

AUTONOMOUS RPOD FOR ARBITRARILY CONFIGURED SPACECRAFT WITH ANOMALY DETECTION

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Autonomous guidance, navigation, and control (GN&C) is essential for a sustainable deep-space logistics architecture. The challenges of establishing robust autonomy are numerous, and include state and parameter uncertainties, anomaly detection, and fault recovery. This work builds on previous research of the authors [1,2] which investigated autonomous GN&C for arbitrary thruster configurations and mass properties by expanding it to include state uncertainty and anomaly detection. Logistics vehicles with off-center-of-mass thruster configurations, operating under large but realistic state uncertainties, are simulated in rendezvous, proximity operations, and docking scenarios. Furthermore, a stuck thruster scenario is simulated, demonstrating the vehicle's ability to identify and overcome thruster anomalies without human intervention. The simulations show that, despite these realistic ambiguities and anomalies, the vehicle can converge to the desired pose to either successfully dock or autonomously abort and retreat.

INTRODUCTION

A permanent human lunar settlement, as a goal of the Artemis program, will require extensive logistics support during the early years, and presumably throughout its sustained phase of operation. In addition to the Moon2Mars architecture that includes logistics support all to Mars and beyond, these challenges further demand the creation of robust autonomy in spacecraft. These spacecraft must perform reliably in the absence of direct human control, lack of availability of ground stations communications, or in the presence of rapidly or frequently changing configurations. This suggests that a notional navigation and control scheme must be robust to a wide range of scenarios and configurations and to a certain degree of off-nominal events such as stuck thrusters (i.e., thrusters that remain on despite being commanded to stop firing) and unmodeled dynamics. There is an expectation that autonomy will be capable of handling such scenarios during long voyages between planets where communication is inhibited by long light-time delays and during rendezvous, proximity operations, and docking around distant celestial bodies. While previous work has demonstrated the conceptual ability of a spacecraft to generate thruster firing histories for both arbitrary thruster configurations [1], and arbitrary mass properties [2], these simulations have not been performed in the presence of noisy measurements or for off-nominal scenarios.

In pursuit of this level of autonomy, much work has been done to examine optimal path-planning and autonomous guidance for entry, descent, and landing (EDL) applications [3]. Furthermore, research has explored the behavior of thrusters and how control schemes might overcome or plan for true thruster behavior as compared to the modeled thruster behavior [4]. Similarly, this work depends on a thorough understanding of the spacecraft configuration, even if the configuration is arbitrary. However, it allows for state uncertainty (in the presence of both process and measurement noise), control uncertainty, and off-nominal thruster behavior. This demonstrates tolerance to changing parameters, so long as those changes are known.

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The use of Lie group $SE(3)$ enables guidance, navigation, and control algorithms within this framework enables the consideration of coupled motion even in the presence of noise without encountering singularities, unwinding, or nonuniqueness issues present in other attitude parameterization sets [5–7]. There has been some recent work focusing on constrained control and distributed control on $SE(3)$ and its tangent bundle $TSE(3)$, but these algorithms do not consider autonomous thruster firing nor autonomous thruster firing histories or constraints generated by duty cycles, such as is the case in reaction control system (RCS) thrusters [7–10]. Furthermore, it has been demonstrated in [7] that the use of this controller in the presence of noise when interpreted through the use of an unscented Kalman filter (UKF) first presented in [11, 12] is robust to measurement and process uncertainty. This same controller and filter have also been used to demonstrate the stability of launch systems in the presence of large uncertainties [9, 13].

In this paper, the algorithm described demonstrates that a spacecraft can autonomously generate a time-optimal thruster firing history in the presence of noisy measurements and off-nominal or constrained thruster configurations, provided that the available thrusters have some access to all degrees of freedom (DoF). A case in which suboptimal thruster configurations restrict motion is considered, both for RCS thrusters displaced from the center of mass and when not all thrusters are available. The spacecraft configuration is defined within the $SE(3)$ framework, which allows the consideration of both the translational and rotational components of its thruster configuration simultaneously and aids in generating precise guidance protocols even in the presence of off-nominal or constrained thruster configurations. Finally, the off-nominal scenario of a stuck thruster is also simulated where the spacecraft demonstrates the ability to identify the anomaly and perform an abort maneuver without human intervention.

PRELIMINARIES

Formulation on Special Euclidean Group $SE(3)$

The pose of a rigid body is expressed in $SE(3)$ as

$$g = \begin{bmatrix} R & r \\ 0_{1 \times 3} & 1 \end{bmatrix} \in SE(3) \quad (1)$$

where $R \in SO(3)$ is the rotation matrix from the body frame $\mathcal{B}$ to the inertial frame $\mathcal{N}$ and r is the position vector in the inertial frame. The rigid body's velocity can be defined using an augmented velocity vector $\mathbb{V}$ as

$$\mathbb{V} = [\omega^T, v^T]^T \in \mathbb{R}^6 \quad (2)$$

such that $\omega \in \mathbb{R}^3$ is the angular velocity in the body frame $\mathcal{B}$ and $v \in \mathbb{R}^3$ is the translational velocity relative to the inertial frame and expressed in the body frame. The dynamic equations for a rigid body with can be expressed in $TSE(3)$ as

$$\dot{g} = g\mathbb{V}^\vee \quad (3a)$$

$$\dot{\mathbb{V}} = \mathbb{I}^{-1} (\text{ad}_{\mathbb{V}}^* \mathbb{V} + \tau + u) \quad (3b)$$

For the purpose of developing a formulation for relative motion or tracking controller on $SE(3)$, the following expression is introduced:

$$\begin{aligned} \epsilon_g &= g_d^{-1}g = \begin{bmatrix} R_d & \epsilon_r \\ 0 & 1 \end{bmatrix} = \begin{bmatrix} R_d^T R & R_d^T (r - r_d) \\ 0 & 1 \end{bmatrix} \in SE(3) \\ \epsilon_{\mathbb{V}} &= \mathbb{V} - \text{Ad}_{\epsilon_g} \mathbb{V}_d \in \mathbb{R}^6 \end{aligned} \quad (4)$$

where the subscript ‘ d ’ denotes the desired or modeled state or parameter. The relative dynamics of a rigid body with a changing center of mass and moments of inertia in $SE(3)$ are represented as

$$\dot{\epsilon}_g = \epsilon_g \epsilon_{\mathbb{V}}^\vee \in \mathfrak{se}(3) \quad (5a)$$

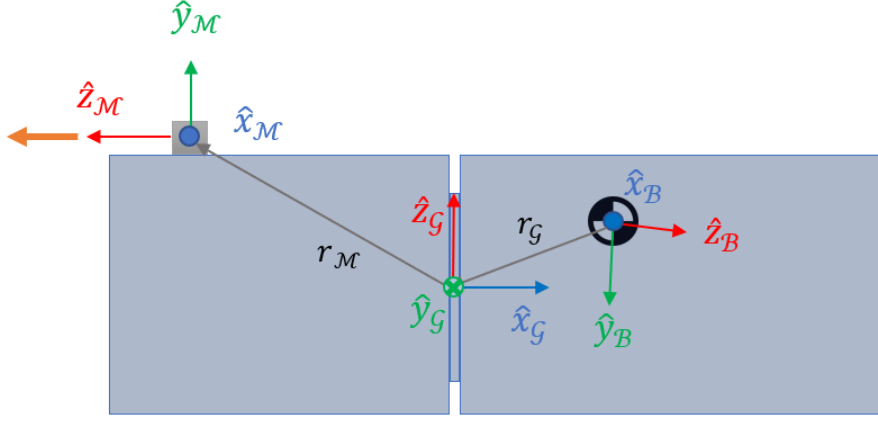


Figure 1: An illustration of the terminology associated with various frames within the spacecraft configuration.

