

Bench-to-Flight Validation of a Passively Air-Cooled 90 Ah Battery Module for Cargo eVTOL: Power Margin and Post-Landing Heat-Soak Characteristics

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Abstract— Battery design for electric vertical takeoff and landing (eVTOL) aircraft requires quantitative verification of high-power capability against actual flight demand. In this study, a passively air-cooled 14S1P 90 Ah battery module was evaluated in bench tests and then installed unchanged in a cargo eVTOL aircraft for eight flight tests. The bench-verified 720 A (8C), 60 s discharge capability was compared with measured flight load on the same time scale. Relative to the maximum 60 s flight demand of 305.2 A, the current- and power-based capability-to-demand margins were 2.36 and 2.38, respectively. A model-based correction for the end-of-mission battery state yielded an estimated current margin of approximately 2.0. In all flights, the maximum measured temperature occurred 140–736 s after landing, with an additional rise of 4.9–10.1 K; the highest value recorded, 54.6 °C, remained below the 60 °C cell limit. Residual-current Joule heating was no more than 0.26% of in-flight Joule heat generation, suggesting that stored-heat redistribution and internal thermal relaxation contributed to the post-landing rise. A two-node thermal model reproduced in-flight stack-center temperature for five unseen flights with a root-mean-square error (RMSE) of 1.30 K but underpredicted the additional post-landing rise.

Index Terms— electric vertical takeoff and landing aircraft, flight test, lithium-ion battery module, passive thermal management, thermal model.

I. INTRODUCTION

Batteries for electric aircraft are required to provide both high specific energy and high specific power. Specific energy strongly affects flight range and endurance, and previous studies have reported that practical eVTOL operation requires a pack-level specific energy of approximately 230–300 Wh/kg [1], [2], [3], [4]. Specific power, on the other hand, constrains the power available during high-power flight phases such as takeoff and hover, and vertical takeoff and landing aircraft require higher transient and sustained power than during cruise [3], [5], [6], [7], [8]. Although extensive research has been conducted on electric-

aircraft propulsion systems, power-conversion systems, and mission-level energy optimization [9], [10], [11], [12], relatively few studies have directly measured the actual battery load during flight and quantitatively linked it to performance demonstrated in bench testing. High power demand also increases the thermal burden on the battery [7], [8], whereas the additional mass of the cooling system required to mitigate this burden reduces system-level specific energy [13], [14]. Therefore, the energy, power, and thermal-management performance of aircraft batteries should be evaluated in an integrated manner [13], [14], [15].

Experimental evaluation of these performance characteristics requires two distinct types of information: the available battery capability demonstrated in bench testing and the mission demand imposed when the same hardware is installed in an actual aircraft. Conventional bench-performance studies can quantify battery energy and power performance, but they do not directly show the operational margin that this performance provides for an actual flight mission. Conversely, flight-demonstration studies provide realistic operating conditions, but the required power cannot be directly compared with bench capability when battery-level current and voltage are not sufficiently measured and reported. In our previous work [16], a 24S NCM battery system was designed, fabricated, and demonstrated in flight, but the evaluation was limited to system-level functionality; neither a duration-matched comparison between bench-verified capability and measured flight demand nor the post-landing thermal behavior was addressed. The present study differs in the battery architecture (14S1P module for a cargo multicopter), and in that both the capability-to-demand margin and the post-landing heat-soak characteristics are quantified for the same physical unit. Previous studies investigating high-power operation of eVTOL batteries have also relied primarily on mission-load profiles reproduced under laboratory conditions [7], [8]. Publicly available eVTOL battery datasets are likewise based on laboratory-reproduced mission-load profiles rather than measurements obtained during actual flight [17]. Battery thermal models have also been validated mainly under laboratory conditions at the cell or pack level [18], [19]. For electric-vehicle battery packs, experimentally validated electrothermal models incorporating cell-to-cell variation and aging have been reported [20], while integrated electrothermal-aging modeling and design optimization have been investigated for liquid-cooled aircraft battery packs [21]. However, the

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thermal-prediction accuracy in [21] was evaluated against computational fluid dynamics results, and the applied load was based on a hypothetical mission profile; therefore, its validation differs fundamentally from validation using loads measured during actual flight. At the cell level, high-rate pulse tests have shown that substantial heat may continue to be released during thermal relaxation after current interruption [22]. However, how such post-load thermal relaxation manifests as module-level post-landing temperature behavior during actual eVTOL flight has not yet been sufficiently demonstrated.

This article links the capability verified in bench testing with the demand measured in flight for the same physical battery module. The contributions of this work are threefold. First, a two-step procedure is proposed to compare capability and demand on a consistent basis: a moving-average window equal to the bench test duration matches the time scale, and a 60 s equivalent resistance together with the post-mission open-circuit voltage transfers the full-charge capability to the state at which peak demand occurs. Second, measurement is extended beyond load termination, which allows post-landing heat soak to be quantified; conventional high-rate discharge tests terminate too early to observe this behavior. Third, a two-node lumped-parameter thermal model driven by measured flight current is validated on flights withheld from identification, and its limits in predicting the post-landing peak are quantified.

It should be emphasized that the 720 A (8C), 60-s discharge condition used in this study is not intended to represent a general requirement for eVTOL batteries, but rather a bench-verified test point for the present module. The hover load of the test aircraft was approximately 2.41C, while the maximum measured load was 3.65C. Accordingly, the results of this study should be interpreted as the power margin between the verified bench capability and the flight envelope measured in the present experiments.

II. BATTERY MODULE AND EXPERIMENTAL METHODS

A. Battery-module Configuration and Mass Boundary

The battery module for the cargo eVTOL consisted of 14 commercial high-nickel NCM pouch cells connected in a 14S1P configuration, and the main cell and module specifications are summarized in Table I. Aluminum heat spreaders and acetal separators were alternately arranged between adjacent cells, providing both structural support and heat-transfer paths between cells. No active cooling device or dedicated coolant passage was employed; instead, the module relied on passive cooling by the ambient airflow around the module during flight. The external configuration and internal cell-stack structure of the module are shown in Fig. 1.

