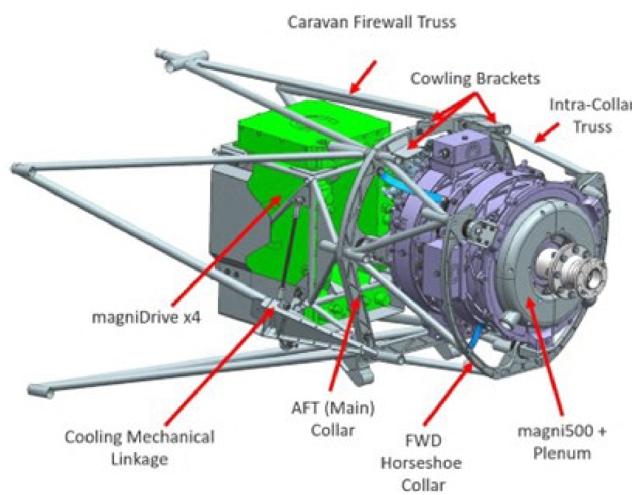


Fig. 6 magniX EPU Integration

2. EPU COOLING

The EPU cooling system consisted of two independent silicon oil-based systems each with its own pump, filter and heat exchanger all installed forward of the firewall as shown in Fig. 7. Each system provided cooling for one half of the EPU and two magniDrives. The cooling pumps were powered from the aircraft 28V LVDC system and were controlled with two toggle switches (labeled Pump 1 and Pump 2 with Auto, On, and Off positions) located on the EPU/ESS test panel in the flight deck.

The aircraft side of the cooling system was designed to provide for sufficient cooling of the EPU at takeoff power with an outside temperature of 35ºC. The system consisted of two heat exchanges, one for each independent cooling oil loop, and required ducting integrated into the exhaust ports for the traditional PT6 (Fig. 9). A new nose bowl was manufactured to provide for a cooling air scoop in the nose, shown in Fig. 8.