$$\begin{aligned}\dot{\epsilon}_{\mathbb{V}} &= \frac{d}{dt} \left(\hat{\mathbb{V}} - \text{Ad}_{\epsilon_g} \mathbb{V}_d \right) \\ &= \dot{\hat{\mathbb{V}}} - \text{ad}_{\epsilon_{\mathbb{V}}} \text{Ad}_{\epsilon_g} \mathbb{V}_d - \text{Ad}_{\epsilon_g^{-1}} \dot{\mathbb{V}}_d\end{aligned}\quad (5b)$$

where

$$\mathbb{I} = \begin{bmatrix} J & 0_{3 \times 3} \\ 0_{3 \times 3} & mI_3 \end{bmatrix} \in \mathbb{R}^{6 \times 6}, \quad (6)$$

and the coadjoint operator is defined as

$$\text{ad}_{\mathbb{V}}^* = \text{ad}_{\mathbb{V}}^T = \begin{bmatrix} -\omega^\times & -v^\times \\ 0_{3 \times 3} & -\omega^\times \end{bmatrix} \in \mathbb{R}^{6 \times 6} \quad (7)$$

In Eq. (5), $\tau \in \mathbb{R}^6$ represents the impressed dynamics acting on the system expressed in the $\mathcal{B}$ frame. Furthermore, the augmented vector of control inputs is given by $u = [L^T, f^T]^T \in \mathbb{R}^6$ where $f \in \mathbb{R}^3$ and $L \in \mathbb{R}^3$ represent the control forces and torques, respectively.

Consider an actuator $g_{\mathcal{M}} \in \text{SE}(3)$ where $\mathcal{M}$ denotes the frame of a motor or actuator with respect to the rigid body. This frame has a pose defined as

$$g_{\mathcal{M}} = \begin{bmatrix} R_{\mathcal{B}/\mathcal{M}} & r_{\mathcal{M}} \\ 0_{1 \times 3} & 1 \end{bmatrix} \in \text{SE}(3) \quad (8)$$

where $R_{\mathcal{B}/\mathcal{M}}$ is the attitude of the thruster in body frame $\mathcal{B}$, and $r_{\mathcal{M}}$ is the position of the body with respect to the thruster. The relative motion dynamics expressed above may be extended to describe the pose of a thruster relative to a central body in either a geometric frame $\mathcal{G}$, which is advantageous for unknown or arbitrary mass properties, or relative to a standard body frame $\mathcal{B}$. This may be expressed as $g_{\mathcal{G}/\mathcal{M}}$ or $g_{\mathcal{B}/\mathcal{M}}$, respectively, as illustrated in Figure 1. Take the linear adjoint operator of this relative configuration to yield

$$\text{Ad}_{g_{\mathcal{M}}} = \begin{bmatrix} R_{\mathcal{B}/\mathcal{M}} & 0_{3 \times 3} \\ r_{\mathcal{M}}^\times R_{\mathcal{B}/\mathcal{M}} & R_{\mathcal{B}/\mathcal{M}} \end{bmatrix} \in \mathbb{R}^{6 \times 6} \quad (9)$$

The transpose of the adjoint operator is defined as

$$\text{Ad}_{g_{\mathcal{M}}}^T = \begin{bmatrix} R_{\mathcal{B}/\mathcal{M}}^T & -R_{\mathcal{B}/\mathcal{M}}^T r_{\mathcal{M}}^\times \\ 0_{3 \times 3} & R_{\mathcal{B}/\mathcal{M}}^T \end{bmatrix} \in \mathbb{R}^{6 \times 6} \quad (10)$$

And finally,

$$\text{Ad}_{g_{\mathcal{M}}}^T = \begin{bmatrix} R_{\mathcal{B}/\mathcal{M}} & R_{\mathcal{B}/\mathcal{M}} \left(R_{\mathcal{B}/\mathcal{M}}^T r_{\mathcal{M}} \right)^{\times} \\ 0_{3 \times 3} & R_{\mathcal{B}/\mathcal{M}} \end{bmatrix} \in \mathbb{R}^{6 \times 6} \quad (11)$$

TSE(3)-Based Unscented Kalman Filter

In this work, the UKF on SE(3) first introduced by [11] and extended to tracking control by [7] is used in this work. This filter works by first taking in initial state estimates $\hat{g}_{k|k} \in \text{SE}(3)$ and $\hat{\mathbb{V}}_{k|k}$ and covariance $P_{k|k} \in \mathbb{R}^{p \times p}$ where p is the number of state variables. An initial guess is propagated to generate an *a priori* state estimate using some nonlinear dynamical function $f(g, \mathbb{V}, \eta_p)$, where $\eta_p \in \mathbb{R}^p$ is the process noise, in order to generate $\hat{g}_{k+1|k}$ and $\hat{\mathbb{V}}_{k+1|k}$. The UKF uses sigma points serve as small perturbations to $\hat{g}_{k|k}$ and $\hat{\mathbb{V}}_{k|k}$ in order to obtain an updated covariance $P_{k+1|k}$. The sigma points are expressed through the matrix $\chi_p \in \mathbb{R}^{p \times (2p+1)}$ of $2p+1$ sigma-point column vectors $\chi_{p,i}$ and the matrix $\chi_q \in \mathbb{R}^{q \times (2q+1)}$ of $2q+1$ sigma column vectors $\chi_{q,i}$. Additionally, the weight matrices are defined as

$$\begin{aligned} W_p^{(m)} &= [W_{p,0}^{(m)}, W_{p,1}^{(m)}, \dots, W_{p,4p}^{(m)}] \\ W_p^{(c)} &= [W_{p,0}^{(c)}, W_{p,1}^{(c)}, \dots, W_{p,4p}^{(c)}] \end{aligned} \quad (12)$$

where

$$\begin{aligned} W_{p,0}^{(m)} &= \frac{\lambda_p}{\lambda_p + p}, \quad W_{p,0}^{(c)} = \frac{\lambda_p}{\lambda_p + p} + (1 - \alpha^2 + \beta) \\ W_{p,i}^{(m)} &= W_{p,i}^{(c)} = \frac{1}{2(\lambda_p + p)} \quad (i = 1, 2, \dots, 4p) \end{aligned} \quad (13)$$

This same procedure is repeated for $W_q^{(m)}$ and $W_q^{(c)}$ except for using the corresponding $\lambda_q = (\alpha^2 - 1)q$ instead of $\lambda_p = (\alpha^2 - 1)p$. The true dynamics are then propagated using the variational integrator first introduced by [5] to take initial true states g_k and $\mathbb{V}_k$ to g_{k+1} and $\mathbb{V}_{k+1}$. These newly generated states are passed through a measurement model $h(g_{k+1}, \mathbb{V}_{k+1}, \eta_m) \in \mathbb{R}^q$ where $\eta_m \in \mathbb{R}^q$ is the measurement noise and q is the dimension of the measurement model. If all states are observable, then $p = q$. This new measured state is used to generate an updated Kalman gain which is, in turn, used to update the *a priori* state estimates to the *a posteriori* state estimates in the update step as $\hat{g}_{k+1|k} \rightarrow \hat{g}_{k+1|k+1}$ and $\hat{\mathbb{V}}_{k+1|k} \rightarrow \hat{\mathbb{V}}_{k+1|k+1}$. Finally, the *a posteriori* error covariance matrix is computed as

$$P_{k+1|k+1} = P_{k+1|k} - K_{k+1} P_{zz,k+1} K_{k+1}^T \quad (14)$$

A more detailed description of the implementation of the TSE(3)-based UKF can be found in [7].

