Model-Based Efficient and Safe Spacecraft Operations Planner  
Focusing on Battery State Management

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There is an increase in the popularity of small spacecraft because they are expected to introduce large opportunities in the space industry. However, there are two issues with increased use of small spacecraft: enhancement of cost-effectiveness and improvement in the mission failure rate. This study defines the battery state as the most important parameter in spacecraft operations because it directly relates to the health status of the spacecraft. We propose solving problems by introducing high-accuracy in situ state estimations for real-time operation mode selection and on-ground future state estimations of the battery in spacecraft operations planning by utilizing a suitable numerical model. The high-accuracy in situ state estimations allow us to minimize operational margins for battery protection, improve the efficiency of each spacecraft, and achieve safe spacecraft operations. The combination of a model-based system development approach and numerical optimization methods helps establish an efficient autonomous mission operations planner system to maximize mission duration. This enables us to reduce the human resource costs for spacecraft operations. This study addresses two topics: battery and spacecraft modeling, and establishing a mission operations planner system. The details of the system and battery modeling considering temperature dependence are explained using parameter identification results from the experiments. The design policy of the mission operations planner, which utilizes the battery status as the threshold of the operations and constraints, is explained via numerical optimization details. Numerical and experimental case studies are performed to evaluate the proposed method and show its effectiveness.

Keywords: Battery management systems, Operations planning, Power system modeling, Spacecraft

1. Introduction

A company report forecasts growth in the number of planning small spacecraft projects, especially those whose mass is less than 50 kg. The popularity of small spacecraft comes from their high cost-effectiveness, which is achieved by the existence and widespread use of the CubeSat standard, its related technology, and equipment. However, to enhance this advantage, it is necessary to understand their characteristics, differences from conventional spacecraft, technology readiness status, and to propose suitable methods for efficient utilization. In this paper, we propose a method to enhance cost-effectiveness by introducing an operational design concept with a suitable numerical system model.

In operational design, one of the most important characteristics of a small spacecraft is strict on-board resource limitations that come from their mass and volume constraints. The conventional spacecraft operations concept and system have been designed based on sufficient or redundant resources. These constraints are not suitable for small spacecraft designs. The noteworthy points of conventional spacecraft systems, and their operations planning is the large operational and design margins for power generation and charging compared with the power consumption because of the high priority on battery protection. The battery operational constraints are defined by considering the relationships between the battery’s depth of charge, cycle number of charge–discharge, and battery lifetime. Even with small spacecraft applications, the conventional operation system considers enormous design and state estimation margins, which prevent efficient resource utilization. In a previous study, battery state estimation was performed through its terminal voltage measurement. The measured voltage is related to the battery state of charge (SOC); however, the quantitative relationships vary depending on the temperature or operation status, and the variation causes the source of the margin. Figure 1 shows the relationships between the simulated terminal voltage measurement and the SOC with different discharge current statuses utilizing the C-rate, which is defined as the discharge current divided by the nominally rated battery capacity. These differences cause conservative operations and prevent efficient operations under limited resource constraints. This paper agrees with the importance of battery protection in operations; however, it proposes performing a more accurate battery state estimation to define a suitable margin and design constraints for operations and implementation. Here, the mission duration rate, which is the mission operations rate for a certain operations period, is set as the efficiency index of the spacecraft, and the operations plan is designed to max-
imimize the index. Another difference with the conventional operational design appears in the project system constraints. The cost efficiency of small spacecraft enables us to construct distributed systems with a number of spacecraft and systems with a single spacecraft. For a conventional single-spacecraft system, operators manage spacecraft conditions by utilizing manually prepared operation procedures and the predicted status. However, distributed system complexities make their operations difficult with the same concept. Such human resource requirements are the main sources of cost growth. As for the current system, the complexity of the operations is suppressed by simplifying the spacecraft and its operations. This paper proposes a suitable system model for efficient and safe mission operations planning, and it utilizes the model-based development concept for operations plan design. The concept achieves a high affinity for automation and autonomy when combined with numerical optimization methods. Some studies have been conducted to achieve autonomous mission scheduling or on-board failure detection systems; however, they only focused on mission availability and continuous operations planning based on conventional methods. The detection system improvement, modeling, and higher accurate state prediction are not well considered. We propose utilizing the power system state model and its operational constraints for efficient and safe mission operations planning.