To evaluate the specific energy of the battery module for the cargo eVTOL, the mass boundary must be clearly defined. In this study, the total mass of the energy-storage assembly actually installed on the aircraft was used as the basis for calculating specific energy. This mass includes not only the cells, but also the battery management system, busbars, and supporting structures; no separate active-cooling system is included. Two test specimens were fabricated for room-

temperature performance evaluation, and the measured masses of Specimens A and B were 18.295 kg and 18.320 kg, respectively. All module-level specific-energy values reported in this study were calculated using this mass boundary. The detailed module configuration and mass distribution are summarized in Table I.

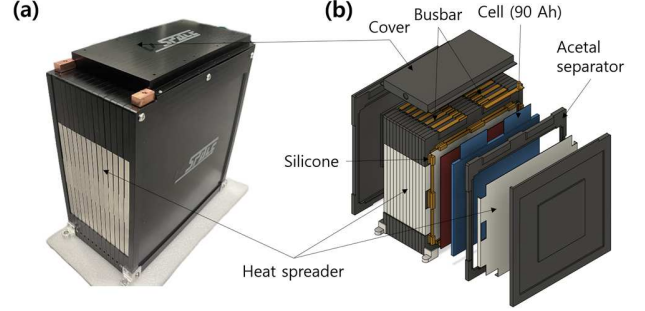


Fig. 1. Passively air-cooled 14S1P 90 Ah battery module for cargo eVTOL applications: (a) assembled battery module and (b) internal cell-stack configuration.

TABLE I

Cell and module specifications and mass boundary of the 14S1P 90 Ah battery module

Category	Parameter	Value
Cell	Chemistry / format	High-Ni NCM / pouch
	Nominal capacity / voltage	90 Ah / 3.8 V
	Cell mass	1.05 kg
Module	Configuration / nominal voltage	14S1P / 53.2 V
	Charge-voltage limit	60.2 V performance evaluation / 58.8 V operation
	Discharge-voltage limit	42.0 V
	Thermal management	Passive air cooling
	Measured module mass	18.295 kg (specimen A)
	Cell-to-module mass ratio	80.4%

Prior to module assembly, the 0.5C discharge capacity and 1 kHz ac internal resistance of each cell were measured. The incoming-inspection criteria were set to a discharge capacity of at least 85.0 Ah and an internal resistance of no more than 1.6 mΩ. Cells that satisfied these criteria were grouped so that cells with similar internal-resistance characteristics were assigned to the same module. The cells used in the modules for performance testing had discharge capacities of 85.00–87.56 Ah and internal resistances of 1.084–1.552 mΩ. However, because no control module assembled without cell screening was tested in this study, the screening procedure should not be interpreted as demonstrating improved electrical or thermal uniformity of the module. Cell screening is therefore described only as part of the module-assembly procedure adopted in this study.

B. Bench-Test Setup and Procedures

The bench performance tests of the battery module for the cargo eVTOL were conducted in an environmental test chamber. The test specimens were connected to a constant-voltage/constant-current charger and a dc electronic load, and pack voltage, current, and multichannel temperature data were recorded at 1 Hz using a regenerative pack cycler. The bench-test program consisted of an energy performance evaluation using a 9.0 A constant-current discharge and a 720 A, 60 s constant-current high-rate discharge test. All tests were conducted at room temperature, and the test setup is shown in Fig. 2. The bench-performance values reported in this paper were calculated from the raw measurement data acquired during the tests.

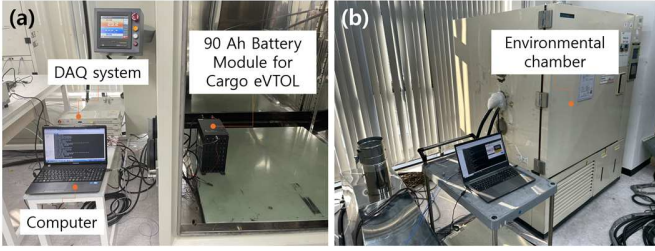


Fig. 2. Bench-test configuration: (a) the module installed inside the environmental chamber and (b) the chamber closed during testing.

Two modules of identical design were used for the energy and high-power discharge tests, and Specimen A was subsequently installed in the aircraft for the flight tests without replacement. Therefore, the bench capability–flight demand comparison in this study was performed using the same physical module rather than different modules, thereby eliminating inter-module variation between the bench and flight tests. This same-unit test configuration provides the experimental basis for directly linking the power capability demonstrated in bench testing to the power demand measured during actual flight.

C. Flight-Test Setup, Instrumentation, and Test Matrix

The cargo eVTOL flight tests were conducted using a fully electric cargo multirotor aircraft with a maximum takeoff mass of 110 kg, equipped with a single 14S1P 90 Ah battery module. The test aircraft employed a coaxial counter-rotating multirotor configuration with four foldable arms and eight rotors, and the battery module was installed at the lower center of the fuselage. The instrumentation and payload were also positioned near the center of the aircraft to minimize changes in the center of gravity. A total of eight flight tests were conducted within a seven-day period. The main specifications of the test aircraft and the battery installation location are presented in Table II and Fig. 3, respectively. The actual takeoff mass during the tests ranged from 72.30 to 89.85 kg, and an additional 16 kg payload was applied under the maximum-payload test condition.

Flight-test measurements were acquired through two independent systems: a dedicated data logger and the aircraft flight-control system. The data logger recorded total pack voltage and current, four channels of electronic speed controller

(ESC) temperature, three channels of module temperature, and ambient temperature at 10 Hz, while barometric altitude was obtained from the flight-control logs. Pack current was measured using a unidirectional Hall-effect current sensor rated at 500 A, and temperatures were measured using K-type thermocouples. The maximum current recorded across the eight flights, 328.5 A, is 66% of this rating, so no sensor saturation or clipping occurred at the current peaks. Of the three module-temperature channels, two were attached to the left and right heat spreaders, while the remaining channel was inserted between Cells 6 and 7 at the center of the 14-cell stack. Because the stack-center channel exhibited the highest measured temperature in all flights, it was used as the representative maximum measured temperature location in the subsequent thermal analysis. However, because temperature was measured at only three locations, the maximum temperature reported in this study refers to the highest value among these three measurement points rather than the physical maximum temperature within the entire module.