Estimation-Based Morse-Lyapunov Control

The TSE(3)-based UKF described was implemented in [8, 9, 13] using the Morse-Lyapunov (M-L) controller from [14]. The controller is defined as

$$u_0 = -\mathbb{I}^{-1} \left(\text{ad}_{\hat{g}}^* \mathbb{I} \hat{\mathbb{V}} - \text{Ad}_{\epsilon_g} \text{ad}_{g_d}^* \mathbb{I} \mathbb{V}_d \right) - \epsilon_\tau + \text{ad}_{\epsilon_v} \text{Ad}_{\epsilon_g} \mathbb{V}_d - \mathbb{I} K_1 \dot{l} - \mathbb{I} K_2 \psi - \mathbb{I} \kappa [0_{1 \times 3}^T (\epsilon_R \epsilon_r)^T]^T \in \mathbb{R}^6 \quad (15)$$

where

$$\epsilon_\tau = \mathbb{I}^{-1} \hat{\tau} - \text{Ad}_{\epsilon_g} \mathbb{I}^{-1} \tau_d \quad (16)$$

is the relative external forces and torques between the estimated and desired models. The gains $K_1 = \text{blkdiag}(k_{11}I_3, k_{12}I_3) \in \mathbb{R}^6 \geq 0$ and $K_2 = \text{blkdiag}(k_{21}I_3, k_{22}I_3) \in \mathbb{R}^{6 \times 6} \geq 0$ are positive definite diagonal gain matrices and κ is a scalar gain coefficient. The backstepping function is defined as

$$\psi = \epsilon_v + K_1 l(\epsilon_g) \quad (17)$$

where

$$l(\epsilon_g) = [s^T(\epsilon_R), \epsilon_r^T]^T \quad (18)$$

such that

$$s(\epsilon_R) = \sum_{i=1}^3 a_i (\epsilon_R^T \tilde{e}_i)^\times \tilde{e}_i \quad (19)$$

and $a \in \mathbb{R}^3$ is a matrix of tunable gain values, where $1 \leq a_1 < a_2 < a_3$ and $\tilde{e}_i$ ($i = 1, 2, 3$) are unit vectors aligned with the vehicle's principal axes. The initial control force $u_0 \in \mathbb{R}^6$ is generated based on the desired control input from the M-L controller, and then constrained based on the spacecraft configuration. The process for generating control on estimated states, constraining that control based on the spacecraft configuration, and then generating the next estimated state is described in Algorithm 1. The numerical stability of this method has been demonstrated here and in previous work [1, 9, 13], but further work is needed to prove the analytical stability of the controller and UKF.

Algorithm 1 Estimation and Control Sequence

Initializing:

Define Initial Conditions $g_0, \mathbb{V}_0, P_0$

Begin Main Iteration Loop:

for $k = 1, 2, \dots$ **do**

1: Estimate Control

$$u_0 = u_0(\hat{g}_{k|k}, \hat{\mathbb{V}}_{k|k})$$

2: Optimally Distribute Control Based on Spacecraft Configuration

$$c = \min \left((\tilde{U}c - u_0)^T (\tilde{U}c - u_0) + kc^T c \right)$$

3: Impose duty cycle by scaling c and recalculate control vector

$$c_{dc} = \frac{c}{\max(c)} d_c$$

$$u_{com} = \tilde{U}c_{dc}$$

$$u_k = u_{com}(1 + \mathcal{N}(0, \eta_T))$$

4: Predict Based on Model

$$[\hat{g}_{k+1|k}, \hat{\mathbb{V}}_{k+1|k}, P_{k+1|k}] = f(\hat{g}_{k|k}, \hat{\mathbb{V}}_{k|k}, P_{k|k}, u_{act})$$

5: Propagate True State

$$[g_{k+1}, \mathbb{V}_{k+1}] = f(g_k, \mathbb{V}_k, u_k)$$

6: Update Based on Measurements

$$[\hat{g}_{k+1|k+1}, \hat{\mathbb{V}}_{k+1|k+1}, P_{k+1|k+1}] = h(\hat{g}_{k+1|k}, \hat{\mathbb{V}}_{k+1|k}, P_{k+1|k}, u_{act})$$

7: Update Variables

$$g_{k+1} = g_k$$

$$\mathbb{V}_{k+1} = \mathbb{V}_k$$

$$\hat{g}_{k|k} = g_{k+1|k+1}$$

$$\hat{\mathbb{V}}_{k|k} = \hat{\mathbb{V}}_{k+1|k+1}$$

$$P_{k|k} = P_{k+1|k+1}$$

end

AUTONOMOUS, DISTRIBUTED, CONSTRAINED CONTROL WITH SATURATION LIMITS

Thruster Configuration Definitions for Autonomous Distributed Control

For a distributed control scheme within the SE(3) framework, consider an arbitrary control input in the body frame $u_0 \in \mathbb{R}^6$. It is desired to determine how this control input is distributed about various actuators and how it determines the gimbaling of those actuators. To distribute control among arbitrary configurations of static and gimbaling thrusters, the components of such a control scheme must first be considered. The static thruster does not move, but when located away from the center of mass has contributions in both translational and rotational motion. The gimbaling thruster may contribute to either or both translational and rotational

motion, depending on gimbal limits. Additionally, gimbaling thrusters have constraints on this contribution within the time domain due to imposed gimbal rate limits.

The total translational and rotational control of the central body is assessed by assembling a matrix of the control vectors with a coefficient describing the throttling or duty cycle of each thruster. The method of distributing the control input to the thrusters as presented as an optimization problem such that

$$u = \sum_{n=1}^N u_n c_n = \sum_{n=1}^N \text{Ad}_{g_{\mathcal{M},n}}^T T_{\mathcal{M},n} c_n = \sum_{n=1}^N \text{Ad}_{g_{\mathcal{M},n}}^T T_{\mathcal{M},n} c_{dc,n} \in \mathbb{R}^6 \quad (20)$$

which could be written as

$$u = \tilde{U}c \in \mathbb{R}^6 \mid \tilde{U} := [u_1, u_2, \dots, u_N] \in \mathbb{R}^{6 \times N} \quad (21)$$

where $\tilde{U}$ is the mapping between individual thrusters and the body frame and c is the duty cycle that best matches the optimal distributed control u . The optimal coefficients for each thruster determined by Eq. (20) may be obtained by solving the optimization problem of the form

$$u^* = \min \left((\tilde{U}c - u_0)^T (\tilde{U}c - u_0) + kc^T c \right) \quad (22)$$

by finding the optimal value of c such that $k \in \mathbb{R}$ is a scalar gain that tunes the deadband region to prevent chattering. To describe the required ratio of duty cycles required to compensate for a maneuver. To implement a guidance algorithm, the above algorithm is extended into the time domain, imposing known constraints on thruster duty cycles. Because a given spacecraft configuration could be configured in a suboptimal way, residual motions about any axis must be given a threshold to avoid saturating the controller.

Saturation Limits

Various methods of imposing saturation limits were considered during the development of this algorithm. It was found that imposing saturation limits on the control vector prior to distributing thrust was ineffective. Furthermore, imposing saturation limits within the optimization algorithm itself was likewise problematic, resulting in all thrusters being on at all times. Instead, an ideal solution of normalizing the vector of thruster ontime coefficients, c demonstrated good behavior. This can be expressed as

$$c_{dc} = \frac{c}{\max(c)} d_c \quad (23)$$

where d_c is the maximum duty cycle during any given time step. For example, for a thruster that is capable of being on 100% of the time during a 1-second thruster firing period during a 1-second time step, $d_c = 1$. For a thruster that can fire at 2 Hz with a full duty cycle of 0.1 second during a 1-second time step, then $d_c = 0.2$. With this, the total control force acting on a given spacecraft configuration with N thrusters is

$$u_{\text{com}} = \tilde{U}c_{dc} \quad (24)$$

where u_{com} is the final actuator value to be commanded to the thrusters and c_{dc} is the optimal value of c solved for in Eq. (22).