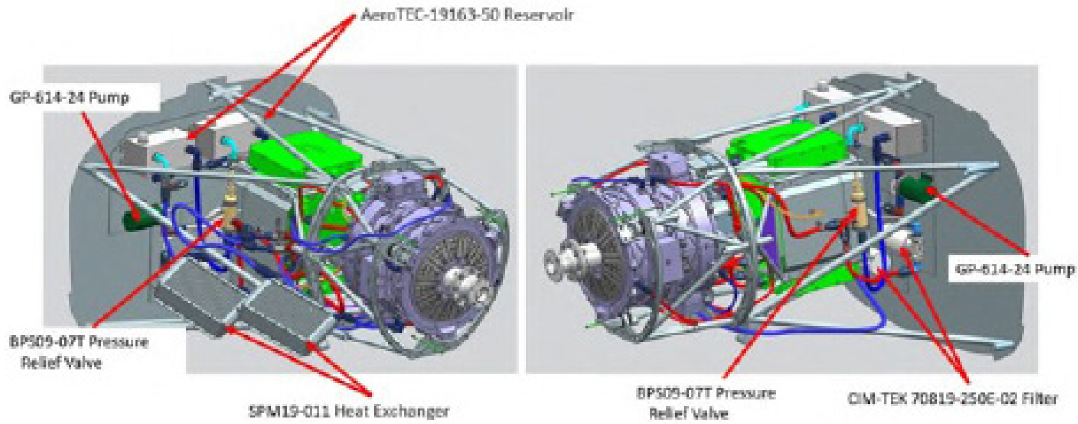


Fig. 7 Cooling System Installation Layout

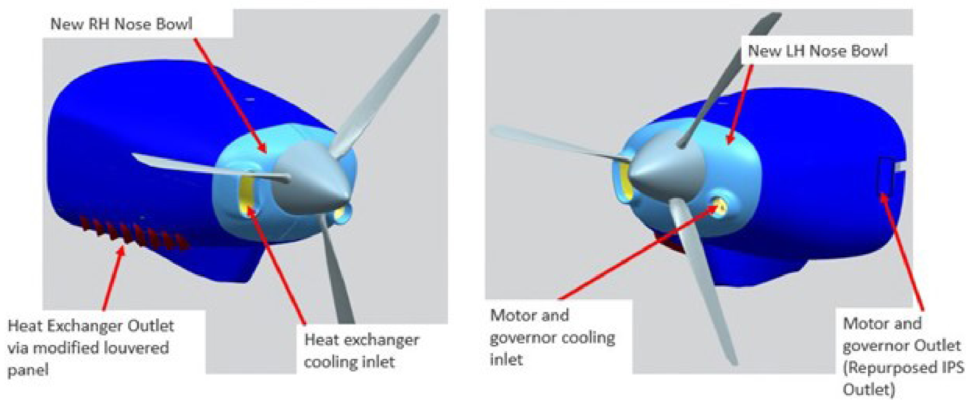


Fig. 8 Custom Nose Bowl and Mounts

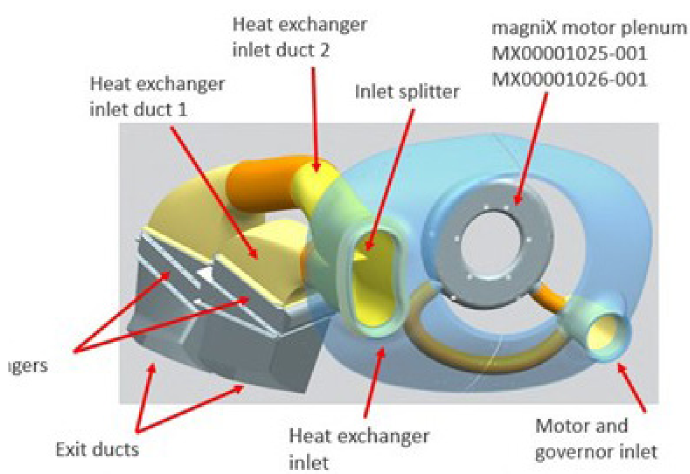


Fig. 9 Custom Ducting

3. BATTERY INTEGRATION

The design of the battery system was intended to provide the highest level of safety level through its internal module design, independent of the integration into the aircraft. The majority of the integration efforts were focused on the mechanical fixtures required to ensure sufficient structural capacity due to the large mass. Optimization of the module configuration was performed to provide enough spacing between modules to aid with passive cooling. Ducting was installed to provide an escape path for gases generated if a thermal runaway event happened. Thermal and system monitoring were some of the principal drivers in the design with each battery cell housed in individual casings that provided a passive means to contain thermal heating and to isolate individual cells from cascading thermal runaway events. Any off-gassing through ports on the top of each cell could be collected and ported overboard through a fan and ducting system designed specifically to handle such occurrences. A false floor was placed over the batteries and ducting to provide an additional egress path for the test pilot and safety for personnel as shown in Fig. 10:

Fig. 10 Battery Cluster Assembly, Ducting, & False Floor

4. EPU CONTROLS

The EPU and ESS were controlled through a custom designed control panel separate to the standard aircraft equipment. It was designed with clearly labeled switch and Circuit Breaker (CB) panels. The rack also housed the data acquisition and telemetry equipment and together was referred to as the Data Acquisition System (DAS). The rack was mounted on a plate attached to the co-pilot's seat rails (Fig. 11). Its placement ensured all switches and CB's were within the pilot's Field of Regard and could be accessed and manipulated from a static seated position using a normal reach.

DAS Rack & Mounting Plate on Co-pilot's Seat Rails

DAS Rack Switches

Fig. 11 DAS Rack & Switches

The following is a list of DAS switch functionality shown in Fig. 11 listed from left to right, top to bottom:

CONDITION (ON/OFF)	Used to start recording of telemetry (TM) data
EVENT	Used to place a mark in the TM data stream for post test data analysis
MASTER ON (ON/OFF)	Power switch to the DAS system. Required to be ON to power up all the systems
BMU 1 28 V (ON/OFF)	Power switch for the #1 BMU that managed half of the ESS batteries
BMU 2 28 V (ON/OFF)	Power switch for the #2 BMU that managed the other half of the ESS batteries
HVDC 1 (ON/OFF)	Power switch for the #1 High Voltage system that provided high voltage DC power to half of the engine inverters
HVDC 2 (ON/OFF)	Power switch for the #2 High Voltage system that provided high voltage DC power to the other half of the engine inverters
MOTOR ENABLE (ON/OFF)	Engine Run/Cutoff switch used to start and shut down the engine
TM XMIT (ON/OFF)	Telemetry power switch
INVERTERS 1/2/3/4 (ON/OFF)	The power switches for the four respective magniDrive(s)
EPU COOLING PUMP 1 & 2 (ON/AUTO/OFF)	The power switches for the two engine and magniDrive cooling circuits
GOVERNOR OIL PUMP (ON/AUTO/OFF)	The power switch for the governor oil pump. The system provided pressurized oil to the governor for propeller pitch control

5. MAGNIX CUSTOM INFORMATION DISPLAY

Engine monitoring was provided through a single magniX provided custom display (Fig. 12) mounted on the main instrument panel adjacent to the Comm/Nav radio head stack. It was powered through the aircraft's 28VDC main engine power bus. Information from each of the four magniDrives and the ESS was routed to the display on the EPU CAN bus. In line with convention, any loss of communications between inverters and components would result in a red X appearing over the associated information.

The primary Engine Instrument Torque and Speed gauges, located in the upper left and right sections of the display, mimicked the analogue legacy gauges of the PT-6 engine that the magni500 had replaced. The Torque dial gauge displayed commanded and engine output (feedback) torque calculated by the inverter. The Speed gauge displayed engine RPM. Like the legacy displays, amber sections were "keep out"/transient zones, i.e. no dwelling, and the red sections were "keep out"/limit zones. See Fig. 13.

Fig. 12 Overview of the Flight Deck Engine Display

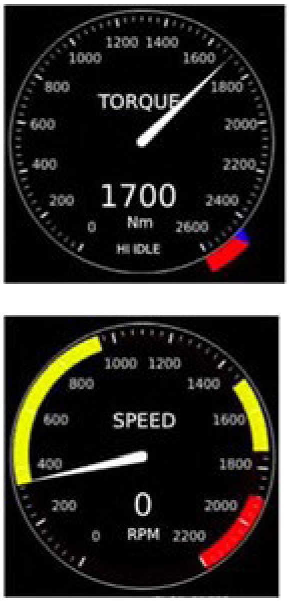


Fig. 13 Primary Instruments

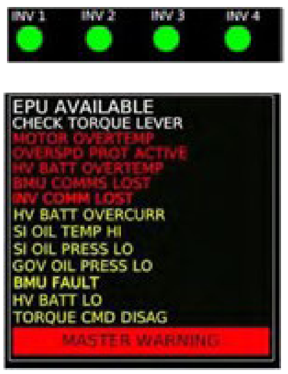


Fig. 14 Crew Alerting System

The Crew Alerting System, shown in Fig. 14, consisted of a conventional CAS messaging area and magniDrive State Indicator lights. These state indicator lights were used to alert the crew in the even any of a number of internal fault conditions were present in the inverter. If no fault state existed, the light would remain green.

