International Journal of Electrical and Computer Engineering (IJECE)
Vol. 7, No. 4, August 2017, pp. 1994~2001
ISSN: 2088-8708, DOI: 10.11591/ijece.v7i4.pp1994-2001

Asymptotic Stability of Quaternion-based Attitude Control System with Saturation Function

Harry Septanto[1], Djoko Suprijanto[2]
[1]Satellite Technology Center, National Institute of Aeronautics and Space (LAPAN), Indonesia
[2]Combinatorial Mathematics Research Group, Faculty of Mathematics and Natural Sciences, Institut Teknologi Bandung, Indonesia

Article Info

Article history:

Received Dec 9, 2016
Revised May 28, 2017
Accepted Jun 12, 2017

Keyword:

Asymptotically stable
Attitude control
Quaternion
Rigid body
Saturation

ABSTRACT

In the design of attitude control, rotational motion of the spacecraft is usually considered as a rotation of rigid body. Rotation matrix parameterization using quaternion can represent globally attitude of a rigid body rotational motions. However, the representation is not unique hence implies difficulties on the stability guarantee. This paper presents asymptotically stable analysis of a continuous scheme of quaternion-based control system that has saturation function. Simulations run show that the designed system applicable for a zero initial angular velocity case and a non-zero initial angular velocity case due to utilization of deadzone function as an element of the defined constraint in the stability analysis.

Corresponding Author:

Harry Septanto,
Satellite Technology Center,
National Institute of Aeronautics and Space (LAPAN),
Jl. Cagak Satelit km 0.4, Rancabungur, Bogor 16310, Indonesia.
Email: harry.septanto@lapan.go.id

1. INTRODUCTION

Design of a spacecraft or satellite attitude control by analyzing rotational motion of a rigid body remains become a challenging research field until recent years. Ashok et al proposed control moment gyros based attitude control system to achieve time-optimal maneuver for agile (rigid) satellite [1]. Chabot and Schaub presented a spherical actuator for satellite attitude control that covers modeling, simulation of attitude control of a rigid body system motion as well as comparison with a configuration of three reaction wheels [2]. in [3], Stevenson and Schaub used rigid body approach in the attitude control development prior to do testbed experiment of remote electrostatic charge control. Rezanezhad in [4] presented Takagi-Sugeno fuzzy-based attitude controller in order to reduce thruster fuel consumption and increase longevity of satellite. Particle swarm optimization algorithm is used to reduce limit cycle on the fuzzy system. Pirouzmand in [5] proposed a model reference adaptive system-based robust model predictive controller for three degree of freedom satellite attitude control system. Whilst the controller gain is obtained through solving a convex optimization problem using linear matrix inequality approach.

The kinematics of a rigid body rotational motion is represented by rotation matrix that is member of the special orthogonal three group, $SO(3)$. Among parameterizations of the rotation matrix, a parameterization using quaternion is the only parameterization with four parameters. Hence, it can represent global attitude of a rigid body rotational motion. However, a physical attitude, which is represented in a unique rotation matrix value, is represented by two values in quaternion, i.e. a pair antipodal value. This fact implies difficulties on the stability guarantee of the quaternion based attitude control system [6].

Many research efforts have been addressed on the design of quaternion based attitude control system: MacKunis et al in [7] developed an adaptive neural network based attitude controller for satellite that is actuated by control moment gyros, where external disturbance as well as disturbance of the control moment gyros' tachometer are considered; Calvo et al. proposed an adaptive fuzzy controller for momentum wheel actuated satellite and its performance is compared with a custom PD controller [8]; in [9], Septanto et al proposed a continuous scheme of quaternion based controller that employs augmented dynamic; to name a few. In [10], a continuous scheme of quaternion-based control system that has saturation function is also proposed with boundedness of solution guarantee. In this paper, the system has been further analyzed to have stronger guarantee, i.e. asymptotically stable.

The organization of the paper is the following: mathematical preliminaries including modeling of the satellite dynamics and kinematic are presented in the next section. Section 3 presents the problem formulation and methodology. The main contribution of this paper will be presented in Section 4. Discussion and numerical simulations are also presented in Section 5. Section 6 provides the concluding remarks.