Anomaly Simulation and Detection

The anomaly considered in this work is a stuck thruster. The thruster firing history for any given timestep described above may be augmented to simulate a continuously firing thruster. If thruster noise $\eta_T \sim \mathcal{N}(0, \xi)$ is included where $\mathcal{N}$ is the normal Gaussian distribution with covariance ξ , this results in an estimated and true thruster firing history described by

$$\begin{aligned} u_{\text{com}} &= \tilde{U}c_{\text{com}} \\ u_{\text{true}} &= \tilde{U}c_{\text{true}}(1 + \eta_T) \end{aligned} \quad (25)$$

where u_{com} is the commanded control, $\hat{c}_{\text{com}}$ is the commanded thruster firing pattern. c_{true} is the true thruster firing pattern. These may be rearranged to give us an expression for error tolerance in the vehicle performance

$$\epsilon_u = \|\tilde{U}^+(\hat{u} - u)\| \quad (26)$$

where $(\cdot)^+$ denotes the Moore-Penrose inverse, or pseudo inverse. An error tolerance may be established such that ϵ_u should not exceed the 3σ variance due to thruster noise. Thus, any error in excess of this is the result of off-nominal performance and will trigger the abort scenario. Since the true control force u is not knowable, it is necessary to estimate it. By considering the estimated control from the previous time step and comparing it to the measured control u_m by subtracting out the estimated state, an anomaly may be detected. Since the control force is not measured directly, the estimated states from the previous time step using Eq. 3b is rearranged yielding

$$\epsilon_u = \|\tilde{U}^+(\hat{u} - u_m)\| = \|\tilde{U}^+ \left(\hat{u} - \left(\mathbb{I}\dot{\hat{V}} - \text{ad}_{\hat{V}}^* \mathbb{I}\hat{V} - \hat{\tau} \right) \right)\| \quad (27)$$

Algorithm 2 Anomaly Detection Logic

Define Initial Conditions $\hat{g}, \hat{V}, g_d, V_d$

- 1: Determine Desired Control
 $u_d = f(\hat{g}, \hat{V}, g_d, V_d)$
 - 2: Generate Estimated Thruster Firing History
 $c_{\text{com}} = \min \left((\tilde{U}c - u_0)^T (\tilde{U}c - u_0) + kc^T c \right)$
 - 3: Estimate New Control Vector
 $u_{\text{com}} = \tilde{U}c_{\text{com}}$
 - 4: Check Accuracy
 $\epsilon_u = \|\tilde{U}^+ \left(u_{\text{com}} - \left(\mathbb{I}\dot{\hat{V}} - \text{ad}_{\hat{V}}^* \mathbb{I}\hat{V} - \hat{\tau} \right) \right)\|$
 - 5: Check tolerance
 - 6: **if** $\epsilon_u < \sqrt{N\xi^2}$ **then**
 No anomaly detected
 - 7: **else**
 - 8: $c_{\text{com}}(1) = 10$
 Anomaly is detected, Abort Scenario Initiated
 - 9: **end if**
 - 10: **if** $c_{\text{com}}(1) == 10$ **then**
 - 11: $u_d = f(\hat{g}, \hat{V}, g_{d,0}, V_{d,0})$
 - 12: $c_{\text{com,new}} = \min \left((\tilde{U}c - u_0)^T (\tilde{U}c - u_0) + kc^T c \right)$
 - 13: $\hat{u} = \tilde{U}c_{\text{com,new}}$
 - 14: **end if**
-

NUMERICAL SIMULATION RESULTS

The TSE(3)-UKF used in this work employs a dynamic model and a measurement model that estimates pose and augmented velocity. Thruster noise is also considered whenever the thrusters are fired, modeled as a noise parameter. The list of noise values used in this work is provided in Table 1. (m) denotes measurement noise while (p) denotes process noise.

Nominal RPOD Scenario

Existing simulations are similar to previous work, but include the addition of measurement and process noise interpreted with a UKF on SE(3) 11, 12. Furthermore, thruster noise has likewise been added to the

system, which introduces uncertainty in control output. The control of the spacecraft is accomplished using a tracking M-L controller described by [7, 14].

The approach profile for the RPOD simulation consists of a reference trajectory created using a 3-DoF simulation with impulse burns to create each phase. The simulation begins with a 19-minute approach and 1-minute hold at the keep-out sphere (KoS), selected to be 10 km in radius, followed by an approach of 6.6-4.2 m/s lasting 170 seconds, a close approach of 2.4 m/s lasting 150 seconds, and a final approach of 50 cm/s lasting 700 seconds, resulting in a final 69 seconds of docking with a velocity of 15 cm/s. This 3-DoF simulation is then used as a reference from which to build the 6-DoF counterpart. To include attitude, the vehicle is presumed to be facing along the (relative) velocity direction during approach until entering the approach sphere, at which point the desired attitude of the spacecraft is one in which the first principal axis points along the Gateway y-axis (docking axis), and the z-axes are aligned.

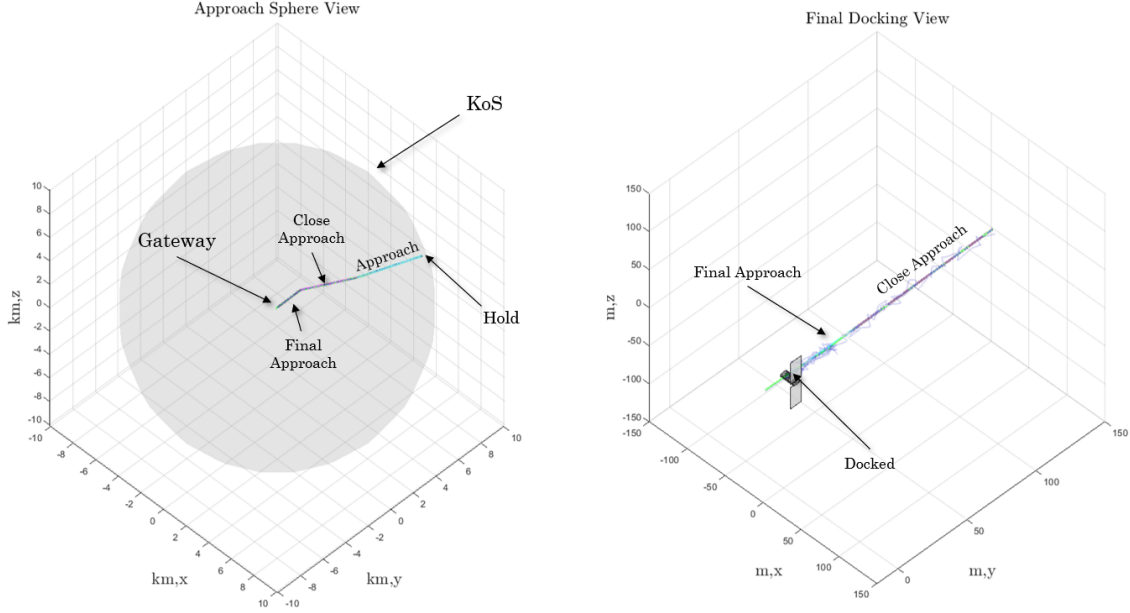


Figure 2: The approach of a 16-thruster tug-cargo configuration spacecraft along a reference 3-DoF RPOD trajectory with impulse burns. Purple and cyan indicate approach segments.

A deadband is imposed for the control that may be tuned to obtain the desired load on the jet firing history. With no deadband, thrusters constantly fire and oscillate between controllable configurations. By allowing the vehicle to drift freely when tighter constraints on pose are not needed, a more fuel-optimal approach is

Table 1: Standard deviation for each source of noise.