Another issue is the high failure rate of small spacecraft. The report shows that almost 50% of small spacecraft fail to achieve their mission; one of the main failure sources is power system management. The on-board autonomous error source detection, isolation, and recovery system contributed to the failure rate improvement. A previous study utilizes the terminal voltage measurement for the threshold of the fault detection system because the battery charging status relates to operational health, and performs mode transition to safer mode to recover the system. In this paper, we propose the utilization of a higher-accuracy battery state estimation as the fault detection threshold. The higher accuracy on-board state estimation enables us to define the precise threshold for fault detection, improve the failure source detection ability, and reduce the operating margin.

For battery state estimation, we propose applying the model-based battery state estimation method that utilizes the same battery numerical model as the mission operations planner. As mentioned above, the terminal voltage measurement has modeling errors based on the temperature or operation status. The Coulomb counting method is another method for estimating the battery state by the time-series current measurement integration. This method requires an initial state estimation, whose errors affect the state estimation accuracy as an offset. In the case of spacecraft applications, it is difficult to estimate the initial state with high accuracy or maintain the estimated values continuously, even under system contingencies. Model-based battery state estimation methods overcome the problems of the other methods, and they are popular in recent commercial battery management systems applications. Comparing to the other on-ground applications, the spacecraft application has a wider range of environmental temperature and power consumption variations, and limitations in maintenance. Further, the orbital motions vary the charge-discharge characteristics together with spacecraft attitude variations defined by the operational mode, affect the spacecraft system as the uncontrollable design constraints and limit the monitoring opportunities.

This paper proposes a model-based efficient and safe operations planning concept that focuses on battery management to enhance the cost-effectiveness of the small spacecraft. The remainder of this paper is organized as follows: Section 2 summarizes the modeling of the main components of a suitable model required to demonstrate the effectiveness of the proposed concept. It comprises a battery and a related spacecraft system, including the temperature-dependent characteristics and its parameter identification through experiments. Section 3 explains the design policy of the mission operations planning system called the mission operations planner. It designs the optimal operations mode transitions profile based on the mission requirement, through numerical optimization. The parameters, cost functions, and constraints are defined to implement the proposed concept. Section 4 demonstrates the effectiveness of the proposed operational concept through a case study. It presents the numerical and experimental case study results and additional techniques for implementation, assuming an imaginary Earth-observing small spacecraft in the low earth orbit. The functions to achieve the concept are validated one by one. Finally, Section 5 concludes the study.

2. Modeling

Modeling targets are limited to spacecraft power status emulation under various operational modes: power supply elements and its management system, including power consumption definitions, as shown in Fig. 2. The power supply elements are divided into the battery as the electrical power storage and solar cell array as the power generation; both have temperature-dependent characteristics.

2.1 Battery Modeling This study uses a lithium-ion rechargeable battery as an electrical power storage device because of its high specific energy and flight-proven evidence. In this study, SOC was set as the main parameter to express the battery status. The SOC is the ratio of the remaining capacity to the full charge capacity (FCC) that describes the
maximum capacity of the battery at a certain time. The depth of discharge (DOD), which has already been explained in the introduction, is the ratio of discharge to the FCC. The sum of the SOC and DOD is 1. The FCC deteriorates gradually over time and usage; however, this study considers it to be a constant value because of the short period of mission operations planning assumption.

2.1.1 Equivalent Circuit Model

Figure 3 shows the equivalent circuit model\(^{(20)}\). The equivalent circuit model can be expressed with the electrophoretic resistance \(R_0\), charge transfer resistance \(R_{ct}\), electrical double layer capacity \(C_{dl}\), and diffusion impedance \(Z_{do}\). The diffusion impedance is known as the Warburg impedance. Here, \(R_0\) is attributed to the electrophoretic process in the electrolyte, \(C_{dl}\) to the change exchange by the electrode reaction, and \(Z_{do}\) to the diffusion process of the electrode reaction. The diffusion impedance \(Z_{do}\) is approximated as three series of resistor-capacitor parallel circuits based on Foster’s reactance theorem, and they comprise \(R_i\) and \(C_i\) \((i = 1\)–\(3)\). The previous study shows 3–5 series circuits are required to approximate with acceptable accuracy the diffusion process \(^{(21)}\). Further, OCV denotes the open circuit voltage, \(i(t)\) denotes the charge–discharge current flow, and \(v(t)\) represents the terminal voltage measurement.