TABLE II

Specifications of the cargo eVTOL test aircraft

Parameter	Specification
Aircraft configuration	Coaxial counter-rotating multirotor with four foldable arms and eight rotors
Overall dimensions (L × W × H)	1,970 × 1,970 × 732 mm
Propeller	40 in (1.016 m) diameter, 13.1 in pitch
Rotor arrangement	4 CW + 4 CCW
Effective rotor-disk area	3.243 m ²
Maximum takeoff mass	110 kg
Empty mass including battery	60.2 kg
Battery module	14S1P, 90 Ah
Battery thermal management	Passive air cooling

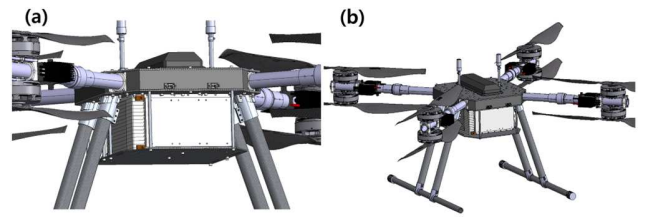


Fig. 3. Cargo eVTOL test aircraft and battery installation: (a) battery-module installation beneath the fuselage and (b) overall aircraft configuration.

The flight procedure was kept as consistent as possible across all tests. Before takeoff, data were recorded for several minutes with the motors stopped to establish the current-sensor zero and the initial thermal state. After takeoff, the aircraft hovered at the designated altitude for approximately 10 min before landing. Measurements were continued for at least 20 min after landing to capture the thermal response of the module following load termination. During this period, the module was not connected to a charger, and the measured current

corresponded to the residual discharge current while the aircraft electrical system remained powered. Therefore, the post-landing temperature rise analyzed in this study does not include heat generated during charging. Measurement uncertainty was bounded from the records themselves rather than from a complete budget. Detrended residual standard deviations over the pre-takeoff stationary period were 0.10 K at the stack center, 0.10–0.12 K at the heat spreaders, 0.33 K for ambient temperature, and 0.001 V for pack voltage. The current-sensor zero offset, taken as the difference between the 30 s pre-takeoff mean and the stationary value at the end of each record, was 0.3–10.5 A; applying it as an upper bound shifts the margins of 2.36 and 2.00 reported in Sections III-C and III-D to 2.28–2.44 and 1.93–2.07, respectively.

TABLE III

Flight-test conditions and training–validation split for the thermal model

Flight	Takeoff mass (kg)	Payload (kg)	Flight profile	Thermal-model dataset
Flight 1	73.80	0	Nominal hover	Validation
Flight 2	72.30	0	Nominal hover	Training
Flight 3	73.85	0	Nominal hover	Training
Flight 4	73.80	0	Nominal hover, lower ambient temperature	Validation
Flight 5	73.85	0	Nominal hover	Training
Flight 6	89.85	16	Payload hover	Validation
Flight 7	73.85	0	Hover followed by three climb–descent cycles	Validation
Flight 8	73.80	0	Nominal hover	Validation

The same charging procedure was applied before each test: 270 A constant current, transition to constant voltage at 58.8 V, and termination when the current remained at or below 1.8 A for 1 s. At termination, the pack voltage and current were 58.800 V and 1.720 A, and the total charging time was 32.6 min. The battery was then rested for at least 30 min. The pre-takeoff open-circuit voltages of the eight flights ranged from 58.03 to 58.33 V, indicating that the initial battery state was maintained within a relatively narrow range. Table III summarizes the takeoff mass, payload, and training–validation split for the thermal model for each flight. Flights 1–5 and 8 were conducted primarily under nominal hover conditions, Flight 6 included an additional 16 kg payload, and Flight 7 employed a dynamic load profile consisting of three climb–descent cycles between 10 and 30 m following the initial hover segment.

III. BENCH-TO-FLIGHT BATTERY PERFORMANCE ASSESSMENT

A. Bench Energy and Power Capability

The bench performance of the battery module was evaluated through an energy performance test and an 8C (720 A), 60 s high-rate discharge test. The energy performance test was

conducted under a 0.1C (9.0 A) discharge condition, and the measured discharge capacities of Specimens A and B were 87.286 Ah and 88.689 Ah, respectively. The discharge energies, calculated by integrating the measured voltage and current over time, were 4,437.1 Wh and 4,507.0 Wh, corresponding to module-level specific energies of 242.53 Wh/kg and 246.02 Wh/kg, respectively, based on the module masses defined in Section II-A. The specific energies calculated using the nominal voltage and measured discharge capacity were 253.81 Wh/kg and 257.49 Wh/kg, respectively. In this study, the specific energy obtained from time integration of the measured discharge data was used as the primary performance metric because it accounts for the voltage variation throughout the actual discharge process. The main bench-performance results are summarized in Table IV.

TABLE IV

Bench performance evaluation results for the two test specimens

Parameter	Specimen A	Specimen B
Energy performance evaluation (9.0 A discharge)		
Module mass (kg)	18.295	18.320
Discharge capacity (Ah)	87.286	88.689
Discharge energy (Wh)	4,437.1	4,507.0
Specific energy based on measured energy (Wh/kg)	242.53	246.02
High-rate discharge performance (720 A, 8C, 60 s)		
Test temperature (°C)	21.9	20.9
Open-circuit voltage before discharge (V)	58.513	58.740
Terminal voltage at 60 s (V)	48.137	49.004
Average power over 60 s (kW)	35.96	36.46
Average specific power over 60 s (kW/kg)	1.97	1.99
$R_{1s}(m\Omega)$	9.402	9.010
$R_{60}(m\Omega)$	14.41	13.52

For the energy performance evaluation, the module charge-voltage limit was set to 60.2 V, corresponding to the cell manufacturer’s maximum charge voltage of 4.30 V/cell. In contrast, the charge-voltage limit during actual flight operation was restricted to 58.8 V (4.20 V/cell) to provide additional voltage, thermal, and cycle-life margins. Accordingly, the specific energy values reported in this section were evaluated under the condition of charging up to 60.2 V and should not be interpreted as the usable energy available over the actual flight operating voltage range. Hereafter, 60.2 V is referred to as the performance-evaluation charge limit, whereas 58.8 V is referred to as the flight-operation charge limit.