2. BACKGROUND AND PRELIMINARIES

2.1. Mathematical Preliminaries

Some mathematical notations will be used in the rest of paper. $\mathbf{R}$ is the set of all real numbers. $\mathbf{R}^{n\times m}$ is the set of all $n \times m$ matrix that all of its entry are real numbers, where $n, m \in \mathbf{Z}_{>0}$, $\mathbf{Z}$ is the set of all integer numbers and $\mathbf{Z}_{>0}$ is the set of all positive integer numbers. Consider a matrix $R \in \mathbf{R}^{n\times m}$, hence R^T denotes transpose of R and $|R|$ determinant of R. Matrix $I^{n\times n} \in \mathbf{R}^{n\times n}$ denotes the identity of $n \times n$ matrix. Suppose there is a column matrix $C \in \mathbf{R}^n$, hence $\|C\|$ denotes 2-norm of C. Matrix $0_{n\times m}$ denotes $n \times m$ matrix that all of its entry are zero.

In this paper, a vector is defined as in (1).

$$\vec{r} = r^{I^T} \mathsf{F}_I \tag{1}$$

where: $\vec{r}$ denotes the vector of r, F_I is a columns matrix consists of three unit vector $\hat{l}_1, \hat{l}_2$ and $\hat{l}_3$ that are associated with the inertial reference frame, and $r^I \in \mathbf{R}^3$ is a column matrix whose three components of $\vec{r}$ that are expressed (decomposed) into the inertial reference frame, F_I.

Besides superscript ^{I} (subscript $_I$) that are associated with the inertial reference frame, this paper uses superscript ^{b} (subscript $_b$) and superscript ^{d} (subscript $_d$) associated with the satellite's fixed body frame and satellite's desired frame, respectively. For brevity, the satellite's fixed body frame, the inertial reference frame and the satellite's desired frame may be written as body frame, inertial frame and desired frame, respectively.

2.2. Dynamic and Kinematics of Spacecraft

In this paper, motion of a satellite is regarded as a rigid body motion. Dynamic of a rigid satellite is given by Euler Equation, [11]. The dynamic of a satellite that is expressed in F_b is represented in (2).

$$J\dot{\omega}_{bI}^{b} = -\omega_{bI}^{b} \times J\omega_{bI}^{b} + \tau = -{}_{\mu}\omega_{bI}^{b} J\omega_{bI}^{b} + \tau \tag{2}$$

where the symmetric positive definite matrix $J \in \mathbf{R}^{3\times 3}$ is the satellite inertia moment about its center of mass that is located in the origin of F_b, (kg.m^2); $\omega_{bI}^{b} \in \mathbf{R}^3$ is the angular velocity vector of F_b with respect to F_I which is decomposed in F_b, (rad.s^{-1}); and $\tau \in \mathbf{R}^3$ is the total external control torque about its center of mass that is located in the origin of F_b, (N.m). ${}_{\mu}\omega_{bI}^{b}$ denotes a skew-symmetric matrix of ω_{bI}^{b}, i.e. the column matrix of vector $\vec{\omega}_{bl}$ that is expressed in F_b, where $\omega_{bl}^b = \begin{bmatrix} \omega_{bl1}^b & \omega_{bl2}^b & \omega_{bl3}^b \end{bmatrix}^T$ and $_\mu\omega_{bl}^b = \begin{bmatrix} 0 & -\omega_{bl3}^b & \omega_{bl2}^b \\ \omega_{bl3}^b & 0 & -\omega_{bl1}^b \\ -\omega_{bl2}^b & \omega_{bl1}^b & 0 \end{bmatrix}$. In addition, $\dot{\omega}_{bl}^b$ denotes first derivative of ω_{bl}^b with respect to time.