The state variable is defined as

\[
x_{\text{batt}}(t) = \begin{bmatrix} \text{SOC}(t) & v_{\text{dl}}(t) & v_1(t) & v_2(t) & v_3(t) \end{bmatrix}^T.
\]

The state space representation becomes

\[
\dot{x}_{\text{batt}}(t) = Ax_{\text{batt}}(t) + bu(t).
\]

\[
y(t) = f_{\text{OCV}}(\text{SOC}(t)) + c^T x_{\text{batt}}(t) + R_0 i(t).
\]

where

\[
A = \begin{bmatrix} 0 & -1/R_{ct} & -1/C_{dl} & -1/C_{dl} & -1/C_{dl} \\ 1/FCC & 1/C_{dl} & 1/C_3 & 1/C_2 & 1/C_1 \end{bmatrix}
\]

\[
b = \begin{bmatrix} 0 & 1 & 1 & 1 \end{bmatrix}^T
\]

\[
c = \begin{bmatrix} C_{dl}/2, R_i = 8R_{ct}/(2l - 1)^2\pi^2, l = 1, 2, 3 \end{bmatrix}
\]

Here, \(A\) denotes a diagonal matrix. The unknown parameters \(R_0, R_{ct}, C_{dl}\), and the SOC-OCV function \(f_{\text{OCV}}(\text{SOC}(t))\) are identified from experimental measurements.

2.1.2 Detailed Modeling and Parameter Identification

The unknown parameters \(R_0, R_{ct}, C_{dl}\) are identified using the alternating-current (AC) impedance method \(^{(28)}\) under temperature and OCV variations. Here, DC current also affects impedance measurement, especially \(R_{ct}\), however; we first made a model based on the temperature and SOC dependences. The results are sorted as functions of the measurement conditions. The SOC-OCV function \(f_{\text{OCV}}(\text{SOC}(t))\) is derived experimentally using the same approach as in Ref. (23).

The AC impedance method measures responses from the weak AC signal of a wide range of frequency conditions and identifies unknown parameters. The impedance measurements are expressed in a Nyquist plot as shown in Fig. 4. In the middle-frequency region, the trajectories draw a semicircle that crosses the real axis at \(R_0\) in the low-frequency region and \(R_0 + R_{ct}\) in the high-frequency region. \(C_{dl}\) can be derived from the frequency \(f_{\text{max}}\), which has the peak of the semicircle with the relationship defined from the characteristics of the resistor–capacitor parallel circuits \(R_{ct}C_{dl} = 1/(2\pi f_{\text{max}})\).

Measurements were performed using a chemical impedance analyzer (IM3590, Hioki E.E. Corporation). The measurement conditions are listed in Table 1. A total of 81 measurements were performed to cover possible spacecraft conditions with a product of nine OCV and nine temperatures. The OCV becomes 4.17 V under the fully charged state. The measurements were performed for a single battery cell with the holder in the temperature chamber after the battery surface temperature became equal to the controlled environmental temperature.

The measurement results are summarized in Figs. 5, 6, and 7 for each parameter. Most measurements show temperature dependence; however, little OCV dependence was observed.
This study modeled the temperature dependence of each parameter. The modeling results are represented in Figs. 5, 6, and 7 by black lines. The modeling range is defined from −10 to 20 °C because of high resistance $R_0$ results under −20 °C. The high resistance causes a large voltage drop in the system, and the battery cannot be charged safely under −20 °C. $R_0$ is modeled as constant value because it shows a small temperature dependence. $R_{ct}$ and $C_{dl}$ are modeled as functions of the battery temperature $T_b$, respectively. The approximation function is derived from the measurement data and the unknown parameters are fitted using the least-squares method. $R_{ct}$ is modeled as

$$R_{ct} = \exp(-(a_{Rct}T_b + b_{Rct})) \quad \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots \cdots 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strongly related to the power status; the power management components control the battery charge and discharge, the output power of the solar cell array, and the system component power status that depends on the operational mode definitions. We consider only the steady-state power consumption for each component at each operational mode. A previous study showed that an important mission-related mode tends to require larger power consumption. In addition, the study uses battery status as the fault detection threshold, and performs mode transition to the safer and low power consumption mode after the fault detection [17].

