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Attitude and Altitude Nonlinear Control Regulation of a Quadcopter Using Quaternion Representation

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ABSTRACT Controlling a quadcopter is a challenging task because of the inherent high nonlinearity of a quadcopter system. In this paper, a new quaternion based nonlinear feedback controller for attitude and altitude regulation of a quadcopter is proposed. The dynamic model of the quadcopter is derived using Newton and Euler equations. The proposed controller is established based on a feedback linearization technique to control and regulate the quadcopter. Global asymptotic stability of the designed controller is verified using Lyapunov stability criterion. A comparison of the proposed controller performance and that of the state-of-the-art quadcopter controllers is performed to ensure the effectiveness of the proposed model. The efficiency of the proposed controller is clearly shown when the quadcopter is in or near a corner pose. Simulations are performed to assess the transient and steady state performance. Steady State Error (E_{ss}) and Max Error (E_M) are used as evaluation metrics of the proposed model performance.

INDEX TERMS Attitude and altitude regulation, feedback linearization, nonlinear control, quaternion, underactuated system.

I. INTRODUCTION

The quadcopter has attracted great interest due to its wide range of applications and capabilities such as vertical take-off and landing, hover capability, high maneuverability, and agility. In addition, the quadcopter is suitable for various applications such as military services, monitoring operation, rescue, research area, remote inspection, and photography. It is also better than standard helicopters in terms of small size, efficiency, and safety [1].

Furthermore, the design of a controller for a quadcopter is a great challenge due to the following reasons; first, the quadcopter model is a highly nonlinear, and multivariable system. Second, it has six Degrees of Freedom (DOF) but only four actuators, so it is an underactuated system [2]. Third, it requires a large region of convergence and fast control response. Fourth, some parameters of quadcopter like inertial moments and aerodynamic coefficients cannot

be obtained or estimated with high accuracy. Finally, as a result of the small size and weight of a quadcopter, a quadcopter is heavily subjected to external disturbances [3]. To solve these problems, various control algorithms were proposed to control the quadcopter. Linear control methods such as proportional-integral-derivative (PID) control and linear quadratic regulator (LQR) were applied [4], [5]. These techniques linearize the quadcopter dynamic model. The stability of these methods is only guaranteed in a restricted flights domain. Nonlinear controller designs have been developed. Considering the nonlinearity leads to a wider region of convergence range. Although the validity of nonlinear control designs is proved in controlling the quadcopter under uncertainties and external disturbance, it is still having some limitations in the region of convergence.

In this paper, the main contribution is to propose a new nonlinear feedback controller design to regulate the attitude and altitude of the quadcopter considering detailed quadcopter dynamics. The proposed controller utilizes the feedback linearization technique to establish the control law

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in which good robustness, and high regulation performance, are achieved under external disturbance. The controller is characterized by reducing the complexity of control design implementation compared to many of the mentioned controllers in related work systems and algorithms section. The gyroscopic moment is considered to obtain higher accuracy that is often ignored in the various literature. The efficiency of the proposed controller is clearly shown when the quadcopter is in or near a critical pose. To evaluate the proposed controller's efficiency and performance, it is compared to two-state-of-the-art quadcopter controllers. Despite the complexity of stability analysis for the proposed model, it satisfies the Lyapunov stability criterion. Moreover, the global asymptotic stability of the quadcopter is proved. To our best knowledge, a few previous works have achieved global asymptotic stability.

The remainder of this paper is structured as follows: Related work is presented in Section II. Section III derives the quadcopter dynamical model and defines the problem statement. In section IV, nonlinear controller design is developed, and stability analysis is performed. Simulation results and discussion are presented in section V. Finally, Section VI is devoted to conclusion and future work.

II. RELATED WORK

Linear controllers can perform only when the quadcopter is flying around nominal conditions, they suffer from a huge performance degradation whenever the quadcopter performs aggressive maneuvers [6]. Nonlinear control methods can substantially expand the domain of controllable flights compared to linear methods. A comparison has been made between linear and non-linear methods to control the UAVs in [7]. In the following, a survey of quadcopter common control techniques is summarized.