Satellite attitude representation in (unit) quaternion that denotes the attitude of the satellite body frame F_b with respect to the inertial frame F_l is given by $q_{bl} = \begin{bmatrix} \eta_{bl} & \varepsilon_{bl}^T \end{bmatrix}^T$, where $\eta_{bl} \in \mathbf{R}$, $\varepsilon_{bl} \in \mathbf{R}^3$ and $\sqrt{\eta_{bl}^2 + \varepsilon_{bl}^T \varepsilon_{bl}} = 1$. In addition, all attitude representations in quaternion regarded in this paper, including q_{bl}, are member of the unit sphere order 3, $S^3 = \left\{ \begin{bmatrix} a_0 & a_1 & a_2 & a_3 \end{bmatrix}^T \in \mathbf{R}^4 : \sqrt{a_0^2 + a_2^2 + a_2^2 + a_3^2} = 1 \right\}$.

Now, consider a kinematics Equation represented in quaternion given by (3)-(5).

$$\dot{q}_{bd} = \begin{bmatrix} \dot{\eta}_{bd} & \dot{\varepsilon}_{bd}^T \end{bmatrix}^T \tag{3}$$

Where

$$\dot{\eta}_{bd} = -\frac{1}{2} \varepsilon_{bd}^T \omega_{bd}^b \tag{4}$$

$$\dot{\varepsilon}_{bd} = \frac{1}{2} \left({}_\mu\varepsilon_{bd} + \eta_{bd} I \right) \omega_{bd}^b \tag{5}$$

Note that q_{bd} represents the attitude F_b with respect to F_d, q_{bd} is also regarded as the attitude error between the satellite body frame F_b and the satellite desired or target frame F_d. Since the information from an attitude sensor is with respect to the inertial frame F_l, i.e. q_{bl} and the target attitude is also with respect to F_l, i.e. q_{dl}, hence the attitude error q_{bd} is obtained form quaternion multiplication between q_{bl} and q_{dl} as given in (6).

$$q_{bd} = q_{dl}^{-1} \breve{\times} q_{bl} = q_{ld} \breve{\times} q_{bl} \tag{6}$$

where $q_{dl}^{-1} = q_{ld}$ is the inverse of q_{dl}. Note that since the attitude is considered as a unit quaternion, where the quaternion norm $\|q_{dl}\| = \sqrt{\eta_{dl}^2 + \varepsilon_{dl}^T \varepsilon_{dl}} = 1$, then quaternion inverse, $q_{dl}^{-1} = \frac{q_{dl}^*}{\|q_{dl}\|}$, is equal to the quaternion conjugate, $q_{dl}^* = \begin{bmatrix} \eta_{dl} & -\varepsilon_{dl}^T \end{bmatrix}^T$.

3. PROBLEM DEFINITION AND METHODOLOGY

Consider the spacecraft (2) and the controller τ (7) proposed in [11], where $\Phi\left(\varepsilon_{bd}\right) : \mathbf{R}^3 \to \mathbf{R}^3$ denotes saturation function with saturation level $\overline{\phi} > 0$. The saturation function is defined element-wise $\Phi\left(\varepsilon_{bd_i}\right)$, $i \in \{1, 2, 3\}$, $\varepsilon_{bd} = \begin{bmatrix} \varepsilon_{bd_1} & \varepsilon_{bd_2} & \varepsilon_{bd_3} \end{bmatrix}^T$, in particular given by (8). For convenience consideration, $\Phi\left(\varepsilon_{bd}\right) = \Phi_X$. Deadzone function Ψ_X is defined as $\Psi_X = \varepsilon_{bd} - \Phi_X$, hence $\tau = -k\left(\eta_{bd}\varepsilon_{bd} - \Psi_X\right) - L\omega_{bl}^b$. In addition, the controller τ (7).

$$\tau = -k\left(\eta_{bd}\varepsilon_{bd} + \Phi_X - \varepsilon_{bd}\right) - L\omega_{bd}^{b},\ k \in \mathbf{R},\ L \in \mathbf{R}^{3\times3} \tag{7}$$

$$\Phi\left(\varepsilon_{bd}\right) = \left[\Phi\left(\varepsilon_{bd_1}\right) \quad \Phi\left(\varepsilon_{bd_2}\right) \quad \Phi\left(\varepsilon_{bd_3}\right)\right]^T, \Phi\left(\varepsilon_{bd_i}\right) = \begin{cases} \bar{\phi}, \varepsilon_{bd_i} > \bar{\phi} \\ \varepsilon_{bd_i}, -\bar{\phi} \le \varepsilon_{bd_i} \le \bar{\phi} \\ -\bar{\phi}, \varepsilon_{bd_i} < -\bar{\phi} \end{cases} \tag{8}$$