Property	Parameter	Value	Units
Attitude (m)	η_R	$5 \frac{\pi}{180}$	rad
Position (m)	η_r	10	m
Angular Velocity (m)	η_ω	$0.5 \frac{\pi}{180}$	rad/s
Velocity (m)	η_v	0.1	m/s
Attitude (p)	η_R	$1.6 \frac{\pi}{180}$	rad
Position (p)	η_r	3.3	m
Angular Velocity (p)	η_ω	$0.16 \frac{\pi}{180}$	rad/s
Velocity (p)	η_v	0.033	m/s
Thrust	η_T	5	%

obtained. The results of the current deadband constraint $k = 0.1\|r_{rel}\|$ where r_{rel} is the relative position is visible in the tracking, measurement, and estimation errors illustrated in Figure 3 and 4.

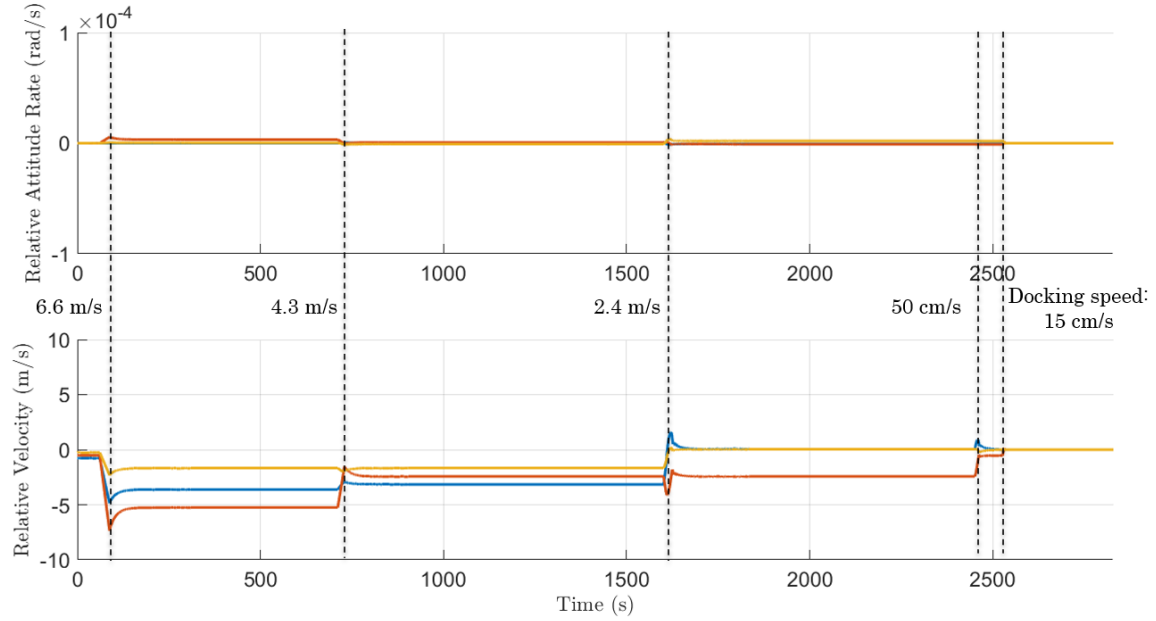


Figure 3: The position and attitude tracking error, where the dotted line is the deadband region for deadband control scalar k .

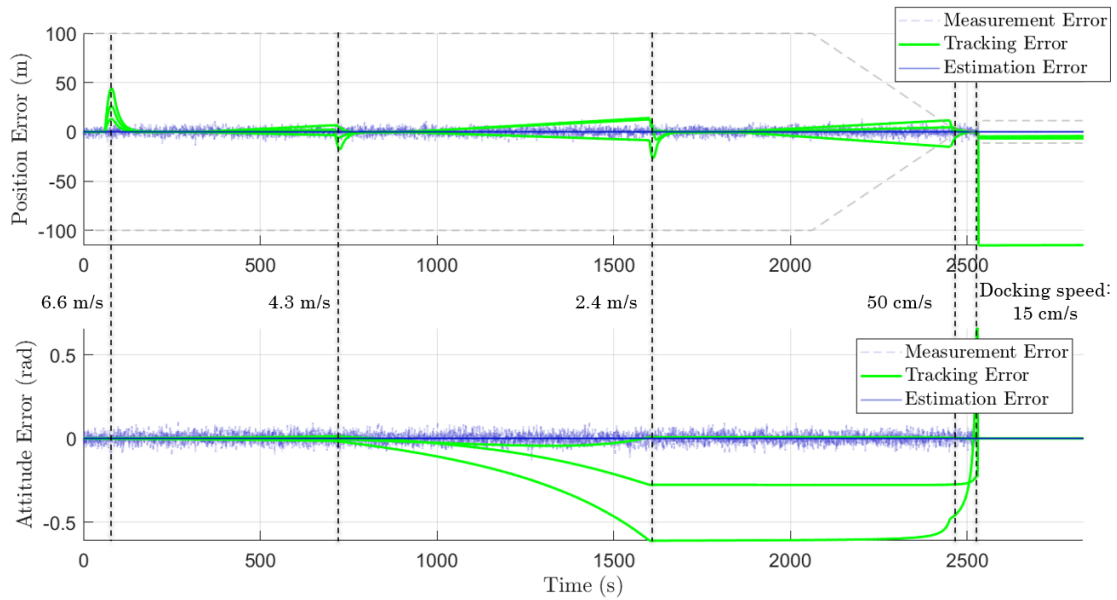


Figure 4: The attitude rate and velocity tracking errors.

The control input from the M-L controller, the estimated distributed control, and the true distributed control are illustrated in Figure 5. The attitude component of the M-L control stands out as the saturation limits imposed by the given thruster configuration make obtaining those attitude rates while still maintaining a con-

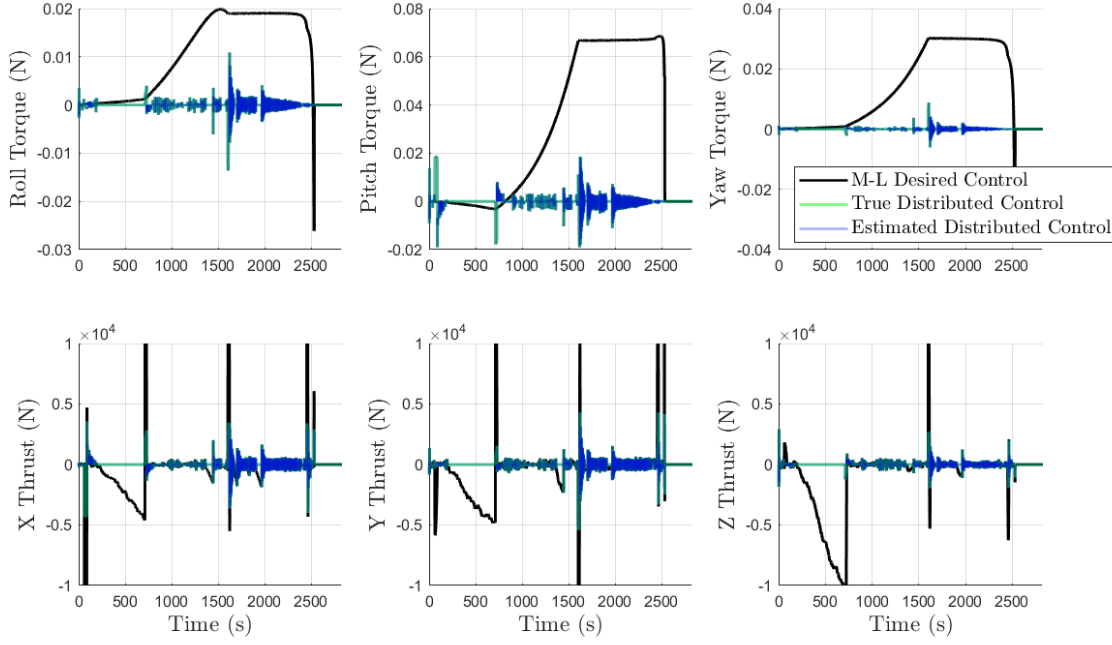


Figure 5: The desired (Morse-Lyapunov) control about the body frame compared to the estimated distributed, and true distributed control input.

trolled translational approach unobtainable. That being said, the desired final pose is obtained by gradually pursuing the the optimal available jet firing histories over the simulation time. The oscillatory behavior of the true and estimated thruster input is a again result of the given thruster configuration in which all the thrusters are mounted in one end of the vehicle and the center of mass of the vehicle lies outside of the geometric center of the thrusters.