The high-rate discharge performance shown in Fig. 4 was evaluated by applying a constant current of 720 A for 60 s to both specimens under fully charged, room-temperature conditions. The 720 A current corresponds to 8C based on the

nominal capacity of 90 Ah and to 8.25C and 8.12C for Specimens A and B, respectively, based on their measured discharge capacities. Both specimens completed the entire 60 s discharge without low-voltage cutoff or activation of any protection function. The 60 s average powers of Specimens A and B were 35.96 and 36.46 kW, respectively, corresponding to average specific powers of approximately 1.97 and 1.99 kW/kg based on the module mass of approximately 18.3 kg. The voltage and current responses during the 720 A, 60 s test are shown in Fig. 4. Because this test point represents a power capability directly verified under fully charged, room-temperature conditions, the available power under a partially discharged state representative of actual flight operation is separately corrected and evaluated in Section III-D. The 1 s equivalent resistances, R_{1s} , obtained from the high-rate discharge tests were 9.402 m Ω and 9.010 m Ω for Specimens A and B, respectively. Their mean value of 9.2 m Ω was used in Section IV as the module-level equivalent resistive heat-generation coefficient, thereby linking the bench electrical performance to the thermal behavior observed during actual flight.

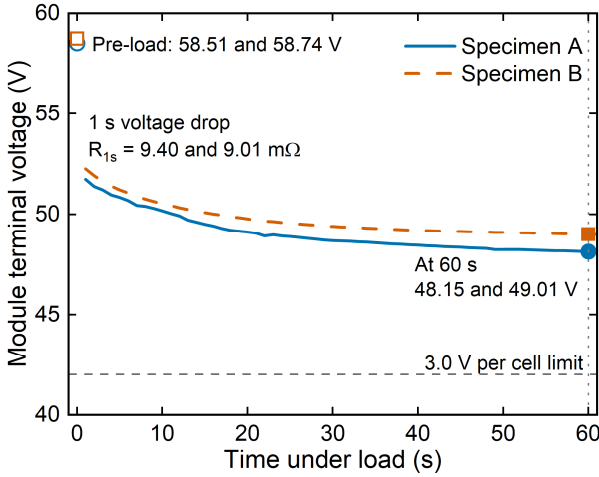


Fig. 4. Validation of 720 A (8C) 60-s high-rate discharge capability.

B. Flight Power Demand and Repeatability

Fig. 5 shows the time histories of pack current and barometric altitude measured during a representative flight. Immediately after takeoff, the current increased sharply and then remained relatively stable at approximately 210–235 A during the hover segment, before decreasing rapidly upon landing. To exclude the effects of the initial-flight and landing transients, the hover segment was defined as the period from 25 s after takeoff to 15 s before landing.

To evaluate repeatability under the nominal hover condition, five flights conducted under comparable ambient conditions (Flights 1, 2, 3, 5, and 8), with takeoff masses ranging from 72.3 to 73.9 kg, were compared; Flight 4 was excluded from this set because it was conducted at a lower ambient temperature. The mean pack current during hover was 216.6 ± 3.1 A, with a coefficient of variation of 1.4%,

corresponding to approximately 2.41C based on the nominal capacity of 90 Ah. The mean pack power was 11.36 ± 0.17 kW, the charge consumed during flight was 39.10 ± 0.55 Ah, and the flight duration was 651.3 ± 4.6 s. Despite being conducted on different days and under different ambient conditions, the flights showed only limited scatter in the principal flight metrics, indicating good repeatability of both the flight procedure and the measured results under the present test conditions.

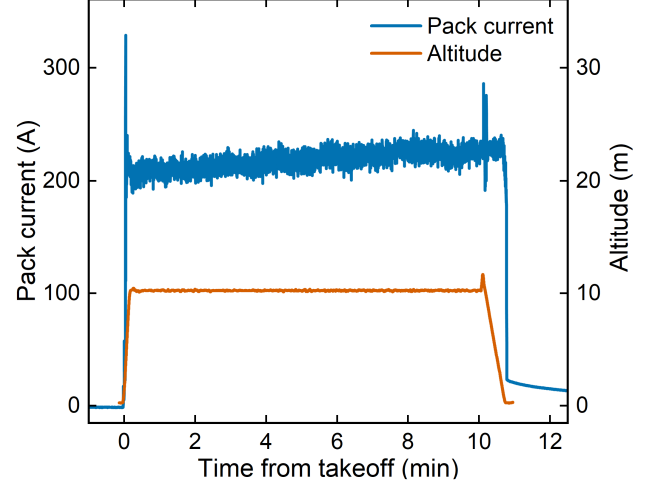


Fig. 5. Representative flight profile and pack-current demand.

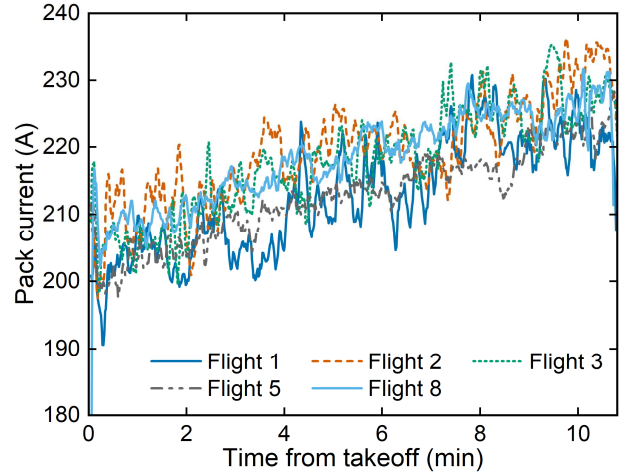


Fig. 6. Repeatability of hover-current demand over five nominal flights.

Fig. 6 compares the hover-current profiles of the five flights, for which a 5 s moving average was applied to reduce the influence of short-term current fluctuations. The current histories remained at similar levels throughout the hover segments, showing only limited scatter around the mean value of 216.6 ± 3.1 A. These repeated-flight results were subsequently used as the reference nominal-hover load when comparing the maximum flight demand with the bench-verified power capability.