The BMU status was communicated via status lights on the display. They were red if the input contractors for either string of the two associated battery strings were OPEN. The status lights were green when both contactors for the associated battery strings were CLOSED. A yellow status light indicated both contactors were CLOSED; however, Battery Control Computer in that BMU encountered a fault.

The State of Charge (SOC) displayed the current total ESS state of charge. The charge state was displayed by both a tape and a digital readout in a percent of current charge.

The CHARGE REMAINING was a digital readout of the amount of energy remaining in the batteries, expressed in kWh. The value was calculated based on SOC information provided by the ESS BMU's SOC of their connected strings.

FLIGHT TIME REMAINING, expressed in minutes, was a digital readout of the number of minutes of flight time remaining based on real time current power settings of the engine and the SOC.

The final section on the display was the EPU and ESS internal temperatures and pressures and real time power indicator. See Fig. 16.

The power indicator showed the instantaneous mechanical shaft power output of the engine as calculated by the magniDrives. The indicator showed a proportional white to black, clockwise rotating indication of current engine output power. At full power the white ring would fill in the entire outer circumference of the circle. A digital readout of the current mechanical power output in kW was displayed in the center of the circle.

Inverter, battery, motor, silicone oil (main EPU cooling loop), and governor temperatures and pressures were displayed on horizontal tapes. The highest and lowest values of each system were presented via carets on the tops and bottoms of the horizontal tapes. The upper carets showed the highest component value, and the lower carets presented the lowest component value.

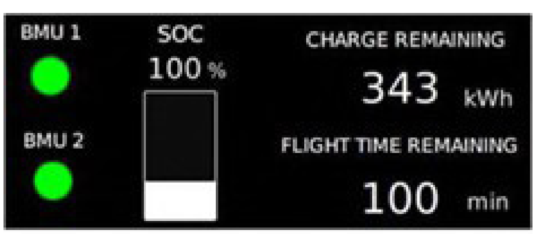


Fig. 15 ESS Status Region

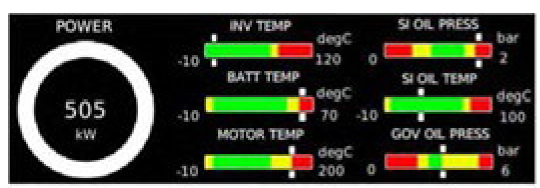


Fig. 16 Engine System Status Bars Section

6. EFFECTS ON W&B

Changes made to the aircraft through the addition of the magniX EPU and ESS system had a significant effect on the aircrafts Basic Empty Weight (BEW), overall Gross Weight (GW), and aircraft Center of Gravity (CG). Following modifications, the aircraft BEW increased to 9,050 lbs. This resulted in a GW of 9,360 lbs on first flight. A modified W&B envelope was developed based on structural, performance, and S&C analysis. An operational CG range was established to ensure the modified forward CG limit reflected the elevator authority range, while ensuring the aft CG limit could be maintained following a catastrophic failure of the crankshaft that resulted in a propeller separation. To accomplish this, 70 lbs of ballast was added near the pilot station to bring the CG to 37.7% Mean Aerodynamic Chord (MAC). See Fig. 17:

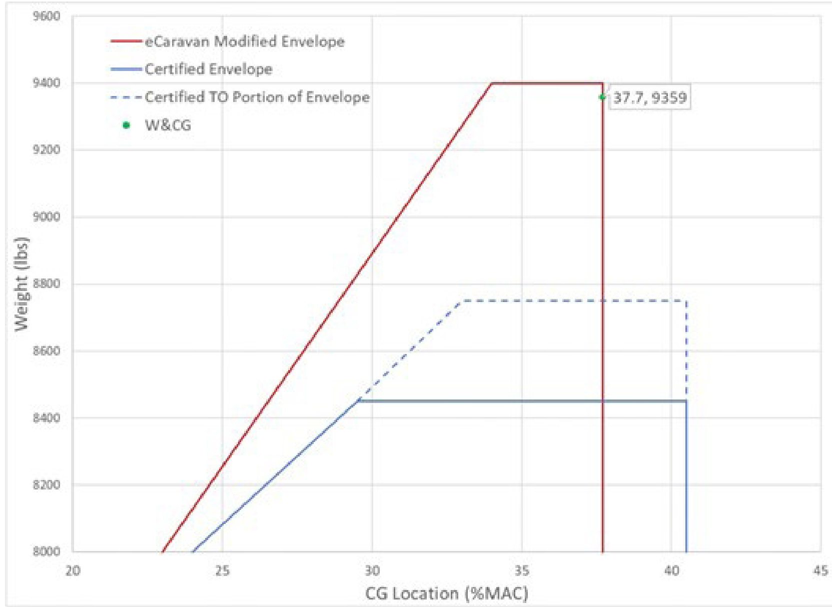


Fig. 17 eCaravan Modified W&B/CG Envelope compared to the Unmodified Caravan

7. OPERATIONAL EFFECTS (MTOW/MLW, LOAD FACTOR, CLIMB/GLIDE PERFORMANCE LIMITATIONS)

The increase in GW had several effects on aircraft performance. The Maximum Takeoff Weight (MTOW) was now 610 lbs above the aircraft's certified maximum GW of 8,750 lbs in this configuration. This, coupled with a Shaft Horsepower (SHP) of 520 kw (610 SHP), effected takeoff, climb, and glide performance. Since consumed energy in an ESS did not affect GW, the MLW was also above the certified GW limit of the aircraft. This meant a Load Analysis and subsequent operating limitations were required for flight tests.