A. INTELLIGENT PID TECHNIQUES

Traditional PID does not seem appropriate for quadcopters due to the characteristics of nonlinearity, underactuated, and coupling between actuators [8]–[10]. These designs suffer from some deficiencies such as ignoring air resistance, external factors, and not being effective to compensate for Coriolis force. Many researchers have developed many designs to cover most of these deficiencies. For example, in [11], the authors proposed a hybrid model by using a PID controller and an adaptive fuzzy sliding mode for attitude stabilization, the efficiency of the proposed method tested under actuator faults. In [12], a good attitude and position tracking performance were achieved by the adaptive proportional integral derivative control (APIDC) system under external disturbances and parameter uncertainties. In [13], the authors showed how the PID controllers would be modified to suit the interference enhanced model when multiple quadcopter control problem is addressed. A Moving target tracking method is proposed in [14]. The tracking algorithm is based on the Fuzzy PI controller and is designed

to follow a moving target at different speeds and at different times of the day and night.

B. FEEDBACK LINEARIZATION TECHNIQUES

Feedback linearization is the operation of converting a nonlinear system into a linear system using a nonlinear feedback method. Recently, Developed linear feedback techniques are developed to be appropriate for the quadcopter model, but ignoring some factors like air resistance, and parameter uncertainties [15]–[17], and then further developed by many researchers. For example, in [18], the authors proposed a backstepping like feedback linearization mechanism to control and stabilize the quadcopter. Attitude controller, altitude controller, and position controller are the three sub controllers of the designed controller. In [19], a quaternion based solution to the attitude tracking problem, without velocity measurement, was proposed. The proposed control scheme includes the attitude regulation problem and guarantees roughly global asymptotic stability. The authors in [20], proposed a feedback linearization, and an adaptive sliding mode controller. FBL controller turns out to be quite sensitive to sensor noise as well as modeling uncertainty. Although an adaptive sliding mode controller could proper for the underactuated property of the quadcopter like sensor noise, and uncertainty. In [19], [20], semi-global asymptotic tracking is attained. In [21], the authors introduced a unit quaternion attitude and altitude tracking of a quadcopter, considering external disturbances and model uncertainty. The main idea in [22], is that shows how a simple strategy can be used to compensate for multiple sources of uncertainty without the use of adaptive parameter estimation or function approximators. In [23], authors presented Feedback Linearization (FL) and LQR methods, in which, they can follow a predefined trajectory, present null steady state position error, and can overcome model disturbances, imprecision, and uncertainty.

C. LINEAR QUADRATIC LQR TECHNIQUES

The LQR control technique is considered as one of the best controllers that deals with a dynamic system at the lowest cost and reduces errors. It is considered a good comparative controller due to its robustness and good performance. In the early stages, in [24], the authors proposed an LQR model with a full-order observer to control the attitude, but it needs to be more robust and effective to compensate for the Coriolis force. Then an evolution occurred in the PID controller [25], without using a sensor, an observer is proposed to estimate state variables used in the control design. Also, in [26], feedback linearization and LQR controllers were used to stabilize the quadcopter position. Feedback linearization is responsible for rectifying any occurring error. The LQR controller was combined with the feedback linearization model to improve the response of the control algorithm.

D. SLIDING MODE TECHNIQUES

Solving uncertain nonlinear system control is an advantage of the sliding mode. For example, the proposed model in [27]

can achieve the determined position and yaw angle and regulate pitch and roll angle at zero, but it is less sensitive to the changes in quadcopter parameters. In [28], the authors improved the robustness of the system to the changes that resulted from external disturbances and model uncertainty. Moreover, the controller does not need to resort to high power gain and can quickly compensate for changes in external disturbances. In [29], the authors introduced feedback linearization based adaptive sliding mode control, which is effective against the ground effects of the quadcopter. The asymptotic stability of the overall system was assured. Also, experimental results prove the robustness and the stability of the control model. An adaptive fuzzy gains-scheduled integral sliding mode control approach is suggested and successfully applied for a quadcopter with a low chattering phenomenon effect [30]. The authors in [31], proposed a new sliding mode controller to address the quadcopter path in the presence of uncertainties and random disturbances. Also, they used a novel time varying sliding mode surface to eliminate the reaching phase, reduce the initial control effort, and meet the impact time requirement with a global sliding mode. In [32], sliding mode control was proposed based on the neural networks to stabilize the position and altitude of the quadcopter under external perturbations.