In addition to the power supplement control of the components, the power management component controls the battery charging status based on in situ measurements to achieve effective charging. The battery is charged using a constant-current and constant-voltage (CCCV) method. This method uses CC charging under low terminal voltage conditions to shorten the charging duration and moves onto CV charging when the terminal voltage becomes high to prevent overcharging; then, it finishes charging if the charging current becomes low. Overcharging causes failures or lifetime shortages because of the lithium-ion battery characteristics. Under these constraints, the battery current status $u(t)$ can be expressed as

$$u(t) = \begin{cases} u_i & (i: \text{operational mode number}) \\ u_{CC} & (u_i \geq u_{CC}) \\ u_{CV} & (y(t) \geq V_{max}) \\ 0 & (u_i > 0 \land SOC(t) \geq SOC_{max}) \end{cases} \quad (13)$$

where $u_i$ denotes the current status of the battery defined by the operational mode numbered by $i$ and battery pack design. Here, $u_i$ is the function of the power consumption of the mode $P_i$, power conversion efficiency $\eta$, and spacecraft bus voltage $V_b$. Further, $u_{CC}$ denotes the current status of CC charging, $V_{max}$ denotes the control voltage threshold, and $u_{CV}$ represents the current status of CV charging. The function $u_i$ is expressed as

$$u_i = (\eta P_0 - P_i)/V_b \quad \quad (14)$$

which is defined for each operation mode, and the mode number increment results in system complexity.

3. Design Policy of Mission Operations Planner

Based on the discussions in the previous sections, a spacecraft power status model whose characteristics vary depending on the operational mode is designed. The method used to design a mission operation planner is explained, and an efficient operational mode sequence is defined by utilizing a numerical optimization method that introduces the operational variables related to the importance of the operational mode based on the mission requirement. The entire operational duration period for mission operations planning is divided into $l$ sections, and the planner defines the suitable operational modes for each section. Here, only two operational modes are assumed for a straightforward discussion: 1) charging mode ($i=1$) and 2) mission mode ($i=2$). The operational variables in the divided section $m$ can be expressed as $X_m \in [0, 1]$ for $m \in N = [1, \ldots, l]$. The variables can be defined based on the importance of the modes. The charging mode variable is defined as zero ($X_m = 0$ with $i=1$), and the mission mode is defined as 1 ($X_m = 1$ with $i=2$).

The class of the optimization variables can be expressed as $X_{mode} = [X_1, X_2, \ldots, X_l]^T$. In this application, the requirement can be defined as the maximization of the spacecraft mission duration, and therefore, the cost function $J$ is defined to maximize the spacecraft mission duration as

$$\text{maximize } J = \sum_{m=1}^{l} X_m \quad \quad (15)$$

Constraints for safe operations are designed in the acceptable SOC range as

$$SOC_{min} \leq SOC(t_{in}) \leq SOC_{max} \quad \quad (16)$$

$$X_m \in [0, 1] \quad \forall m \in N = [1, \ldots, l] \quad \quad (17)$$

where $SOC_{min}$ and $SOC_{max}$ are defined as arbitrary values between 0 and 100% to satisfy the mission or project objectives considering the charge–discharge cycle number or lifetime of the battery. $SOC(t_{in})$ is expressed as

$$SOC(t_{in}) = \frac{100}{FCC} \int_{t_{in}}^{t_{fin}} u(t) dt + SOC(t_{in}) \quad \quad (18)$$

where $u(t)$ is defined by Eq. 13, and $t_{in}$ and $t_{fin}$ denotes the beginning and finishing time of section $m$, respectively.

The defined equations have difficulties in obtaining realistic implementation solutions without modification. Here, we perform the continuous relaxation of the optimization variables and obtain effective information to solve the original problems. The current status $u_i$ of the battery is a discretized parameter that depends on the operational mode $i$. The relaxation of $u(t)$ is performed as the product of the current status of the battery of each mode and the charging condition flags defined as a sigmoid function.

The constraints with the relaxation become

$$SOC_{min} \leq SOC(t_{in}) \leq SOC_{max} \quad \quad (19)$$

$$0 \leq X_m \leq 1 \quad \forall m \in N = [1, \ldots, l] \quad \quad (20)$$

where $SOC(t_{in})$ can be expressed with flag functions $h_1$ of the constant-current charging, and $h_2$ of the constant-voltage charging as

$$SOC(t_{in}) = \frac{100}{FCC} \int_{t_{in}}^{t_{fin}} u(t) dt + SOC(t_{in}) \quad \quad (21)$$

$$u(t) = (X_m u_2 + (1-X_m)u_1)(1-h_1)(1-h_2) + h_1 u_{CC} + h_2 u_{CV} \quad \quad (22)$$

$$h_1 = \frac{1}{1 + \exp(-(d_1(X_m u_2 + (1-X_m)u_1 - u_{CC}) + b_1))} \quad \quad (23)$$

$$h_2 = \frac{1}{1 + \exp(-(d_2(y(t) - V_{max}) + b_2))} \quad \quad (24)$$

The relaxation problem results show an arbitrary value between 0 and 1 for each section. The results can be converted to operational mode variables — 0 or 1 — using an arbitrary threshold because there are only two operational modes in this case. The integerization threshold was defined to maximize the cost function. For example, the trial-and-error approach is a straightforward solution. These approaches enable automatic mission operations planning. Figure 10 shows...
the example of the integerization. The relaxation problem results are integerized by the integerization threshold.