The designed attitude control system is for the case of constant attitude tracking, i.e. $\omega_{dl}^{b} = R_{bd}{}^{T}\omega_{dl}^{d} = R_{db}\omega_{dl}^{d} = 0$, where R_{bd} is the rotation matrix that represents the attitude error between the satellite body frame F_b and the satellite desired or target frame F_d. Rotation matrix R_{bd} is a member of special orthogonal group order 3, i.e. $R_{bd} \in SO(3) = \left\{R \in \mathbf{R}^{3\times3} : RR^T = I^{3\times3}, |R| = 1\right\}$. In [11], there is relation between rotation matrix R_{bd} and quaternion q_{bd} as given by Equation (9).

$$R_{bd} = \left(\eta_{bd}{}^{2} - \varepsilon_{bd}{}^{T}\varepsilon_{bd}\right)I^{2\times2} + 2\varepsilon_{bd}\varepsilon_{bd}{}^{T} - 2\eta_{bd\ \mu}\varepsilon_{bd} \tag{9}$$

Definition 1 Let $\tilde{E} = S^3 \times \mathbf{R}^3 \subset \mathbf{R}^6$ is the set of all solutions of a control system. Let E is the set that consists of all non-equilibrium solution E_0 such that $E \cup E_0 = \tilde{E}$, where $\bar{E}_0 \in E_0$. If the following statements are satisfied:

a. $\exists y \in E$, $V(y) > 0, \dot{V}(y) \le 0$
b. $\forall y \in \bar{E}_0, V(y) = 0$
c. $\bar{E}_0$ is the largest invariant set in $\Omega = \left\{y \in E : \dot{V}(y) = 0\right\}$

then $\bar{E}_0$ is the locally asymptotically stable set.

Remarks 1. The stability notion in Definition 1 is not a standard stability in the sense of Lyapunov which is used to guarantee stability of a point. Instead, LaSalle's invariance principle theorem in conjunction with Lyapunov function properties is used to guarantee stability of the set. One may said that LaSalle's theorem extends Lyapunov's theorem since, naturally, it can be used for the system that has an equilibrium set [12].

The main objective of this paper is to find necessary conditions of the system that consists of the rigid spacecraft (2) and the controller (9) such that asymptotic stability guarantee as in Definition 1 is achieved. Through Lyapunov stability method, conditions of the attitude control system parameters will be resulted. In addition, some numerical simulations are done to illustrate its performance.

4. MAIN RESULT

Theorem 1 Consider the quaternion-based spacecraft attitude control system consists of &2), (7) and Definition 1, where $y = \left(q_{bd}, \omega_{bl}^{b}\right)$. The set $\bar{E}_0 = \left\{\left(q_{bd} = \left[\eta_{bd} \quad \varepsilon_{bd}{}^{T}\right]^T, \omega_{bl}^{b}\right) \in S^3 \times \mathbf{R}^3 : \varepsilon_{bd} = 0_3, \eta_{bd} = \pm 1, \omega_{bl}^{b} = 0_3\right\}$ that consists of two equilibrium points of the system is locally asymptotically stable and it satisfies the following properties

a. There exist positive k and symmetric definite positive matrix L such that fulfills the inequality $k - \lambda_{\min}(L) < 0$, where $\lambda_{\min}(L)$ is minimum Eigen value of L
b. The saturation limit $\bar{\phi}$ fulfills $0 < \bar{\phi} < \sqrt{\frac{1}{3}}$
c. There exist initial condition $\left(q_{bd}(t=0), \omega_{bl}^{b}(t=0)\right) \in E$ such that constraint (10) is fulfilled

$$\left\|\omega_{bl}^{b}(t)\right\|^2 - \left\|\Psi_X(t)\right\|^2 \ge 0, t \ge 0 \tag{10}$$