E. BACKSTEPPING DESIGN TECHNIQUES

The main idea of the backstepping design is to model the dynamic system in several steps while designing the control law. These techniques suffer from some deficiencies such as low system robustness, but other methods can be applied to overcome these deficiencies. For example, in [33], the authors proposed a backstepping control algorithm, which can stabilize a quadcopter to the required trajectory. The simulation results showed a good performance of the proposed control approach. In [34], a sequential nonlinear technique controller was used to stabilize a quadcopter. The controller was established using feedback linearization combined with a PD controller for the translational subsystem, and a backstepping based PID nonlinear controller for the rotational subsystem of the quadcopter. A good performance of the proposed controller is demonstrated in semi-stationary flights by simulation results. In [35], the authors presented an underactuated quadcopter system with model parameter uncertainty. In [36], a backstepping controller is proposed to control the altitude and attitude of quadcopter systems. The validity of the controller is ensured by the Lyapunov function, and simulation results showed a high precision transient and tracking response. In [37], A backstepping controller for quadcopter tracking based on a neural network approach has been proposed although uncertainties arise.

III. PROBLEM STATEMENT AND DYNAMIC MODEL

Quadcopters are flying vehicles with four propellers placed at equal distances from the vehicle center. Each propeller induces a normal force F_i and a moment M_i around the normal direction. The vehicle motion is referenced to a fixed

inertial ground frame with axes $XI-YI-ZI$. A body frame $X-Y-Z$ is attached to the vehicle body at its center, as shown in Figure.1. Let the vehicle body frame origin vector be $\mathbf{r} = [X_I \ Y_I \ Z_I]^T$, vehicle velocity be $\dot{\mathbf{r}}$, acceleration be $\ddot{\mathbf{r}}$, vehicle rotation matrix with respect to ground frame be R , and vehicle angular velocity is $\boldsymbol{\Omega}$.

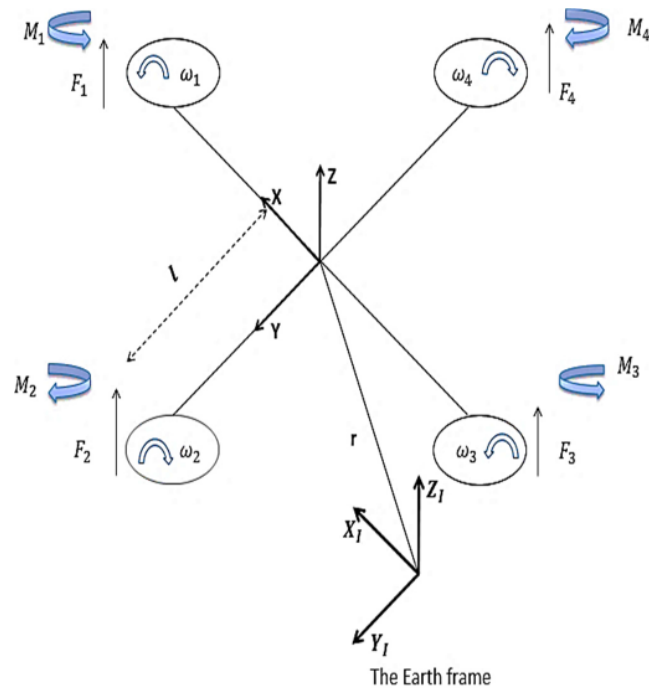


FIGURE 1. Quadcopter system model.