The jet firing history resulting from the control input in Figure 5 is illustrated in Figure 6, where the y-axis is scaled as simply on/off (alternatively, it can be thought of as a normalized value between d_c and 0). These values are not distributed and constrained, but not discretized. Since true duty cycles are actually discretized on some number of intervals between 0 and d_c , it is desired to implement discretization of normalized outputs in future work. However, efforts to impose this discretization on the current methodology has resulted in unstable behavior.

The propellant use during the nominal RPOD simulation, both in terms of delta-V and propellant mass, is illustrated in Figure 7. The values of around 100 kg and 20 to 25 m/s align with expected values for a vehicle of this size and for these approach speeds. Other burns required for rendezvous between spacecraft such as the insertion burn into the orbit are not considered.

RPOD Scenario with Stuck Thruster and Abort

In the second scenario, a stuck thruster (thruster number 4) is simulated 1 km from the Gateway. The anomaly detection logic is implemented and demonstrated. The scenario is initiated with identical initial conditions as discussed in the first scenario, including a hold at the KoS and approach burns at 6.6-4.3 and 2.4 m/s. However, before the final approach burn at 50 cm/s can be performed, thruster 4 is stuck open. Upon detecting the anomaly, the vehicle initiates an emergency abort maneuver in the direction opposite of Gateway. Although a pure abort vector cannot be obtained due to the stuck thruster, the combined force of the other thrusters opposing any force from the stuck thruster ensures a successful abort.

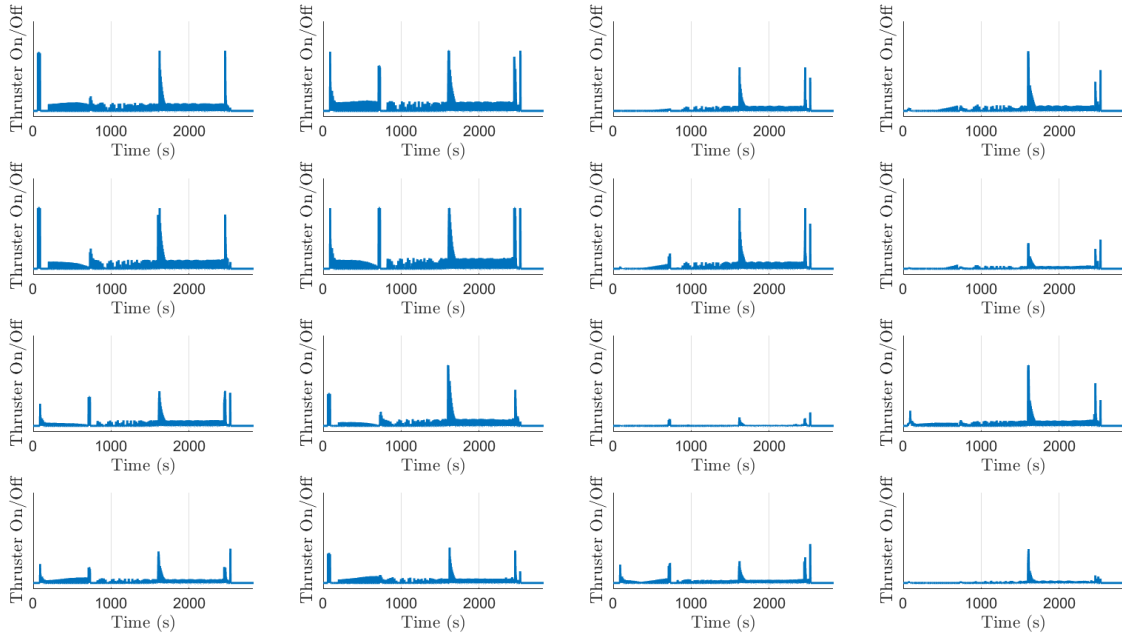


Figure 6: The distributed jet firing history during a nominal RPOD trajectory.

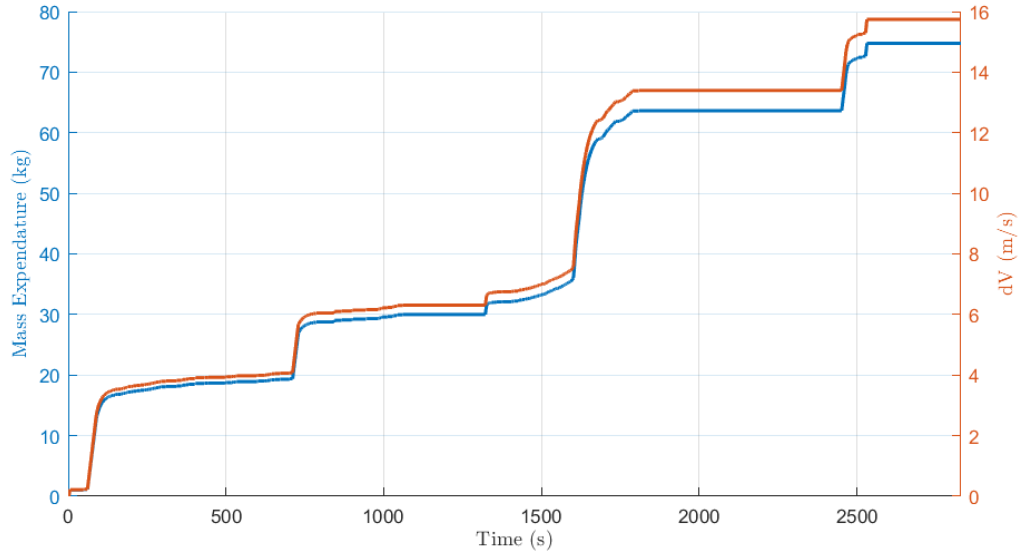


Figure 7: The propellant used for the full, nominal RPOD trajectory.

Figure 8 shows the abort scenario play out from rendezvous down to 1 km. After an abort is achieved and all thrusters (including the stuck thruster) are disabled, the vehicle enters a dormant state, simulating time for diagnostics to occur. During this time, the vehicle is spinning slowly as a result of that stuck thruster but not in such a way that cannot be nominally recovered. Comparing Figures 9 and 10 with their counterparts from the nominal scenario, it is clear that the first phases of the approach are identical. However, the undesired motion imposed by the stuck thruster and rapid retreat overwhelms the otherwise nominal approach. The