Proof:

Consider the Lyapunov function candidate given in Equation (11) to show set stability of $\overline{E}_0$.

$$V = k\varepsilon_{bd}{}^{T}\varepsilon_{bd} + \frac{1}{2}\omega_{bl}^{b\,T} J \omega_{bl}^{b} \tag{11}$$

where $V > 0$, for all $k > 0$. Note that, instead of q_{bd}, only ε_{bd} is appeared in (11) since η_{bd} is inherently in ε_{bd}. Furthermore, the time derivative of V is given by (12).

$$\dot{V} = k\varepsilon_{bd}{}^{T}\left({}_{\mu}\varepsilon_{bd}\omega_{bd}^{b} + \eta_{bd}\omega_{bd}^{b}\right) + \omega_{bl}^{b\,T}\left(-{}_{\mu}\omega_{bl}^{b} J\omega_{bl}^{b} - k\left(\eta_{bd}\varepsilon_{bd} - \Psi_X\right) - L\omega_{bl}^{b}\right) \tag{12}$$

From the facts that $\omega_{dl}^{d} = 0_3$ and $\varepsilon_{bd}{}^{T}\,{}_{\mu}\varepsilon_{bd} = \omega_{bl}^{b\,T}\,{}_{\mu}\omega_{bl}^{b} = 0$, hence (13) is satisfied.

$$\dot{V} = k\omega_{bl}^{b\,T}\Psi_X - \omega_{bl}^{b\,T} L\omega_{bl}^{b} \tag{13}$$

If L is a symmetric definite positive matrix, then (14) is fulfilled.

$$\dot{V} \le k\left\|\omega_{bl}^{b}\right\|\left\|\Psi_X\right\| - \lambda_{\min}\left(L\right)\left\|\omega_{bl}^{b}\right\|^2 \tag{14}$$

Note that the constraint (10) implies $\dot{V} \le 0$, if $k - \lambda_{\min}\left(L\right) < 0$.

Let $\Omega = \left\{\left(q_{bd}, \omega_{bl}^{b}\right) \in S^3 \times \mathbf{R}^3 : \dot{V}\left(q_{bd}, \omega_{bl}^{b}\right) = 0\right\}$. Then $\dot{V} = 0$, if $\left\|\omega_{bl}^{b}\right\| = 0$. In addition, since constraint (10) is hold, $\left\|\omega_{bl}^{b}\right\| = 0$ implies $\left\|\Psi_X\right\| = 0$. Note that $\left\|\omega_{bl}^{b}\right\| = 0 \Rightarrow \omega_{bl}^{b} = 0_3$ and $\left\|\Psi_X\right\| = 0 \Rightarrow$ $\Psi_X = 0_3$. Substituting these values to the system (2) and (7) it would result $\varepsilon_{bd} = 0$. Therefore, $\Omega = \overline{E}_0$.■

Remarks 2 In according to Proposition 1 in [10], the attitude control system is locally Lipschitz in $\left(q_{bd} = \left[\eta_{bd} \quad \varepsilon_{bd}{}^{T}\right]^{T}, \omega_{bl}^{b}\right) \in S^3 \times \mathbf{R}^3$. Hence, a prerequisite condition for Lyapunov theorem utilization is fulfilled.

Remarks 3 The stability analysis presented in this paper is no claim of global stability. In addition, global stability claim in [13] is incorrect since there are equilibrium points other than $\overline{E}_0$, i.e. $\left\{\left(q_{bd} = \left[\eta_{bd} \quad \varepsilon_{bd}{}^{T}\right]^{T}, \omega_{bl}^{b}\right) \in S^3 \times \mathbf{R}^3 : \omega_{bl}^{b} = 0_3, \eta_{bd}\varepsilon_{bd} - \Psi_X = 0_3\right\}$. Note that this fact is an implication of the chosen saturation limit $\overline{\phi}$ that fulfills $0 < \overline{\phi} < \sqrt{\frac{1}{3}}$.