The motion of the quadcopter can be described using the Euler torque and Newton force equations. Each propeller rotating with angular speed ω_i is subjected to an aerodynamic normal force F_i that is given by $F_i = k_f \omega_i^2$ and an aerodynamic moment in the reverse direction of its rotation that is given by $M_i = k_m \omega_i^2$. k_f and k_m are propeller specific proportionality constants. Each propeller applies a torque \mathbf{M}_{pi} for $i \in \{1, 2, 3, 4\}$ on the vehicle body around the normal direction. The torque \mathbf{M}_{pi} can be expressed in vehicle body frame as follows:

$$\mathbf{M}_{pi} = \pm \left(M_i \begin{bmatrix} 0 \\ 0 \\ 1 \end{bmatrix} + \boldsymbol{\Omega} \times \mathbf{J}_r \begin{bmatrix} 0 \\ 0 \\ \omega_i \end{bmatrix} \right). \tag{1}$$

The second term in the RHS represents the gyroscopic moment contribution to propeller torque [38]. The propeller torque is positive or negative based on the sense of propeller rotation. The rotational motion of the quadcopter body can be described in body frame by using Euler equation as follows:

$$\mathbf{J}\dot{\boldsymbol{\Omega}} + \boldsymbol{\Omega} \times \mathbf{J}\boldsymbol{\Omega} = \sum_{i=1}^4 \mathbf{M}_{pi} + \mathbf{F}_i \times \mathbf{l}_i \tag{2}$$

where \mathbf{F}_i is the normal force exerted by propeller i located at position \mathbf{l}_i from the quadcopter body center, and $\mathbf{J} \in \mathbf{R}^{3 \times 3}$ is the symmetric positive definite inertia matrix of the vehicle.

The previous expression can be further simplified as:

$$\mathbf{J}\dot{\boldsymbol{\Omega}} + \boldsymbol{\Omega} \times \mathbf{J}\boldsymbol{\Omega} + \mathbf{J}_r\boldsymbol{\Omega} \times \begin{bmatrix} 0 \\ 0 \\ \omega_r \end{bmatrix} = \mathbf{M}_B \quad (3)$$

Eq. (3) represents the quadcopter rotational dynamics expressed in the body frame, where $\omega_r = \omega_1 - \omega_2 + \omega_3 - \omega_4$, \mathbf{J}_r is the rotors' inertia, and $\mathbf{M}_B = [M_x \ M_y \ M_z]^T$ is the total torque applied on the quadcopter body. It can be shown as:

$$M_x = F_2l - F_4l = lk_f(\omega_2^2 - \omega_4^2) \quad (4)$$

$$M_y = F_3l - F_1l = lk_f(\omega_3^2 - \omega_1^2) \quad (5)$$

$$M_z = M_1 - M_2 + M_3 - M_4 = k_m(\omega_1^2 - \omega_2^2 + \omega_3^2 - \omega_4^2) \quad (6)$$

The translation equations of motion for the quadcopter are based on Newton's second law. It can be shown as:

$$m\ddot{\mathbf{r}} = mg \begin{bmatrix} 0 \\ 0 \\ 1 \end{bmatrix} + \mathbf{R}\mathbf{F}_b \quad (7)$$

where m is the quadcopter's mass, g is the gravitational acceleration $g = 9.8 \text{ m/s}^2$, and $\mathbf{R} \in \mathbf{SO}(3)$ is the rotation matrix that describes the orientation of the vehicle body. \mathbf{F}_b represents aerodynamic thrust forces acting on the vehicle body expressed in body frame, as shown in Eq. (8).

$$\mathbf{F}_b = \begin{bmatrix} 0 \\ 0 \\ F_b \end{bmatrix} = \begin{bmatrix} 0 \\ 0 \\ k_f(\omega_1^2 + \omega_2^2 + \omega_3^2 + \omega_4^2) \end{bmatrix} \quad (8)$$

Assuming the propeller speeds can be instantaneously controlled, and the system has four inputs M_x , M_y , M_z , and F_b . The input vector is defined as $\mathbf{U} = [M_x \ M_y \ M_z \ F_b]^T = [u_1 \ u_2 \ u_3 \ u_4]^T$.