C. Time-Scale-Matched Bench-to-Flight Power Margin

The 720 A power capability demonstrated in the bench test

corresponds to a test point sustained for 60 s. Therefore, to directly compare this capability with actual flight demand, the flight load must be evaluated on the same time scale. Because an instantaneous current peak and a high-power load sustained over a finite duration impose different demands on the battery, moving-average currents were calculated over the entire airborne segment of each flight. The moving-average current for a window length τ was defined as

$$\bar{I}_\tau(t) = \frac{1}{\tau} \int_{t-\tau}^t I(\xi) d\xi \quad (1)$$

where $I(t)$ is the measured pack current and τ is the moving-average window length. The maximum flight demand at each time scale was then defined as

$$I_{\text{dem},\tau} = \max_t [\bar{I}_\tau(t)] \quad (2)$$

In this study, $\tau = 1, 5, 10, 30,$ and 60 s were considered. This procedure distinguishes flight loads of different durations and enables direct comparison with the 60 s bench-test point on a consistent time basis.

Across the eight flights, the maximum moving-average currents for window lengths of 1, 5, 10, 30, and 60 s were 312.5, 310.6, 309.0, 305.7, and 305.2 A, respectively, and all occurred during Flight 6 with the additional 16 kg payload. In contrast, the maximum instantaneous current in the raw 10 Hz data was 328.5 A and occurred during the takeoff transient of Flight 8. However, the maximum 1 s moving-average current in the same flight was only 247.6 A, indicating that the instantaneous peak represented a very short takeoff transient rather than a sustained power demand. Therefore, the maximum 60 s moving-average current, rather than the instantaneous peak current, was used when comparing the 60 s bench capability with the sustained flight-power demand.

Fig. 7 compares the maximum current demand of each flight as a function of moving-average window length with the directly verified 720 A, 60 s bench-test point. Flight 6 exhibited the highest demand at all moving-average time scales, whereas the mean load of the five repeated nominal-hover flights, 216.6 ± 3.1 A, remained lower. On the same 60 s basis, the current capability-to-demand margin was defined as $M_I = I_{\text{cap},60} / I_{\text{dem},60}$. The directly verified 60 s current capability was $I_{\text{cap},60} = 720$ A, while the maximum measured 60 s flight demand across the eight flights was $I_{\text{dem},60} = 305.2$ A. Accordingly, $M_I = 720/305.2 = 2.36$.

The same time-scale matching was applied to power. The lower of the two bench 60 s average powers, 35.96 kW, was conservatively adopted as the bench capability. Because the maximum 60 s flight-power demand was 15.08 kW, the power capability-to-demand margin was $M_P = P_{\text{cap},60} / P_{\text{dem},60} = 35.96/15.08 = 2.38$. The maximum 60 s flight-power demand of 15.08 kW corresponds to 824 W/kg based on a module mass of approximately 18.3 kg. Therefore, on the same 60 s time scale, both the current- and power-based bench-to-flight margins were approximately 2.4.

This comparison matches only the time scale; the state difference is corrected in Section III-D. The margins of 2.36 and 2.38 should therefore be interpreted as comparisons

between the fully charged bench capability and the flight demand occurring late in the mission.

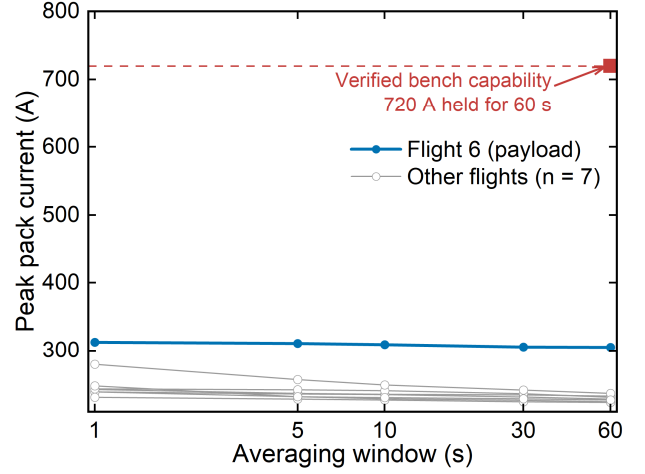


Fig. 7. Maximum flight-current demand as a function of averaging window.

D. State-Corrected Capability-to-Demand Margin at End of Mission

Although the capability-to-demand comparison in Section III-C matched the time scale at 60 s, the battery states during the bench and flight tests were not identical. The 720 A bench capability was verified under fully charged, room-temperature conditions, whereas the maximum 60 s flight demand of 305.2 A occurred during Flight 6 between 588 and 648 s after takeoff, by which time 52.15 Ah had been consumed. Therefore, to evaluate the power margin near the end of the actual mission, the bench-measured capability must be corrected to the corresponding operating state.

For this purpose, the 60 s equivalent resistance R_{60} was defined from the 720 A, 60 s bench test of Specimen A as

$$R_{60} = \frac{V_{\text{oc},0} - V_{60}}{I_{\text{test}}} \quad (3)$$

where $V_{\text{oc},0}$ is the rested open-circuit voltage immediately before discharge, V_{60} is the terminal voltage after 60 s of discharge at 720 A, and I_{test} is the test current. For Specimen A, $V_{\text{oc},0} = 58.513$ V and $V_{60} = 48.137$ V, yielding $R_{60} = 14.41$ m Ω . This value should not be interpreted as a purely ohmic resistance; rather, it is an equivalent coefficient representing the terminal-voltage reduction over the 60 s interval and therefore includes both resistive voltage drop and the voltage change occurring during the pulse.

The current that could be drawn for 60 s at the end-of-mission state was estimated using the pack open-circuit voltage $V_{\text{oc,EOM}}$, measured after the aircraft electrical system had been powered off and the battery had rested for more than 20 min after landing, together with the pack discharge-voltage limit $V_{\text{min}} = 42.0$ V:

$$I_{\text{cap},60}^{\text{EOM}} = \frac{V_{\text{oc,EOM}} - V_{\text{min}}}{R_{60}} \quad (4)$$

For Flight 6, the end-of-mission open-circuit voltage was 50.785 V. Applying Equation (4) gives an estimated 60 s

available current of approximately 610 A. Accordingly, the state-corrected current capability-to-demand margin was $610/305.2 = 2.00$. Thus, although the margin decreased from the value of 2.36 obtained in Section III-C using the directly measured fully charged capability, a margin of approximately 2 remained relative to the maximum measured 60 s flight demand. For the five repeated nominal-hover flights, the end-of-mission open-circuit voltages ranged from 52.02 to 52.22 V. Using the same procedure, the estimated 60 s available currents were 696–709 A, corresponding to margins of 3.01–3.15 relative to the maximum 60 s demand of each flight. The full-charge and end-of-mission margins are summarized in Fig. 8.