5. DISCUSSION AND NUMERICAL EXAMPLES

The result presented in the previous section provides advancement into two directions. First, its asymptotic stability guarantee presents a stronger stability guarantee than the stability guarantee presented in [10], i.e. boundedness of solution guarantee. Second, this result corrects the global stability claim in [13], as stated in Remarks 3.

In addition to the theoretical result, the attitude control system's performance will be illustrated via simulations that are run through three scenarios presented in Table 1 and the rest arbitrary-chosen parameters are given in Table 2. Simulation results are represented in Figure 1and Figure 2.

The simulation of Scenario 1 is run with non-zero initial of angular velocity ω_{bl}^{b}. Whilst in Scenario 2, the simulation is run with zero initial of angular velocity ω_{bl}^{b}. Figure 1(a), Figure 1(b) and Figure 2(b) verify that the designed controllers satisfy the Theorem 1. In particular for Scenario 2, it is very interesting and shows that utilization of the deadzone function Ψ_X in the defined constraint (10) provides its advantage to allow the attitude control system to start in a non-zero angular velocity. A graphic in relating to the angular velocity ω_{bl}^{b} trend is shown in Figure 2(a). In addition, it is also interesting to observe the unwinding phenomenon existence. Hence, euler angle trends from the Scenario 1 is compared to the one form Scenario 3. Note that all parameters of Scenario 1 and Scenario 3 are same, except the desired attitude q_{dl} value where both actually represent a same physical condition. The Euler angle trends outlined in Figure 2(c) show that the designed system is regulated without demonstrating the unwinding phenomenon.

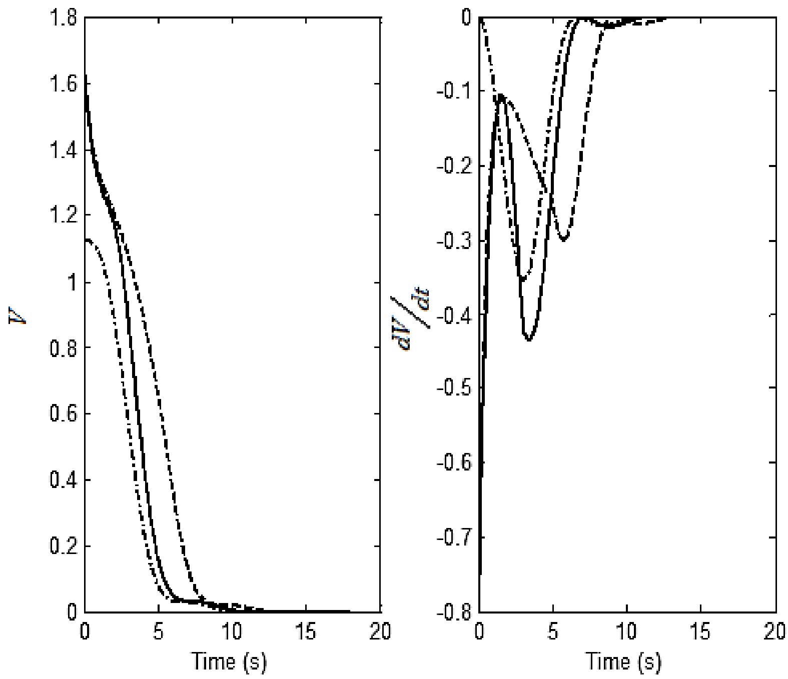
Table 1. Parameters of the Three Scenarios

	q_{dl}^{T}	$\omega_{bl}^{b\,T}(0)$
Scenario 1	$[0 \quad 1 \quad 0 \quad 0]$	$[0.5 \quad -0.5 \quad 0.5]$
Scenario 2	$[0 \quad 1 \quad 0 \quad 0]$	$[0 \quad 0 \quad 0]$
Scenario 3	$[0 \quad -1 \quad 0 \quad 0]$	$[0.5 \quad -0.5 \quad 0.5]$