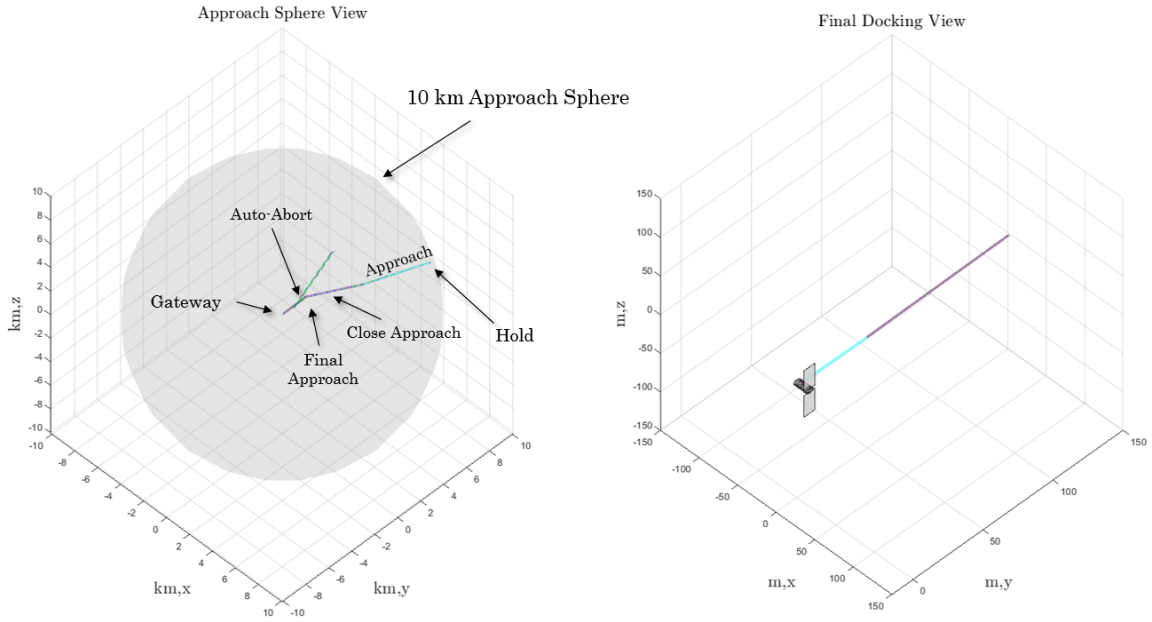


Figure 8: The aborted approach of the same spacecraft.

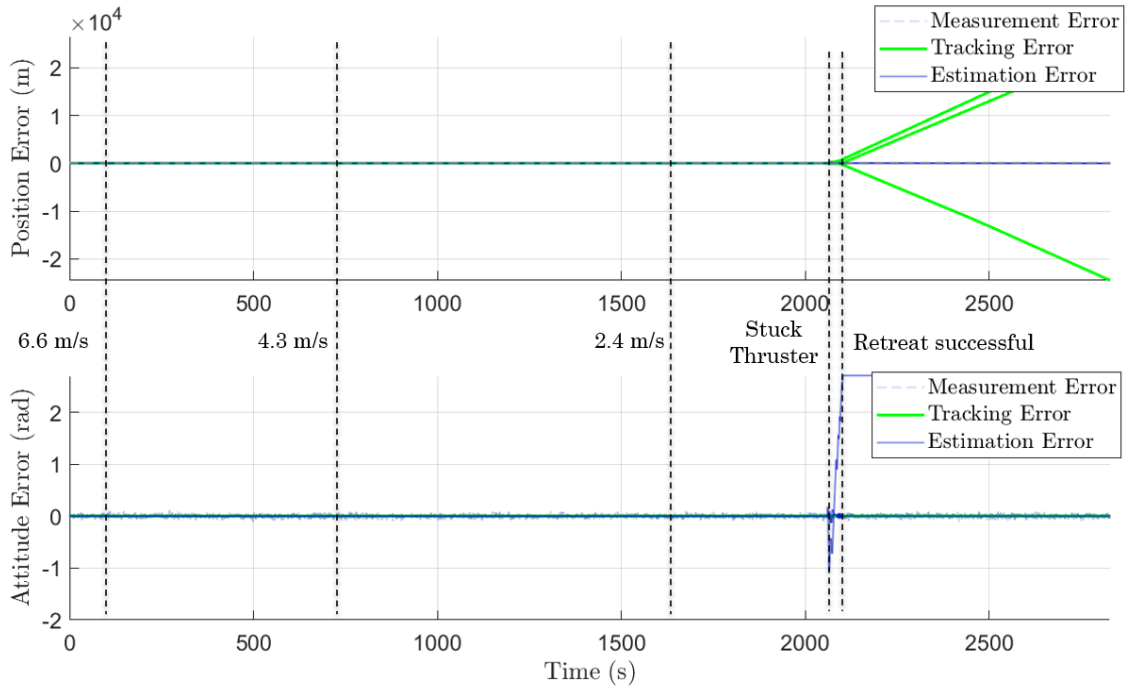


Figure 9: The relative attitude during the RPOD trajectory with abort scenario.

tracking error grows over time due to the vehicle departing the original desired trajectory.

The stuck thruster simulated by during the abort scenario is clearly indicated in Figure 12 as the green, true performance of thruster number 4. The different between the true control input and estimated control input

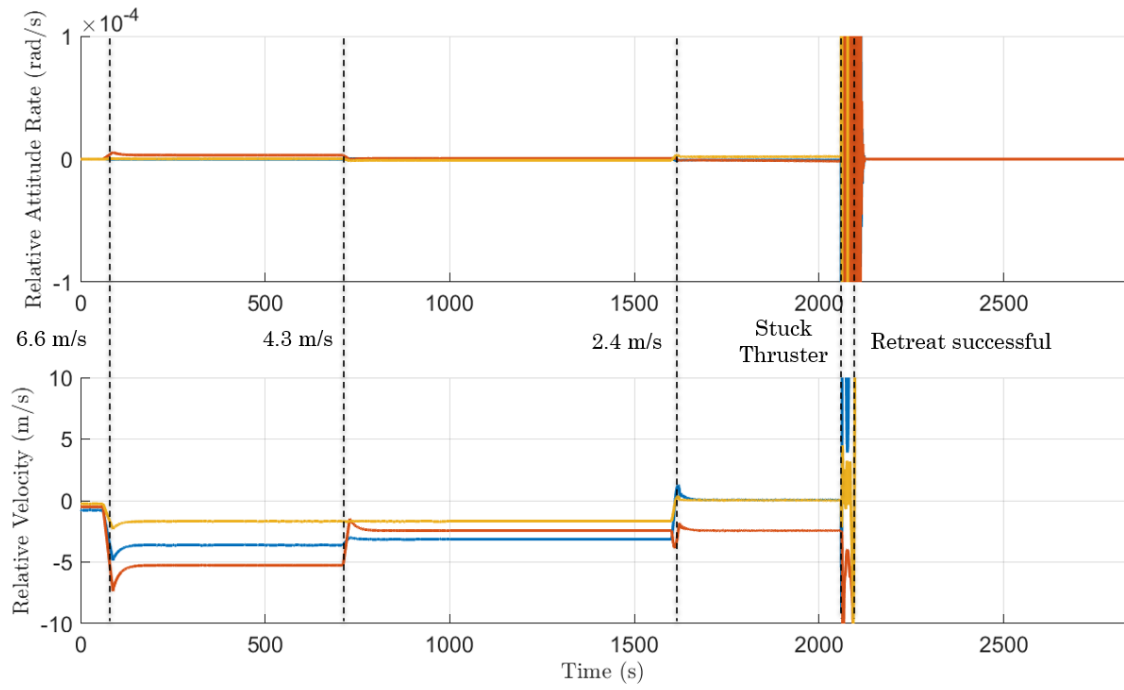


Figure 10: The relative position and velocity during the RPOD abort scenario.