Using the same model, if 720 A were sustained for 60 s at the end-of-mission state of Flight 6, the predicted pack voltage would decrease to approximately 40.4 V, below the specified discharge-voltage limit of 42.0 V. Therefore, the 720 A test point directly verified at full charge should not be interpreted as being directly available at the end-of-mission state. It should be emphasized that the 610 A value was not directly measured in a 60 s discharge test at the end-of-mission state; rather, it is a state-corrected estimate derived from the open-circuit voltage and the bench-derived R_{60} . This approach does not independently account for the temperature and state-of-charge dependence of resistance, the dynamic behavior of polarization, cell-to-cell imbalance, or the minimum voltage of individual cells; the resistance increase expected at a lower state of charge is, however, partly offset by the elevated module temperature at the end of the mission. In addition, the calculated current margin is defined at the pack-voltage level and does not guarantee that individual-cell voltage or temperature limits would not be reached first.

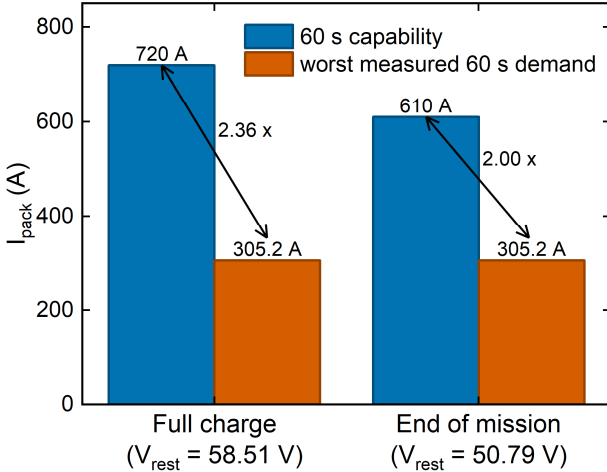


Fig. 8. Measured and estimated capability-to-demand margins at full charge and end of mission.

E. Takeoff-Mass Effect on Hover Power

To evaluate the effect of increased payload on hover-power demand, the nominal hover condition was compared with Flight 6, which carried an additional 16 kg payload, as shown in Fig. 9, in which all eight flights are plotted. The mean takeoff mass of the five nominal-hover flights was 73.52 ± 0.68 kg, whereas

the takeoff mass of Flight 6 was 89.85 kg, representing an increase of 22.2%. Accordingly, the mean hover current increased by 34.1%, from 216.6 to 290.4 A, while the pack power increased by 30.9%, from 11.36 to 14.87 kW.

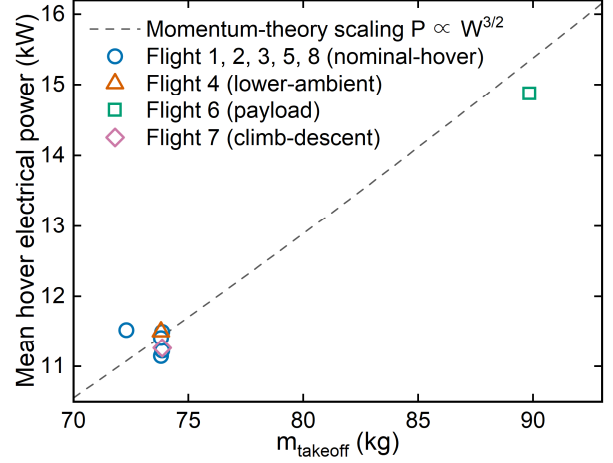


Fig. 9. Hover-power scaling with takeoff mass and comparison with momentum theory.

Under hover conditions, the ideal induced power is proportional to thrust raised to the power of 3/2 according to momentum theory [23]. Because the hover thrust for the same aircraft can be taken as equal to its weight, the power ratio between the two weight conditions can be expressed as

$$\frac{P_{i,2}}{P_{i,1}} = \left(\frac{W_2}{W_1}\right)^{3/2} \quad (5)$$

Applying this relationship, a 22.2% increase in takeoff mass is predicted to increase the ideal induced power by approximately 35.1%. The measured pack-power increase was 30.9%, which was 4.2 percentage points lower than the theoretical prediction, while the overall trend in power demand with increasing mass was broadly consistent with momentum theory. Because momentum theory predicts power rather than current, the measured power increase of 30.9%, rather than the current increase of 34.1%, was used for comparison. The larger increase in current resulted from the reduction in operating voltage as the payload increased. The minimum pack voltage under the nominal-hover condition was 49.93 ± 0.17 V, whereas that of Flight 6 decreased to 48.38 V.

IV. POST-LANDING THERMAL BEHAVIOR AND MODELING

A. Post-Landing Heat-Soak Characteristics

Fig. 10 shows the measured temperature histories and post-landing heat-soak behavior during representative repeated-hover Flight 8. The stack-center temperature increased from an average of 27.7 °C during the ground-recording period to 37.3 °C at landing. The left and right heat-spreader temperatures increased by averages of 6.1 K and 7.3 K, reaching average maxima of 33.8 °C and 34.5 °C, respectively. Across all flights, the ambient temperature ranged from 25.7 to 31.1 °C. The stack-center temperature was consistently the highest among the three measurement locations, and the

temperature difference between the stack center and the heat spreaders at landing was approximately 6–11 K. This indicates that heat generated during flight was stored nonuniformly within the module.