Table 2. Parameters of Simulations

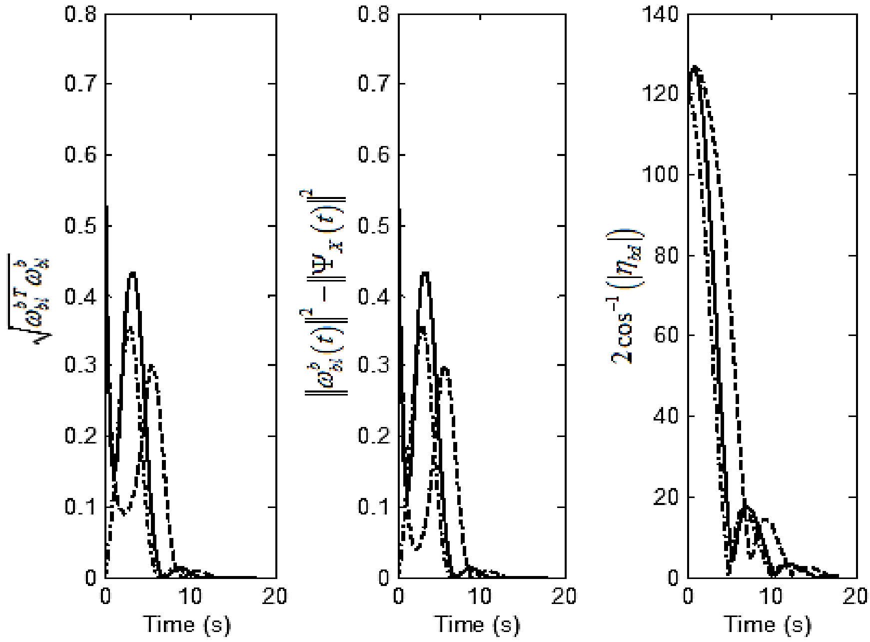
$$J = \begin{bmatrix} 1.49 & 0.054 & 0.0442 \\ 0.054 & 1.51 & 0 \\ 0.0442 & 0 & 1.56 \end{bmatrix}, \; k = 0.5, \; L = I^{3\times 3}, \bar{\phi} = 0.57$$

$$q_{bl}(0) = [-0.5 \quad 0.5 \quad -0.5 \quad 0.5]^{T}$$



(a) (b)

Figure 1. (a) Lyapunov Function and (b) Time Derivative of Lyapunov Function; Dashed line (--), dash-dotted line (-.) and solid line (-) represent Scenario 1, Scenario 2 and Scenario 3, respectively.



(a) (b) (c)

Figure 2. (a) 2-Norm of angular velocity, (b) constraint in Equation (10) and (c) Euler angle; Dashed line (--), dash-dotted line (-.) and solid line (-) represent Scenario 1, Scenario 2 and Scenario 3, respectively

6. CONCLUDING REMARKS

Proposed stability analysis for a continuous scheme of quaternion-based control system that employs saturation function has been presented. This analysis results that the designed system has asymptotically stability guarantee. To verify and observe a designed attitude control system, three scenarios of simulation are run. Simulations run show that the designed system applicable for a zero initial angular velocity case as well as a non-zero initial angular velocity case. The deadzone function Ψ_x utilization in the defined constraint shows its benefit such that the designed system is also allowed for a condition that has non-zero initial angular velocity.

ACKNOWLEDGEMENTS

First author wish to acknowledge Abdul Karim, MT, Head of Programs and Facilities Division, Satellite Technology Center LAPAN for supporting this work. Second author is supported by Penelitian Unggulan Perguruan Tinggi (PUPT) Kemenristekdikti 2017.

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BIOGRAPHIES OF AUTHORS

Harry Septanto received the BSc degree from Engineering Physics, Institut Teknologi Bandung in 2002. He received MSc degree and PhD from School of Electrical Engineering and Engineering, Institut Teknologi Bandung in 2010 and 2015, respectively. Currently, he is a researcher in Satellite Technology Center, National Institute of Aeronautics and Space (LAPAN). His current research interests include satellite attitude controller design, wheel drive electronics for satellite’s reaction wheel and software simulator development for satellite attitude control systems.

Djoko Suprijanto received his PhD form Kyushu University in 2007. Currently, he is a lecturer and member of Combinatorial Mathematics Research Division, Faculty of Mathematics and Natural Sciences, Institut Teknologi Bandung.