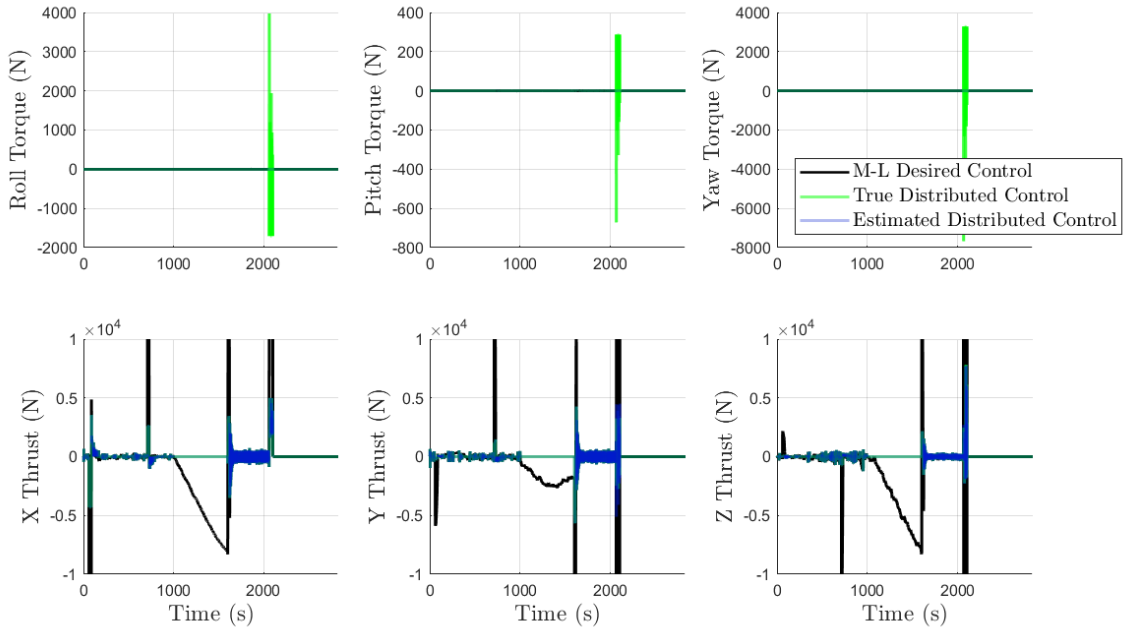


Figure 11: The body-frame Morse-Lyapunov control vs. the estimated and true distributed control inputs.

is also illustrated in Figure 11 where red indicates estimated thruster performance while green indicates true thruster performance. Figure 13 shows slightly increased propellant use due to the abort burn taking slightly longer than the final approach burn as it fights the stuck thruster. Notionally, with additional prop budget

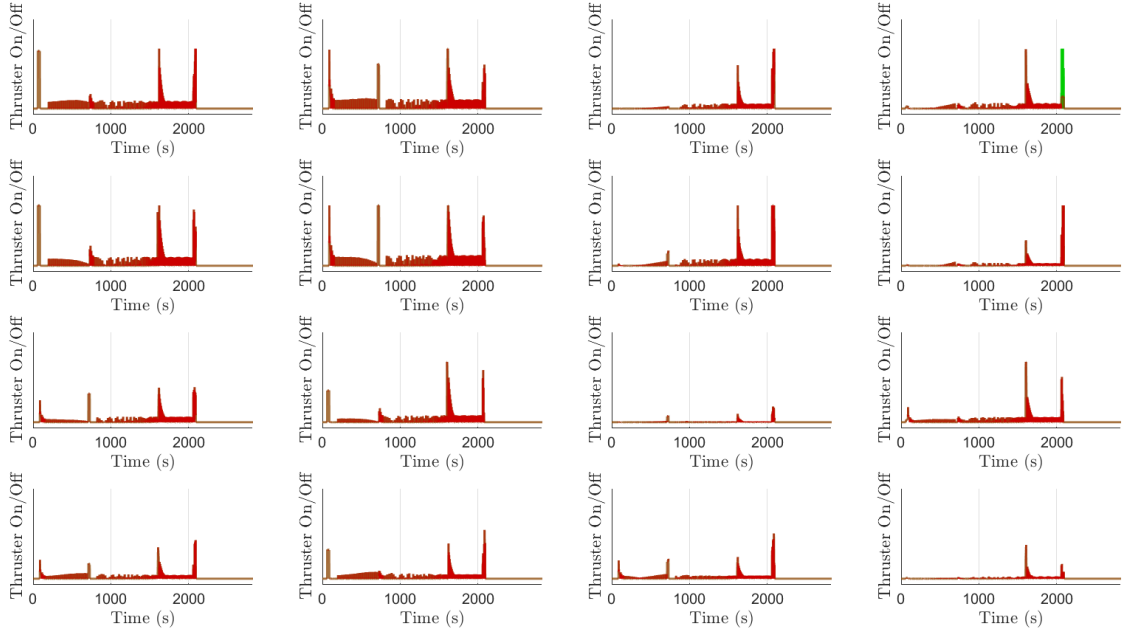


Figure 12: The distributed RCS firing history during the abort scenario.

being allotted for such scenarios this approach may still be recoverable.

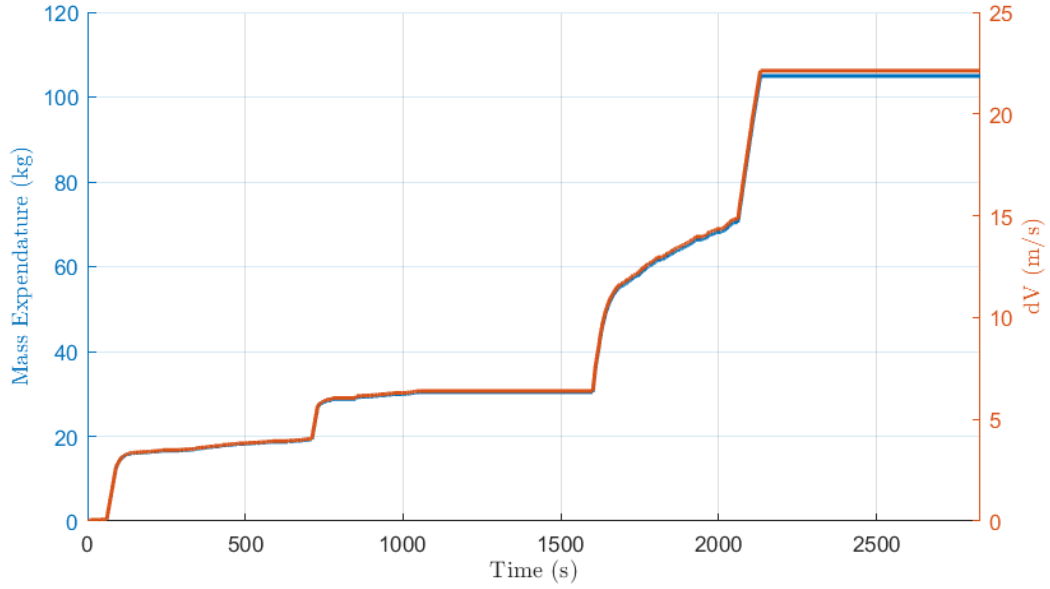


Figure 13: Propellant used during the abort scenario.

CONCLUSIONS

In this work, a Morse-Lyapunov (M-L) tracking controller on TSE(3) first introduced in [7, 14] has been implemented for the problem of a rendezvous, proximity operations, and docking (RPOD) problem between

a logistics module and NASA's Gateway in the presence of thruster, measurement, and dynamical model uncertainties. The uncertainties have been filtered using an augmented unscented Kalman filter (UKF) on SE(3) first proposed by [11, 12]. This work has demonstrated the extension of the aforementioned M-L controller with a UKF to include autonomous jet firing history generation, even when the thruster configurations are not oriented about the center of mass of the vehicle based on the findings of [1, 2]. Furthermore, it has also been extended to include autonomous abort scenarios when a difference between the commanded and measured control force exceeds a given threshold, this time set as the expected 3σ noise distribution of the thrusters. The simulation was able to guide the spacecraft to a safe docking state with a velocity of 15 cm/s at the axial docking port. The benefits of this work are an increased robustness in the current state-of-the-art for RPOD operations, which have historically been well within the regime of human intervention and observation. Furthermore, the ability for the vehicle to abort autonomously has the potential to reduce the risk of implementing autonomous control systems near high-value assets such as the Gateway. Future work may seek to identify current gaps in this work - specifically the question of thruster bias, conditional attitude constraints on docking, dynamic mass properties, and incomplete sensor models.

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