IV. NONLINEAR CONTROLLER DESIGN AND STABILITY ANALYSIS

The orientation of a rigid body with respect to an inertial frame can be represented using several representations. Rotation matrices, Euler angles, and quaternions are examples of such representations. Rotation matrix representation utilizes nine parameters; hence, it overparameterizes the orientation. Orthogonality constraints must be maintained over the rotation matrix parameters, leading to a system of complex nonlinear equations. Euler angle representation provides a minimum orientation representation with only three parameters; hence, it is a popular choice for representing orientation in controller design. However, Euler angle representation has some drawbacks. Euler angle representation has singular configurations, in which the angular velocity loses one degree of freedom. These drawbacks can be avoided using a fourparameter representation, namely, unit quaternions. For a body frame rotation about a unit axis \mathbf{n} with an angle $-\pi < \theta < \pi$, the rotation can be described using a unit quaternion defined by [39] as shown in Eq.(9).

$$\mathbf{q} = \begin{bmatrix} \cos(\theta/2) \\ \mathbf{n} \sin(\theta/2) \end{bmatrix} = \begin{bmatrix} q_0 \\ \mathbf{q}_v \end{bmatrix} \quad (9)$$

In this section, a nonlinear stabilizing controller, which includes attitude and altitude regulation control, of the quadcopter is developed. The main objective of the attitude regulation controller is to design control inputs u_1 , u_2 , and u_3 to ensure that the desired orientation quaternion is equal to $\mathbf{q} = [1 \ 0 \ 0 \ 0]^T$. The altitude regulation controller specifies the control input u_4 to ensure that the quadcopter altitude is zero. Stability analysis of the designed controller is verified by the Lyapunov stability theorem.

A. ATTITUDE CONTROLLER

Nowadays, attitude tracking has various applications. For example, quadcopters are now commonly used to photograph landscapes. The quadcopter's roll pitch Euler angle has a significant impact on the final images, thus it must be controlled with high quality performance [40]. So, the attitude stability was met with great interest which is clearly shown in flexible flight movement [3]. The attitude control of the quadcopter can be designed independently from altitude, the system state space is non-Euclidean. The attitude model of a quadcopter is represented by two equations. The first equation is the orientation matrix derivative equation given by Eq. (10).

$$\dot{\mathbf{R}} = \mathbf{R}s(\boldsymbol{\Omega}) \quad (10)$$

where $s(\boldsymbol{\Omega})$ is a skew-symmetric matrix of $\boldsymbol{\Omega}$. Instead of using the rotation matrix \mathbf{R} to describe the orientation of the rigid body, the unit quaternion is used. The last equation can be replaced by the following dynamic equation in terms of the unit quaternion.

$$\dot{\mathbf{q}} = \mathbf{q} \otimes \boldsymbol{\omega} \quad (11)$$

where $\boldsymbol{\omega} = \begin{bmatrix} 0 \\ \boldsymbol{\Omega}/2 \end{bmatrix}$ and the operator \otimes is the quaternion multiplication operator. Eq. (11) with Eq. (3) define the rotational dynamics of the quadcopter. The equilibrium point $\mathbf{R} = \mathbf{I}$ is equivalent to the two equilibrium points $[\pm 1 \ 0 \ 0 \ 0]^T$, because $q_0 = 1$ corresponds to $\theta = 0$ and $q_0 = -1$ corresponds to $\theta = 2\pi$.

Variables of the quadcopter system change from $\boldsymbol{\Omega}$ and \mathbf{q} to $\boldsymbol{\Omega}$ and \mathbf{x}_1 is performed, where \mathbf{x}_1 is defined as:

$$\mathbf{x}_1 = \boldsymbol{\Omega} + k_1\mathbf{q}_v = \boldsymbol{\Omega} + k_1\mathbf{M} \quad (12)$$

where $\mathbf{M} = \begin{bmatrix} 0 & 1 & 0 & 0 \\ 0 & 0 & 1 & 0 \\ 0 & 0 & 0 & 1 \end{bmatrix}$, and k_1 is a suitable scalar controller parameter. The state equation of \mathbf{x}_1 is given by:

$$\mathbf{J}\dot{\mathbf{x}}_1 = \mathbf{J}\dot{\boldsymbol{\Omega}} + k_1\mathbf{J}\mathbf{M}\dot{\mathbf{q}} \quad (13)$$