Performance analysis was performed based on an assumed GW of 9,360 lbs, an Outside Air Temperature of 20ºC, and an altitude band between 1,189 and 3,189 ft MSL. A summary of analysis results is presented in Table 2.

Table 2 Takeoff, Climb and Landing Performance Analysis

PARAMETER	NAME/VALUE
Takeoff Distance	2,288 ft
Takeoff Distance to Clear a 50 ft Obstacle	4,277 ft
Rate of Climb (ROC)	541 ft/min
Maximum Climb Gradient	377 ft/NM
Glide Ratio at 2,000 ft MSL	15.0
Landing Distance	1,109 ft
Landing Distance to Clear a 50 ft Obstacle	2,022 ft

Despite operating above the certified GW of the aircraft, no modifications to the aircraft structure were made. A comparative approach to restrict the aircraft maneuver and landing load factors kept the total load into the aircraft from exceeding the certified design loading. Should an exceedance occur, an airframe and component overstress inspection would have been triggered. Both the flight limit maneuver load factor and the landing load factor were analyzed. Of the two, the landing load factor was determined to be the most restrictive and therefore used as the modified configuration load limit for test.

The landing load factor was estimated using Eq. (1) (Ref. [1] and [2]): (1)

$$n_{LG} = \frac{\frac{V_S^2}{2g} - \frac{\eta_t K_t X_t^2}{2W} + \left(1 - \frac{L}{W}\right)(X_t + X_S)}{\eta_s X_S},$$

η_s	Steel Spring Gear	L	Wing Lift (CFR 23.473(e))	g	32 ft/sec^2
V_s	Sink Rate (CFR Part 23)	X_s	Strut Stroke	S_W	Wing Area
K_t	Tire Spring Constant	η_t	Tire non-linear factor	N_Z	Load Factor
X_t	Tire Deflection	W	Max Landing Weight	n_{lg}	Landing Gear Load Factor

The test load factor limit was reduced from the certified limit of 3.6 G down to 2.5 G. To ensure operation within this limit, a Test Operational Limit (TOL) maximum sink rate of 6 ft/sec during landings was established.

C. ENERGY STORAGE AND MANAGEMENT

1. SOC VS. FUEL DISPLAY SYSTEMS – PHILOSOPHY

The design of information on the ESS energy use and charge remaining on the display was intended to provide information to the pilot in a way that the pilot would understand from experience flying aircraft with conventional power systems. In some battery related products, it is common to have a State of Charge percentage indication, which doesn't give an absolute indication of the energy available, similar to how a % fuel remaining does not inform the amount of fuel in the tank without knowing the size of the tank. Based on the ESS developed for this project the absolute SoC information was presented as charge remaining in kWh, or total energy remaining. This would help the pilot decide on endurance available based on the current power setting.

That parameter associated with instantaneous power consumption and a direct calculation of time remaining would give a set of indications to assist the pilot to make decisions on routes or alternative landing sites in case of necessity.

The discussions around the available energy and state of charge indications show the relevance of bringing the electrification to the operational environment to fully realize the requirements and necessities. With the indications set and agreed the operational limits, such as how much energy would be required for go around or to fly to an alternative landing site, could be defined.

Fig. 18 ESS Status Display

D. ESS CHARGING

A new charging infrastructure capability was set up to allow charging the airplane in the operational environment planned for the project. A few points were realized in the beginning of the discussions around this point:

- For the 260kWh expected size of the battery an ultra-fast charger would be required to give reasonable charging times
- Ultra-fast charges typically require specific power cabinets with dedicate high voltage input infrastructure
- The traditional supplier for EV chargers were not open to customization of their chargers for the project so an off the shelf solution had to be used
- It was desirable to have a long charger cable connection to keep the charger at a reasonable distance from the aircraft
- The battery selected for the program relied on a specific power connection solution and CAN bus communication not typically found on commercial chargers

Based on the above a solution of acquiring a commercial charger and building a "junction box" that would allow the team to manage the interface between it and the battery system was the selected solution. The junction box would also include an extension cable.

The redundant architecture of the ESS also allowed the team to have each pack charge independently. In that case, having 2 independent 75kW DC fast charger systems would allow the team to re-charge the aircraft at a reasonable rate. The figure below shows the schematic for each of the charging systems.