However, the maximum measured temperature did not occur at landing. In all eight flights, the stack-center temperature continued to increase after landing and reached its maximum 140–736 s later, with an additional rise of 4.9–10.1 K relative to the landing temperature. In particular, during Flight 6 with the additional 16 kg payload, the stack-center temperature increased from 44.6 °C at landing to 54.6 °C after 257 s, corresponding to the largest additional rise of 10.1 K. Even this maximum of 54.6 °C remained below the 60 °C upper limit of the cell manufacturer's specified discharge temperature range, indicating that passive cooling was sufficient under the conditions tested. Under the climb–descent condition of Flight 7, the temperature increased from 39.9 °C at landing to 49.8 °C after 452 s, corresponding to an additional rise of 9.9 K. For the nominal-hover flights, the post-landing temperature rise ranged from 4.9 to 8.4 K.

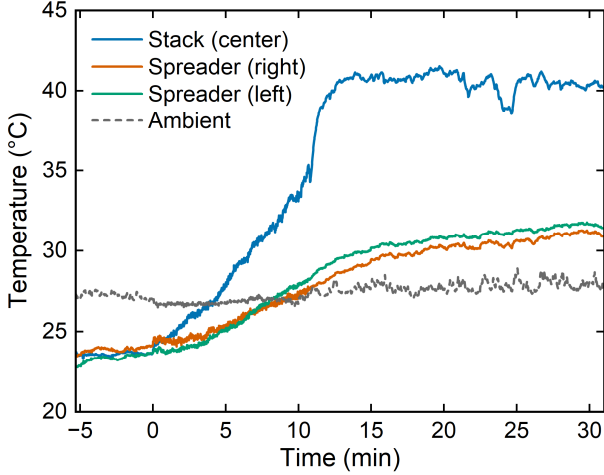


Fig. 10. Measured temperature histories and post-landing heat-soak behavior during representative repeated-hover Flight 8.

To determine whether this post-landing temperature increase could be explained by additional heating from residual current, the post-landing current was examined. During the post-landing recording period, the module was not connected to a charger, and the measured current corresponded to the residual discharge current while the aircraft electrical system remained powered. The mean residual current from landing until the time of maximum temperature ranged from –4.6 to 18.1 A among the flights.

The resistive heat generation was calculated using the mean 1 s equivalent resistance of the two bench specimens, 9.2 mΩ, as the equivalent heat-generation resistance R_{mod} :

$$Q_J(t_1, t_2) = R_{\text{mod}} \int_{t_1}^{t_2} I^2(t) dt \quad (6)$$

Using Equation (6), the resistive heat generated from landing to the time of maximum temperature was 74–752 J (0.02–0.21 Wh) across the eight flights, corresponding to only

0.01–0.26% of the in-flight resistive heat generation of 271–506 kJ (75–140 Wh) for the corresponding flights. Converting this energy using the combined heat capacity of the two nodes, $C_1 + C_2 = 5,964 \text{ J/K}$ (Section IV-B), gives a maximum equivalent temperature rise of 0.13 K, which is less than approximately 3% of the observed 4.9–10.1 K post-landing increase. Therefore, Joule heating associated with the residual post-landing current alone cannot explain the observed additional temperature rise.

These results suggest that redistribution of heat stored within the module during flight, internal thermal relaxation after load termination, and reduced convection once the rotor downwash ceased contributed to the post-landing temperature rise. This interpretation is also consistent with cell-level high-rate pulse-test results showing that heat release can continue during thermal relaxation after current interruption [22].

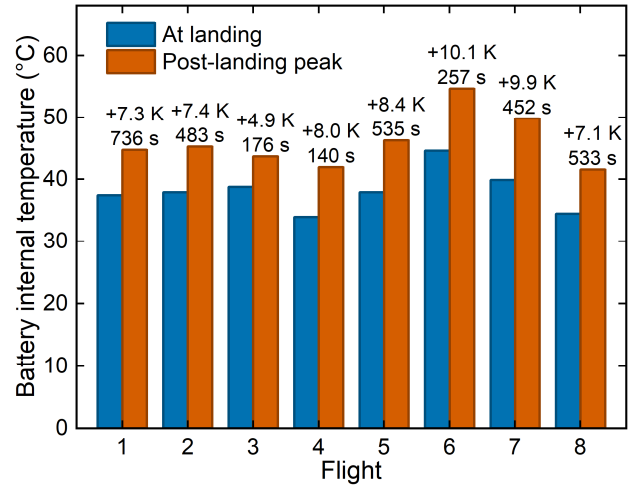


Fig. 11. Post-landing temperature rise and peak delay over eight flights.

The additional post-landing temperature rise and the delay to the maximum measured temperature for all eight flights are compared in Fig. 11. The observation of a post-landing temperature increase in every test indicates that the temperature at landing does not necessarily represent the maximum measured temperature for the corresponding mission. In particular, conditions associated with increased thermal loading, such as higher payload or elevated current demand near the end of the mission, also exhibited larger additional temperature rises after landing. However, because the number of flights and operating conditions examined in this study were limited, these results should be interpreted as trends observed under the tested conditions rather than as a general quantitative relationship between load history and post-landing temperature rise.

These results highlight the importance of extending the measurement period beyond load termination in battery testing and operational procedures. If the measurements in the present tests had been terminated at landing, the actual maximum measured temperature could have been underestimated by approximately 5–10 K. In addition, because heat redistribution within the module continued for several minutes after landing,

subsequent operations such as reflight or charging should consider not only the temperature at landing but also the ensuing thermal transient.

B. Two-Node Thermal Model Formulation and Identification

To predict the measured stack-center temperature from the flight-current history, a two-node lumped-parameter thermal model was employed. The first node, T_1 , represents an equivalent heat-generation region, while T_2 represents the measured stack-center temperature between Cells 6 and 7. The governing equations are

$$C_1 \frac{dT_1}{dt} = R_{\text{mod}} I^2 - \frac{T_1 - T_2}{R_{12}} \quad (7)$$

$$C_2 \frac{dT_2}{dt} = \frac{T_1 - T_2}{R_{12}} - \frac{T_2 - T_{\text{amb}}}{R_{2a}} \quad (8)$$

where T_{amb} is the ambient temperature, C_1 and C_2 are equivalent thermal capacitances, R_{12} is the thermal resistance between the two nodes, and R_{2a} is the resistance from the measurement node to ambient. Heat generation was represented by $I^2 R_{\text{mod}}$, with $R_{\text{mod}} = 9.2 \text{ m}\Omega$ fixed from the mean 1 s effective dc resistance of the two bench specimens. The two-node structure was adopted as a minimal representation capable of reproducing delayed post-landing heat transfer from the internal heat-generation region to the measured stack-center location. Higher-order and distributed-parameter models have also been proposed for spatial temperature resolution [24], [25], but the three available measurement points do not support their identification.