The proposed method is used for nominal mission design that does not contain unexpected on-orbit events. Real operations may differ from the designed state because of modeling errors or system failures; these may exceed the operational constraints. In this case, we propose changing the integerization threshold to achieve a profile similar to that of the designed scenarios. The spacecraft memorizes the relaxation problem results and temporarily changes the integerization threshold for the operational mode variable in real time to achieve a suitable profile if the differences between the designed SOCs and in situ SOC estimation exceed the predefined acceptable value. The integerization threshold takes the initial value after the difference converges to a value that is smaller than that is acceptable. By introducing these constraints, the problem can be solved autonomously.

4. Case Study: Mission Operations Planning for Earth-Observing Small Spacecraft

The proposed concept is verified using numerical and experimental case studies assuming an Earth-observing small spacecraft on a low Earth orbit.

4.1 Case Study Condition

The assumed spacecraft has two operational modes: 1) charging and 2) observing mission. The spacecraft has two solar cell array wings and no other solar cells on the spacecraft surface; the Earth observation equipment is on the opposite surface of the solar cell arrays. The charging mode has a relatively small power consumption only with fundamental bus system operations, and the spacecraft attitude is controlled so that the solar cell array always faces the sun. The observing mission mode consumes additional power than the charging mode because it has to operate the mission equipment; the attitude is controlled so that the opposite side of the solar cell array points to the Earth’s center continuously. The spacecraft generates power only with solar cell arrays. Therefore, the power generation varies depending on the orbital position varies on time, because it causes solar incoming angle change to the solar cell array. In addition, the designed operational mode status which varies based on the designed results also affects the solar power generation because of its attitude change. The spacecraft is in a circular orbit, and only the planar motion is considered. It contains an eclipse phase and has to charge power for the eclipse during the sunshine phase as shown in Fig. 11. Its altitude is defined as 300 km. Sunlight is considered to be parallel, and the sun diameter effects are not considered. The entire mission operations design duration is defined as the multiplication of the orbital number $T_o$ [rev] and its divided section $l$ [1/rev]. The operations plan design is performed by optimization considering the periodic orbital motion. The simulation constraints are summarized in Fig. 11 and the main simulation parameters are summarized in Table 4. Here, the entire simulation duration is set to 0.5 days. The controller is expected to maximize the mission mode duration for efficient operations while satisfying the SOC constraints for achieving safe operations.

4.2 Numerical Study

The effectiveness of the proposed concept is verified via numerical studies. The proposed concept has two aspects: 1) improvement of state estimation accuracy and 2) mission operations plan optimization with the mission operations planner based on the spacecraft system model. The on-board SOC estimation can be achieved by using the developed model, measured temperature, measured voltage, and measured current. To evaluate the proposed method, the terminal voltage measurement method was set as the conventional state estimation method providing reference.

4.2.1 Effects of State Estimation Accuracy

The effectiveness of the on-board state estimation accuracy improvement is discussed via a numerical study by utilizing the state estimation value as mode transition thresholds related to the battery state constraints. Here, the “threshold” differs from the integerization threshold for optimization and it is derived by the design constraints; SOC$_{min}$ and SOC$_{max}$. Verifications were performed through mode transition simulations with two different threshold sets: 1) threshold based on the model-based battery charging state estimation, and 2) threshold based on the conventional terminal voltage measurement state estimation. The numerical simulations start with the mission mode, and it changes the operation mode when it exceeds the predefined range and threshold. The charging mode transition occurs with a minimum threshold and the mission mode transition occurs with a maximum threshold. The threshold is defined for a single cell of a well-balanced assembled battery. For model-based battery charging estimation, the threshold is defined as acceptable SOC, with a minimum value of 73% and a maximum value of 98%. The terminal voltage threshold is defined as $V_{min} = 3.697$ V and $V_{max} = 4.132$ V. Here, $V_{min}$ is defined considering the OCV value of the SOC 73% together with the voltage drops of the maximum discharge current condition, and $V_{max}$ is defined as an OCV value of the SOC of 98%. Simulations were performed under two different temperature conditions: 1) Constant temperature environment $T_S = 293$ K and $T_{SC} = 330$ K; and 2)
realistic temperature environments by numerical simulation considering the orbital motion.