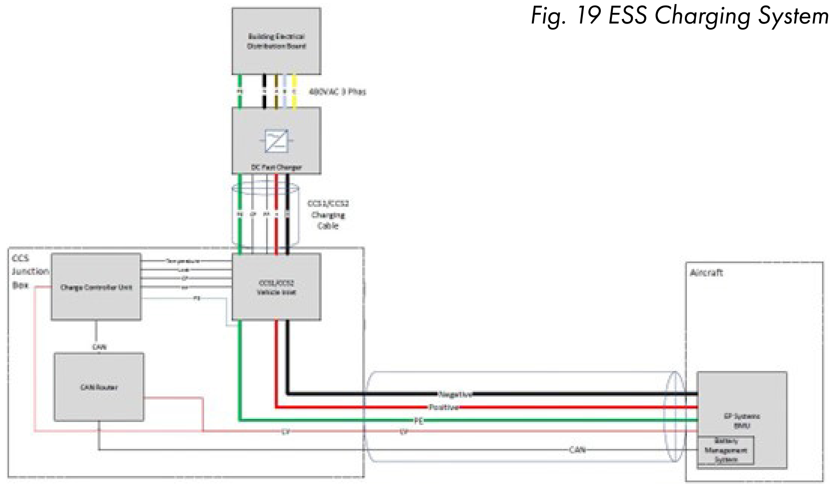


Fig. 19 ESS Charging System

III. PREPARATION FOR FIRST FLIGHT – TEST BUILD UP

A. THREAT ASSESSMENT

1. SINGLE ENGINE AIRCRAFT W/ A PROTOTYPE ENGINE

Integration of an experimental engine with an experimental power source on a single engine aircraft introduced several single point failures that could have led to unpowered flight. The flight path and associated ground track of the first flight needed to be assessed for hazards that could impede a glide to a prepared surface. At the time, Grant County International Airport was undergoing a major runway improvement project on the main runway, runway 32R. Large sections of asphalt and concrete from the runway had been removed and placed in large mounds at the center of the field, which varied in height between 100-200 ft. Construction equipment such as trucks, cranes, conveyor belts, water tanks, and numerous other support equipment and vehicles were scattered across many open sections on the field. Additionally, open tarmac space that year was not available due to a large volume of stored B737 MAX aircraft. This significantly reduced potential off runway emergency landing site options.

2. REDUCED PERFORMANCE

The test configuration GW & CG were greater than the base aircraft's certified envelope and the planned operating SHP of the magni500 engine was 68 SHP less than the PT-6 engine that it had replaced, which affected climb performance. As shown in Table 3 and Table 4.

Table 3 Base Caravan Performance

RATE OF CLIMB						
				0º C	*20º C*	*40º C*
Flap	Weight LB	Pressure Altitude Feet	Climb Speed KIAS	**Rate of Climb**		
				Ft/Min	*Ft/Min*	*Ft/Min*
0	8750	2000	90	850	830	770
0	8750	4000	90	825	805	685
20	8750	2000	104	967	905	625
20	8750	4000	104	945	840	505

Table 4 eCaravan Performance

RATE OF CLIMB						
				0º C	20º C	40º C
Flap	Weight LB	Pressure Altitude Feet	Climb Speed KIAS	Rate of Climb		
				Ft/Min	Ft/Min	Ft/Min
0	9360	2000	106	650	585	311
0	9360	4000	106	625	521	176
20	9360	2000	92	554	530	470
20	9360	4000	92	529	508	589

3. HANDLING QUALITIES AND STRUCTURAL INTEGRITY UNKNOWNS/ASSUMPTIONS

Analysis provided the team with a high level of confidence that the aircraft's handling qualities would be good and that the aircraft structural integrity throughout the flight test envelope were known; however, this was based on two assumptions.

1. The aircraft would exhibit similar stability and control characteristics in the test configuration as the normal certified configuration, based on no changes to the outer mold line and use of the OEM propeller and governor.
2. Reducing the load limit of the aircraft would provide adequate structural margins

Actual handling qualities and structural integrity status would require flight testing and post-flight data analysis.

4. BATTERY THERMAL RUNAWAY HAZARD

The potential threat of a battery thermal runaway presented a couple of unique in-flight hazards. These included:

- Crew injury from a battery explosion
- Loss of visibility in the cockpit from off-gassing
- Inhalation of battery smoke and fumes from off-gassing
- Chemical burns
- Structural damage around the main gear box and wing
- Loss of pitch control due to burn through of elevator control cables

5. MAGNIDRIVE & BMU STATE INDICATION INTERPRETATION DEFICIENCY

It was discovered during the integration process, that initial assumptions on the interpretation of information on the display had been made such that all indications were not clearly understood, i.e. information interpreted one way by engineers is interpreted differently by pilots and operators given their unique and specified roles, training and prejudice of experience. This resulted in what the operations team perceived as confusing, incorrect or "stuck" indications where the magniDrive and BMU state indicators would "stay green" regardless of DAS Inverter and BMU switch positions. To the engineer this simply means no fault conditions are present, however to the operations team this indication means "ready to operate", which with a switch in the off position is not the case. A programmatic decision was made to proceed into flight testing with this known deficiency.

B. RISK MITIGATION

1. PROFILE PLANNING

A flight path and vertical profile was chosen to ensure that throughout all phases of flight, the aircraft could glide to a safe landing on a prepared surface following an unplanned loss of thrust. Analysis showed that the glide ratio remained unchanged from the baseline aircraft. Several surveys of the airport surfaces and obstructions were

Fig. 20 Initial Climb Out Profile

conducted. Fields and structures adjacent to the airport grounds were also assessed as potential off airport emergency ditching sites. All surveys and data were used to settle on the first flight profile. The takeoff would be made from runway 14. Climb data suggested a turn could be made upon reaching the crossing of runway 32. Climb data showed the aircraft should be at 130 ft AGL at this point. Should an engine failure occur during the turn a landing could be made straight ahead on the remaining surface of runway 14 or on the roughly prepared surface of runway 32 following completion of the turn. The next critical point was 400 ft AGL. At this point a turn was planned to parallel runway 18. By the end of runway 18, the aircraft would be at 900 ft AGL. Throughout all phases of the initial climb, performance data showed a landing could be made on one of the prepared surfaces. See Fig. 20.