Flights 2, 3, and 5, all nominal-hover flights, were used for parameter identification. The parameters R_{12} , R_{2a} , C_1 , and C_2 were obtained by nonlinear least squares using the complete record from the pre-takeoff ground period to 20 min after landing, without segment weighting. The state equations were integrated using forward Euler with $\Delta t = 0.1 \text{ s}$, and the initial conditions were set as $T_1(0) = T_2(0) = T_{\text{meas}}(0)$.

Optimization was performed in log space using the trust-region reflective method to ensure positive parameter values. The identified parameters were $R_{12} = 1.036 \text{ K/W}$, $R_{2a} = 0.4233 \text{ K/W}$, $C_1 = 5,417 \text{ J/K}$, and $C_2 = 547.2 \text{ J/K}$. These are equivalent model parameters and should not be interpreted as one-to-one physical properties of specific cells or material regions.

C. Flight Validation and Prediction Limits

Flights 1, 4, 6, 7, and 8, spanning nominal-hover, lower-ambient, payload, and climb-descent conditions, were used exclusively for validation. The coupled two-node model exhibited thermal time constants of 7,972 and 163 s, representing slow internal heat redistribution and a comparatively faster heat-rejection response to the ambient.

During the in-flight period from takeoff to landing, the model achieved a root-mean-square error (RMSE) of 1.30 K for the five validation flights (0.66 K for the three training flights). During the post-landing period, the RMSE was 1.29 K for the four validation flights excluding Flight 7 and 4.94 K for Flight 7. The corresponding maximum-temperature errors were -0.90 to -2.40 K and -6.26 K , respectively, while the peak-time error

ranged from -208 to $+309 \text{ s}$.

Fig. 12 compares the measured and predicted stack-center temperatures for representative validation flights. The model reproduced the in-flight temperature trends reasonably well, whereas the post-landing temperature rise was systematically underpredicted, particularly for the climb-descent condition of Flight 7.

For the four validation flights excluding Flight 7, the measured post-landing temperature rise was 7.1 – 10.1 K , whereas the model predicted 3.5 – 5.7 K , corresponding to 46 – 55% of the measured rise. Thus, although the model reproduced the overall in-flight temperature response reasonably well, it systematically underpredicted the magnitude of post-landing heat soak.

One- and three-node models were identified using the same training-validation split. The one-node model gives an in-flight RMSE of 3.01 K and reproduces none of the post-landing rise, whereas in the three-node model the added capacitance degenerates to 4.8 J/K and the condition number of the Jacobian in log-transformed parameters rises from 1.10×10^2 to 9.19×10^6 , with a maximum parameter correlation of 1.000 . The underprediction therefore reflects the model order supported by three-point temperature sensing rather than a failure of the optimization.

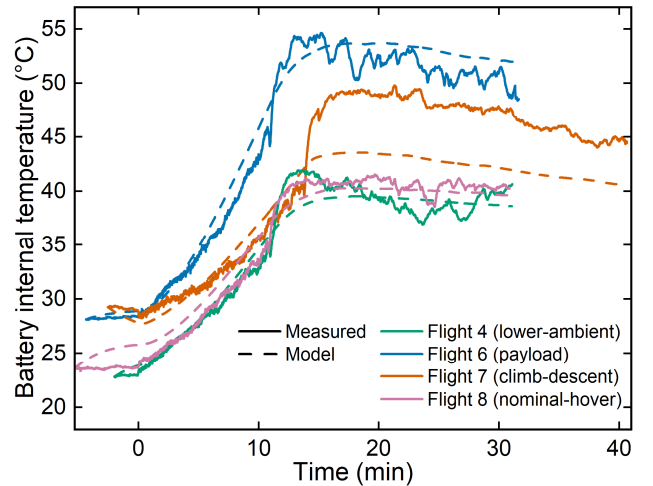


Fig. 12. Representative validation results of the two-node thermal model under unseen flight conditions.

The present results are limited to eight flights using the same aircraft and battery module, with takeoff masses of 72.3 – 89.9 kg and ambient temperatures of 25.7 – $31.1 \text{ }^\circ\text{C}$. In addition, the end-of-mission 60 s capability was model-estimated rather than directly measured, and module temperature was monitored at only three locations.

V. CONCLUSIONS

This study linked bench testing with eight flight tests of the same physical 14S1P 90 Ah passively air-cooled high-nickel NCM module, evaluating the bench-verified power capability, flight demand, and post-landing thermal behavior.

First, when the time scales of the bench and flight tests were matched at 60 s , the directly verified 720 A bench

capability under fully charged, room-temperature conditions provided current- and power-based margins of 2.36 and 2.38, respectively, relative to the maximum 60 s flight demand of 305.2 A. Even after correction for the battery state near the end of the mission, the estimated current margin remained approximately 2.00. The 720 A value represents a verified test point rather than the maximum power capability of the module; therefore, these margins should be interpreted relative to that verified test point.

Second, in all eight flights, the maximum measured temperature occurred 140–736 s after landing rather than at landing, with an additional post-landing temperature rise of 4.9–10.1 K. Residual-current Joule heating accounted for less than 3% of this rise, indicating that redistribution of heat stored during flight was the dominant mechanism. The highest value recorded, 54.6 °C, remained below the 60 °C cell limit; actual operation should nevertheless consider the post-landing transient rather than the temperature at landing.

Third, the two-node lumped-parameter thermal model driven by measured flight current reproduced the in-flight stack-center temperature with an RMSE of 1.30 K for the five flights not used for training. However, for the four validation flights excluding Flight 7, the model reproduced only 46–55% of the measured additional post-landing temperature rise, demonstrating the limitations of directly using such a low-order model for precise prediction of the post-landing peak temperature. Future work should therefore include performance testing over a broader flight envelope and a wider range of state-of-charge and temperature conditions, together with thermal-model validation using multi-point temperature measurements.

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