Figure 12 shows that the mode transition results under the constant-temperature environment. It includes the time-series histories of the selected operations modes, SOC, battery terminal voltage, and the current status of the battery. In the SOC results, the minimum and maximum SOC constraints are indicated by the red dashed lines. The battery is in charging status when the current status of the battery is positive; it is in the discharging status when the current status is negative. The mission mode histories show the advantages of the threshold with conventional terminal voltage measurement in terms of mission duration rate. The terminal voltage measurement shows variations depending on the load condition even if it has the same SOC condition. The terminal voltage measurement increment makes mission durations longer. The mission duration rate with the terminal voltage measurement threshold was 36.2%, which is higher than that with the proposed SOC estimation threshold 23.7%. This difference is attributed to the mode transition status difference between the charging and mission modes, and the battery charging method variations depending on the battery status. The threshold using the proposed SOC estimation shows advantages in safe operation. The maximum SOC value is 91.8% at the mode transition with the conventional terminal voltage measurement, except for the initial phase. This is the result of the terminal voltage measurement increment caused by the charging overvoltage.

In this simulation, the terminal voltage measurement increment makes mission durations longer; however, it may make mission durations shorter depending on the spacecraft conditions, the threshold design policy, or constraints. The maximum SOC value was 97.8% during the mode transition with the proposed SOC estimation. During the high-voltage range, the battery is charged with the CV method, which takes a longer time than the CC method; in this case, the charging current becomes smaller if the SOC becomes larger. This aspect can be improved by adjusting the mode transition threshold. The minimum SOC results show that the conventional terminal voltage measurement threshold periodically exceeds the lower limit constraints; the minimum value was 69.3%. The proposed SOC estimation threshold shows SOC exceedance, and its minimum value was 72.3%. This is caused by the position of the spacecraft during mode transitions. The mode transition to charging occurs around the beginning of the eclipse, and the spacecraft cannot start charging during the eclipse. With the proposed SOC threshold, the maximum SOC is higher than that with the conventional terminal voltage threshold, and it has the advantage of satisfying the minimum SOC limitations of the proposed threshold.

Figure 13 shows the mode transition results under realistic temperature environments; it includes the time series histories of the selected operation mode, SOC, battery terminal voltage, solar array temperature, battery temperature, and current status of the battery. In the SOC results, the minimum and maximum SOC constraints are indicated by the red dashed lines. The spacecraft is divided into three components: the body and each solar cell array; the balanced temperature is calculated for each component for each time step. The battery temperature was assumed to be the same as the spacecraft’s body temperature. The results show the same tendencies as the constant-temperature simulation; however, the quantitative results show the differences. The results are quantitatively summarized in Table 5. Here, $SOC_{\text{min}}$ shows the minimum SOC results for the whole simulation durations and the value has to be within the required limitation, in other
Table 5. Simulation results summary, state estimation accuracy improvement

| Temperature condition | Threshold | SOC_{\text{min}} [%] | Under SOC_{\text{min}} rate [%] | Mission duration rate [%] |
|-----------------------|-----------|----------------------|---------------------------------|--------------------------|
| Constant              | SOC       | 72.3                 | 0.6                             | 24.7                     |
|                       | Terminal voltage | 69.3       | 12.9                           | 32.2                     |
| Variable              | SOC       | 71.3                 | 3.6                             | 24.7                     |
|                       | Terminal voltage | 70.7       | 8.6                             | 38.5                     |

words, larger than the lower limit: SOC_{\text{min}} = 73%. “Under SOC_{\text{min}} rate” shows the durations whose SOC shows smaller than the SOC_{\text{min}}. It shows in the percentages, by dividing the duration into the whole simulation time. This value has to be zero for safe operations. Mission duration rate is the mission duration time in percentage for the whole simulation time defined as J/(T_n l). Here, cost function J is the spacecraft mission duration describing as Eq. 15, and whole mission duration time is defined as the multiplication of the orbital number T_n and its divided section l. The value has to be maximized for efficient operations.