Sufficient altitude was planned in the cruise and landing phases of flight to ensure a power off glide to runway 14 could be made at any time.

2. PITCH AUTHORITY IN THE LANDING FLARE

Two low speed control evaluation configurations were planned to identify any potential pitch authority deficiencies in the takeoff flap (F10) and landing flap configurations due to the GW and CG configuration. For both conditions, the aircraft was configured in level flight, 5 kts above the calculated approach speed. Thrust and trim were then adjusted to establish the aircraft in a steady 700 ft/min descent. Once in a steady state condition, a slow 1 kt/sec deceleration was conducted until either the stall warning horn sounded, or full aft yoke was achieved. If at any time during the test condition the aircraft altitude reached 3,200 ft MSL (2,000 ft AGL) or an unexpected pitch down or loss of control authority were to occur, a Knock-it-Off callout would have been made and the test condition would be terminated. If any aircraft buffet, nose drop-off, or wing rock were encountered prior to activation of the stall warning horn activation, or full aft yoke travel achieved, approach speeds would be increased 5 knots above the anomaly occurrence.

3. ENGINE-OUT GLIDE AND LANDING PERFORMANCE TRAINING

April 27, 2020, the test team completed a training flight on a rental Caravan in preparation for the eCaravan first flight. The objective was to understand the handling characteristics of a Caravan configured similarly to the eCaravan in the landing pattern. The two primary conditions we focused on were short field takeoffs/maximum angle of climbs and power off/simulated feathered propeller landing. In order to simulate the MGTOW planned for the eCaravan, the aircraft was ballasted with two boxes filled with 1,600 lbs of sand bags, which brought the MGTOW up to 9,000 lbs (Fig. 21 shows the rental aircraft and ballast system). To simulate the reduced thrust performance of 605 shaft horsepower, the team planned to manually limit takeoff/go around thrust to 1,800 lbs of torque. The flaps 20 short field takeoff procedures were used for all takeoffs up to 420 ft AGL, followed by accelerations to 95 KIAS flaps 10 and 105 KIAS flaps up.

The aircraft climbing up to 6,000 ft MSL then slowed to a best glide speed of 95 KIAS. The pilot pulled the throttle back to idle and feathered the propeller. The aircraft was flown in low idle for the entire flight. The aircraft settled into an 800 ft/min descent and an altitude loss of approximately

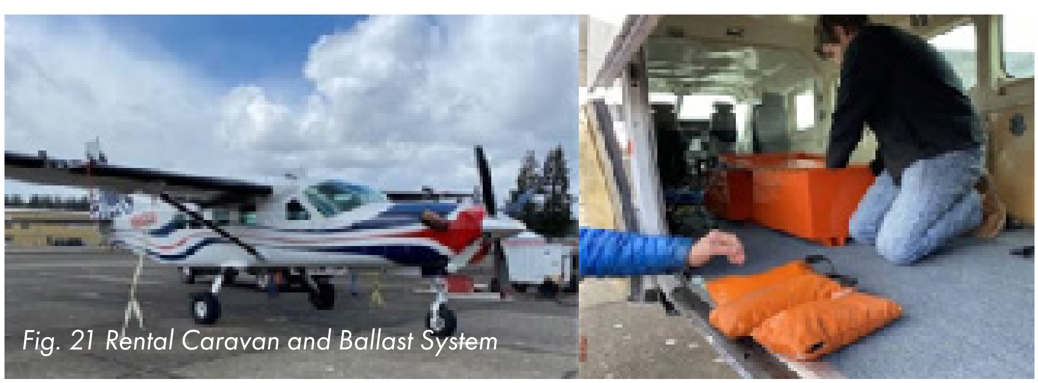


Fig. 21 Rental Caravan and Ballast System

400 ft was encountered during the transition to the feathered condition. This altitude loss was taken into account when determining emergency landing patterns for the eCaravan flight tests. The rate of descent increased slightly by 50-100 ft/min in 15 degree Angle of Bank (AOB) turns and 100-150 ft/min for 30 degree AOB turns. Next the pilot commanded a climb back up to 6,000 ft MSL and attempted to recreate the performance of a feathered propeller using power. It was determined that the performance could be duplicated using approximately 300 ft/lbs of torque and the propeller RPM set to minimum. This performance data was used to conclude that a minimum pattern altitude of 1000 ft AGL would be required to safely execute simulated power off/feathered propeller approach to landings from the abeam position in the eCaravan.

For the shortfield takeoffs/maximum angle of climbs, it was found the manufacturer's suggested technique of setting the rotation speed to 70 knots worked best to intercept an 85 KIAS climb out speed. Once achieving a minimum altitude of 420 ft AGL the aircraft was leveled off, flaps were retracted to 10 degrees, and the aircraft was accelerated to 95 KIAS. This resulted in a ROC of approximately 500 ft/min. The ROC could be increased to 800-900 ft/min by raising the flaps to the up position and continuing the acceleration to 105 KIAS. Fluctuations in vertical speed were attributed to light to moderate turbulence, intermittent rainstorms, and high winds encountered throughout the flight.

Simulated power out/feathered landings were executed from both left hand and right-hand patterns. Consistent patterns and landings within the first 1,000 ft of the touchdown zone were achieved by flying a continuous 15-25 degree AOB turn, flaps 10, and slow deceleration from 95 KIAS to 90 KIAS until short final. Flap selections of 20 to 30 degrees were delayed until achieving the runway was assured. A full cross control sideslip was consistently applied to achieve precise touchdown points (Fig. 22).

Fig. 22 Simulated Power Out/Feathered Landing in the Practice Caravan

After a 2 hour flight and 18 successful landings, the Test Pilot was able to consistently place the aircraft in the touchdown zone and safely bring the aircraft to a stop with no power and a simulated feathered propeller.

4. SOC

The test team determined that a minimum of 30% SOC energy reserve was required for landings, in order to accommodate the possibility of a go-around. This requirement dictated the flight profiles and planning that went into each flight.

5. EMERGENCY & EVACUATION PREPARATION

The test pilot wore appropriate flight gear, and the aircraft was equipped with safety equipment designed to provide the test pilot with a means to combat the hazards posed from a battery thermal runaway event. The pilot wore a Nomex flight suit, Nomex gloves, a helmet, a parachute, and flight boots with an OSHA EH rating. An oxygen bottle, mask, and smoke goggles were mounted within a dynamic functional reach of the pilot on the floor next to the DAS station. The captain's main left door was modified with a quick release system that allowed the door to fall away in the slip stream when activated, providing the test pilot with an unobstructed bailout path in flight. The door quick release system was functionally tested by the test crew in full flight gear, simulating an in-flight bailout scenario.
Test enabling hardware was designed to support alternate means of egress. The DAS rack was sized to allow the pilot to step over it during an egress through the right main co-pilot door. The ESS system was fitted with a secondary floor over the top of it to ensure the test pilot's egress path to the aft entry/exit doors was not impeded. During the ground test phase, emergency egress in full flight gear was practiced by the test pilot and flight test engineers. All egress attempts were timed.

Multiple means of egress were allowed for in the design to mitigate worst case scenarios to the greatest extent. While the pilot door was always the primary means of escape, planning for and practicing all possible scenarios helps to reduce risk in the event of unexpected situations. Electric aircraft have unique risk profiles as a battery fire remains a significant hazard which have the potential to develop very rapidly. Design mitigation remains the most effective means of risk mitigation, but this cannot be solely relied upon when the hardware or system is unproven.

6. TEST LOCATION AND FACILITIES

Grant County International Airport is a public airport that supports experimental flight test from multiple organizations, military operations, commercial operations, and a flight school. The airport is well versed in experimental flight test and was extremely accommodating to the eCaravan program. AeroTEC worked directly with the Aircraft Rescue and Firefighting team to provide training specific to the eCaravan. Training included an overview of the high voltage system, PPE, and where to cut the aircraft in the event of an emergency. On the day of first flight, the local flight school, Big Bend Community College, delayed the start of their day so that the test team would have a traffic-free pattern to work in. This kind of community and communication was instrumental to the safety and successful execution of the first flight. Additionally, prior to the first flight, AeroTEC executed an emergency drill to ensure the internal emergency process thoroughly covered all the necessary steps. Several lessons were learned during this drill and the process was updated accordingly.

IV. TEST EXECUTION

A. FIRST FLIGHT

1. SUCCESSFUL PROFILE

On May 28, 2020, the team took off on the first flight of the magniX eCaravan. Takeoff was nominal with the initial climb to 2500 ft completed in 5 minutes. The safety chase aircraft executed an airborne pickup, joining and remaining on the outside of the turn during liftoff and climb out. Ground roll and climb rates were in line with estimates and all altitude and turn checkpoints were met. No anomalies were noted.

Upon leveling off at 1,500 ft AGL the team set up for the slow speed and control authority assessment. A power setting of 1460 nm at 1900 RPM was required to sustain a 125 KIAS level condition. While at flaps 10, the power was reduced to establish a 1 kt/sec deceleration. The test point concluded with activation of the stall warning horn. No adverse pitch up, control lightening, or wing rock were noted. The maneuver was repeated with the flaps set to full. The stall warning horn was activated at 64 KIAS with similar handling qualities reported. With the assessment complete, the flaps were raised for the next phase of the flight.

The test aircraft and safety chase set up for the media portion of the flight. From a loose right echelon position, the chase aircraft crossed under the test aircraft and moved into an acute left echelon position for the lead change. The eCaravan moved into various left and right echelon positions to facilitate various still photo shots and video. Periodic SOC callouts were made throughout the event. Energy consumption tracked with flight time as expected. At an SOC of 36% the lead was passed back to the eCaravan and the flight prepared for landing.

The eCaravan entered the landing pattern and requested clearance to land on runway 14 with a SOC of 34%. From a high abeam position the flaps were set to 10 and a 700 ft/min descent was established. After a short extension on downwind, a constant turn to final was established and flaps full was selected. Shortly after moving the flaps handle, the pilot noticed the radios went dead. A quick investigation revealed that a complete power loss had occurred. The approach turn was continued while the pilot attempted to quickly trouble shoot and re-establish engine power. The aircraft Master Switch, not required for either aircraft systems or engine power in the test configuration, was found to be in the OFF position. It was placed to the ON position which resulted re-establishing radio 1 power. During this time ESS power returned (note, this was unrelated to the aircraft Master Switch being placed to ON). The magniX display came back and the BMU and engine systems began a power up sequence. With no more actions to take, attention was shifted to continuing the approach and landing. During rollout, one of the four inverters came back online and power to the engine was restored. The aircraft exited the runway and taxied back to the ramp under power. No additional anomalies were noted.