The mode transitions from safe mode to mission mode have occurred end of the sunshine phase for both thresholds and conditions. In the SOC threshold cases, the mode transitions under the realistic temperature environment occur earlier than those under the constant temperature environment and it causes a larger mission duration rate and “Under SOC_{\text{min}} rate” because of the eclipse effects. In the conventional terminal voltage threshold case, the mode transitions from safe mode to mission mode have occurred at almost the same timing for both environments. The mode transitions from mission mode to the safe mode under the realistic environment have occurred a bit later than those under the constant temperature condition and they are during the eclipse phase. It causes a larger mission duration rate and smaller “Under SOC_{\text{min}} rate” together with the power generation difference under the realistic temperature condition. Here, the thresholds are almost equal to the limitations, and the mode transition only occurred when the measurements exceed the thresholds. The simulation results indicate that only estimation accuracy improvement is insufficient to achieve mission duration maximization or safe operations. The efficient mode transition policy is important to improve these aspects. It is important to design the mode transition considering spacecraft orbital position and the future state estimation.

4.2.2 Effects of Mission Operations Planner

This study solves defined problems in Section 3 using the sequential quadratic programming (SQP) method \(^{(27)}\) for each period. The optimization results of the initial period are set as the initial values of the later period because the spacecraft operational status changes periodically in this problem. The integerization threshold of the SQP result was determined based on rule-based trial-and-error. The simulation conditions and constraints are the same as those described in the previous section. Two types of mode transition thresholds related to the battery state constraints were designed for the mission operations planners: 1) the proposed SOC-based threshold, and 2) conventional terminal voltage measurement threshold, and the definition is the same as the previous section.

Figure 14 shows the mode sequences design results with the mission operations planners; the contents are the same as those in Fig. 12. The mission mode histories are the results after the integerization and they show a higher mission duration rate with the proposed SOC threshold and its value is 51.4%. This value shows the effectiveness of the proposed planner because it is improved compared to the value presented previously. The mission duration rate was 36.2% with the conventional terminal voltage threshold and mission operations planner, and it shows no improvement from mission design optimization. For the proposed threshold, the mode transition is performed at the optimum timing to maximize the mission duration by utilizing an improved model with accuracy. For example, there are some durations whose power balance becomes positive depending on the orbital position even if the spacecraft is in mission mode. With the proposed SOC threshold and mission operations planner, the spacecraft’s mode becomes the mission mode under those conditions. Further, the spacecraft transitions to the charging mode before the eclipse to enlarge the mission opportunities during the eclipse.

Further, the SOC results show the advantages of the proposed threshold in terms of constraint satisfaction. With the conventional threshold, the SOC periodically exceeds the limit constraints; the minimum value was 69.3%. With the proposed threshold, the minimum SOC limitation was satisfied, and its minimum value was 73.3%. Using a conventional threshold, the mode transition to the charging occurred during the eclipse, and the spacecraft could not charge during the eclipse; however, with the proposed threshold, the spacecraft achieves mode transition at suitable conditions to achieve mission duration maximization and constraint satisfaction. With both mode transition thresholds, the spacecraft satisfies the upper constraints, and the mode transition occurs at approximately 91.8% for the conventional threshold and approximately 85% for the proposed threshold. The mode transition result with the proposed threshold shows the effects of the optimization with the CV charging avoidance. The CV
Table 6. Simulation results summary, design results with the mission operations planners

| Threshold | SOCmin [%] results | Under SOCmin rate [%] | Mission duration rate [%] |
|-----------|--------------------|-----------------------|--------------------------|
| SOC       | 33.3               | 0                     | 51.4                     |
| Terminal voltage | 69.3           | 12.9                  | 36.2                     |

charging mode tends to increases the charging duration and it has to be avoided to increase the mission duration rate. The results are summarized in Table 6.

4.3 Experimental Verification

Experimental verification is performed with a single battery cell, PC, battery charge–discharge power supply, and monitoring system, as shown in Fig. 15. The model-based numerical simulation with the mission planner designs the operational sequence as shown in the previous section. The system model is adjusted for a single battery cell by considering additional components such as measurement system effects to emulate the experimental setup. The experiment PC installs the three software; 1) the model-based spacecraft battery state simulator which can calculate the current condition of the battery based on the operational sequence; 2) real-time battery state estimator using battery model and real-time temperature, voltage, and current monitoring results; and 3) operational mode adjustment system using the designed and estimated SOC values. The latter two software are assumed to be the same functions as the real spacecraft on-board software. The battery charge–discharge state is controlled by the charge–discharge power supply (ECD35-3: Matsusada precision). The PC sends commands to the power supply based on the installed system and obtained measurement. The monitoring system uses a current sensor (HPS-5-AP: U–RD) and data logger measurements (USB-6002: National Instruments and the midi LOGGER GL840: Graphtec) and these data are utilized in the state estimator. Figure 16 shows the flowchart of the experimental sequence.