V. LESSONS LEARNED

A. CONFIGURATION CONTROL

Post flight data analysis coupled with discussions with the design team revealed internal component limits that were not clearly understood when coupled with load and resulting temperature profiles. The Power Supply Units (PSU) internal to the BMUs had a degraded output profile with elevated temperature which was known but the impact of which was not clearly understood until after first flight. Operational load profiles were tested successfully during ground tests but did not fully replicate the time dependent nature of the loading which turned out to be important to identification of a latent failure. With electrical loads being carried for a longer time period the PSUs output degraded to the point where operational loads were impeded, this was not a visible issue until the flaps were deployed on approach. The increased load exceeded the PSUs capability and voltage dropped, increasing current until the circuit breakers activated on BMU #1. A mechanical switching relay was initially used in the circuit design to provide essential bus power to the inverters, with the failure of PSUs in BMU #1 this dropped BMU#1 offline cutting power to two inverters or ½ of the motor. The mechanical relay activated to continue supplying essential bus power but the switching time was long enough for the two remaining inverters to also drop offline with one coming back online.

Post flight fault mode testing found that this failure mode was not easily replicated as it took a considerable amount of testing to accurately replicate the problem because it was time dependent. Initial testing of the full flight profile did not initially reveal the fault until several attempts were made. Root cause investigations were essential to isolating the issue and verifying the design fix with additional fault mode testing to show no additional latent failures or regression of the system.

A key lesson learned was that strict adherence to configuration control best practices is important for any flight test program. Configuration control becomes even more important for complex systems that rely significantly on software and firmware as well as hardware. The failure of the system in flight was due to complex interactions between components and not the failure of any single component. In this case no component failed catastrophically, instead a single electrical component had degraded performance under high load and temperature in a way that was ultimately predicable, but the effects were not easily predicted.

Good supplier management and team communication is also an essential part of any complex system design where no single person has the knowledge base of all components to understand every aspect of the system. This is increasingly true as we move forward with electric power train integration in aircraft. We must rely on testing fundamentals that in turn rely on traditional safety analysis techniques like Failure Modes and Effects Analysis (FMEA) and System Safey Analysis (SSA) approaches. These approaches do not require a complete understanding of the technology but the potential failure modes. This is not trivial work and does require knowledge of a system, but not at a fundamental level.

B. INTERFACE DESIGN & ENABLING HARDWARE

During the implementation of the ESS, the functionality of the legacy Aircraft Master Switch changed from a primary electrical system controller to a secondary electrical system supply switch. In the certified aircraft configuration, the Aircraft Master Switch was required to be in the ON position for electrical system components to be powered (radios, navigation, transponders, flaps, electric trim, internal & external lights, etc.). Following test integration modifications, all electrical system components were powered through the ESS. The Aircraft Master Switch functionality was relegated to connecting and disconnecting the 28VDC legacy aircraft battery to the ESS circuit. The legacy aircraft battery was only used to power the radio, external aircraft lighting, and flaps prior to the ESS being brought online.

Had the legacy aircraft battery been connected to the ESS circuit during the current surge demand from the flap motor activation, its possible the ESS disconnect and subsequent loss of engine power may have been avoided. During flight profile ground testing, the aircraft 28 VDC battery was proven to be more than capable of picking up surge current demands from flap motor activations during low ESS SOCs. Unfortunately, other than physically looking at the Aircraft Master Switch position, no feedback loop existed to alert the team that the aircraft 28 VDC battery was not connected to the ESS circuit prior to the event.

C. CHECKLISTS

Frequent checklist interruptions occurred throughout the program. With only one seat in the aircraft, the pilot would have to vacate the cockpit when technicians were required to troubleshoot or reset systems. If switches were not returned to their previous position prior to the pilot returning or if switches got disturbed during seat changes, incorrect aircraft configurations could be missed prior to takeoff due to system display deficiencies and limited system status feedback loops. This was a contributing factor to the Aircraft Master Switch not being in the ON position during the first flight. During subsequent ground and flight tests, any time the pilot vacated the cockpit, checklists would be re-run from the beginning of the Preflight Checklist regardless of when the disruption occurred.

D. EMI

During EMI ground testing, magniDrive 4 was posting intermittent error messages. In addition, radio static during radio communications suggested the presence of EMI. Further investigation led to the source of the EMI. The legacy aircraft beacon was leading to radio static and the intermittent error messages posted on the magniDrive 4. Consideration should be given to insulating legacy equipment and wiring on future electric propulsion integration projects.

VI. CONCLUSION

The flight of the eCaravan was a tremendous success and a significant milestone achievement in the development and commercialization of electric aviation, not only the technical achievement of flying an electric airplane of this size and type for the first time, but also the quality of the lessons learned from the integration and operation of an aircraft in this configuration. These lessons have been instrumental in the further development of the magniX product line. This product line is now progressing towards certification as the magni650 based EPU, with simplified configuration, tighter integration of supporting systems and higher power ratings. The team at AeroTEC and the test pilot on the program were key in this program's success, extracting maximum technical value, and pushing the boundaries of aviation towards a cleaner, more affordable and accessible future.

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