4.3.1 Performance Verification of Mission Operations Planner

This section describes the experiment based on the designed operational scenarios with the proposed mode transitions threshold generated in Section 4.2.2. The environmental temperature or initial conditions were set to achieve the designed condition.

Figure 17 shows the experimental results with the designed operations profile. The mode transition profile was designed using a mission operations planner with the proposed threshold. For comparison, the results are shown together with the designed operational scenario in the previous section; the plotted contents are the same as those in Fig. 12 or Fig. 14. The time series SOC difference between the design and the experiment is summarized in Fig. 18. The experimental profile follows the designed profile well and satisfies SOC constraints. This result shows that the battery-related system model emulates the real system with acceptable accuracy. However, the voltage and current profiles show differences around the mode transition. Further, small errors were also shown continuously during the experiment. One of the reasons for the error is the modeling method. This study constructs a battery model based on a static state experiment, and the transient is not modeled well. This aspect results in the differences between real and mission operations planning. The modeling has room to improve if more frequent mode transition is required by the project.

4.3.2 Performance Verification of On-board Threshold Adjustment

The on-board integerization threshold adjustment technique is equipped in the on-board software.
to follow the designed scenario even if the modeling error existence. The on-board software has the mission operations planner’s results consisting of SQP results, integerization threshold, integerized operational mode profile, and designed SQP histories. The integerization mode threshold is changed if the differences between the designed and estimated SOC become larger than the predefined value. We set the predefined value to 10 points in absolute. The integerization threshold is changed to 1.1 when the difference becomes positive, and the spacecraft transitions to the mission mode. The integerization threshold becomes 0 when the difference becomes negative, and the spacecraft transitions to the charging mode. The integerization threshold becomes a nominally defined value if the difference converges to less than 10. The verification experiment was performed at room temperature, and we used the mission scenario designed with the mission operations planner for a 50 °C environment. A large modeling error is caused by the temperature difference.

Figure 19 shows the time series histories of the mission mode, SOC, and on-board integerization threshold for the mode definition. The results of this experiment indicate that the designed scenario can be achieved without extreme over/undershoot when using this technique.

The mode history figure shows that the SOC differences between the designed and in situ measurements increased gradually because of the modeling errors caused by the temperature difference. The designed profile could not be achieved at room temperature; the on-board integerization threshold adjustment occurred three times within the entire mission duration. First, the adjustment occurred to satisfy the safety constraints. Then, adjustment was performed to maximize the mission duration. Finally, an adjustment made to satisfy the safety constraints was performed again. It can be seen in the threshold variations and related mode transitions in Fig. 19. The differences between the designed scenario and the actual status become smaller during the threshold adjustment is enabled and it is disabled when it converged close to the designed value; the difference remains within the designed acceptable range.

5. Conclusion

This paper proposes the concept of a mission operations planner for spacecraft operations to achieve small spacecraft utilization enlargement by enhancing its cost-effectiveness and improving the mission failure rate. This paper has two main points: 1) spacecraft mission operations planner (concept, design parameters, and optimization for mission planning), and 2) utilization of onboard state estimator for safe and efficient operations. Spacecraft system modeling was performed with suitable accuracy for realizing the proposed planner development. The core part of the model is the battery state defined as the most important parameter in spacecraft operations because it has a direct relationship with the health status of the spacecraft. Power-related spacecraft component modeling methods considering their temperature dependence were introduced with experimental-based parameter identification. The developed model achieved high-accuracy in situ state estimation and on-ground future state estimations.

Further, we introduced the design and implementation method of the proposed mission operations planner that utilizes the developed system model and numerical optimization. Continuous relaxation was introduced to design ideal mission mode profiles. The model-based optimization maximizes the mission duration. The high accuracy in situ state estimation reduces the operating margins for battery protection, improves the efficiency of each spacecraft, and achieves safe spacecraft operations. Further, we designed an implementation technique for dealing with the modeling error. The spacecraft system monitors the state differences between the in situ measurement and the designed value, and it performs mode transitions by adjusting an on-board parameter to satisfy the designed status tracking. The utility of the proposed system is verified via numerical discussions and experimental simulations which assume the application to the virtual Earth-observation spacecraft case study. The case study results show that the spacecraft system achieves a design mission scenario with acceptable accuracy. The effectiveness of the adjustment system was confirmed through experimental demonstration with a modeling error. Through the discussions, this paper shows the effectiveness of the proposed mission operations planner concept and the application possibility for real spacecraft operations, even though there is room for improvements to real spacecraft implementation.

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