By substituting Eqs. (3) and (11) into Eq. (13) to obtain the following equation:

$$\mathbf{J}\dot{\mathbf{x}}_1 = -\boldsymbol{\Omega} \times \mathbf{J}\boldsymbol{\Omega} - \mathbf{J}_r\boldsymbol{\Omega} \times \begin{bmatrix} 0 \\ 0 \\ \omega_r \end{bmatrix} + \mathbf{M}_B + k_1\mathbf{J}\mathbf{M}\mathbf{q} \otimes \boldsymbol{\omega} \quad (14)$$

Eq. (14) with Eq. (3) define the quadcopter rotational dynamics in the state space of Ω and \mathbf{x}_1 . The control law is to be defined by:

$$\mathbf{M}_B = -k_1 \mathbf{J} \mathbf{M} \mathbf{q} \otimes \mathbf{q}_\Omega - k_2 \mathbf{x}_1 + \Omega \times \mathbf{J} \Omega + \mathbf{J}_r \Omega \times \begin{bmatrix} 0 \\ 0 \\ \omega_r \end{bmatrix} \quad (15)$$

By substituting Eq. (15) into Eqs. (14), and (3) the following equations are obtained:

$$\mathbf{J} \dot{\mathbf{x}}_1 = -k_2 \mathbf{x}_1 \quad (16)$$

$$\mathbf{J} \dot{\Omega} = -k_2 \mathbf{x}_1 - k_1 \mathbf{J} \mathbf{M} \mathbf{q} \otimes \mathbf{q}_\Omega \quad (17)$$

The system described by Eqs. (16), and (17) has an equilibrium point at $\mathbf{x}_1 = \mathbf{0}$ and $\Omega = \mathbf{0}$, which correspond to $\mathbf{q} = [\pm 1 \ \mathbf{0} \ \mathbf{0} \ \mathbf{0}]^T$ and $\Omega = \mathbf{0}$. With a proper choice of k_1 and k_2 , the system described by Eqs. (16) and (17) is shown to be stable.

B. PROOF OF ATTITUDE CONTROLLER STABILITY

To prove the validity of this controller, the Lyapunov stability theorem is used. The Lyapunov function $V(\mathbf{x}_1, \Omega)$ is defined as follows:

$$\begin{aligned} V(\mathbf{x}_1, \Omega) &= 0.5 (\mathbf{M} \mathbf{q})^T \mathbf{M} \mathbf{q} + 0.5 \mathbf{x}_1^T \mathbf{J} \mathbf{x}_1 \\ &= \frac{1}{k_1^2} 0.5 (\mathbf{x}_1 - \Omega)^T (\mathbf{x}_1 - \Omega) + 0.5 \mathbf{x}_1^T \mathbf{J} \mathbf{x}_1 \end{aligned} \quad (18)$$

The Lyapunov function $V(\mathbf{x}_1, \Omega)$ is positive definite. $V(\mathbf{x}_1, \Omega) = 0$ if and only if $\mathbf{x}_1 = \mathbf{0}$ and $\Omega = \mathbf{0}$. The time derivative of $V(\mathbf{x}_1, \Omega)$ is given by:

$$\dot{V}(\mathbf{x}_1, \Omega) = \frac{1}{k_1^2} (\mathbf{x}_1 - \Omega)^T (\dot{\mathbf{x}}_1 - \dot{\Omega}) + \mathbf{x}_1^T \mathbf{J} \dot{\mathbf{x}}_1 \quad (19)$$

Substituting for \dot{x}_1 And $\dot{\Omega}$ From Eqs (16), (17)

$$\dot{V}(\mathbf{x}_1, \Omega) = \frac{1}{k_1^2} (\mathbf{x}_1 - \Omega)^T (k_1 \mathbf{M} \mathbf{q} \otimes \mathbf{q}_\Omega) - k_2 \mathbf{x}_1^T \mathbf{x}_1 \quad (20)$$

Eq. (20) can be simplified as:

$$\dot{V}(\mathbf{x}_1, \Omega) = (\mathbf{M} \mathbf{q})^T (\mathbf{M} \mathbf{q} \otimes \mathbf{q}_\Omega) - k_2 \mathbf{x}_1^T \mathbf{x}_1 \quad (21)$$

Substituting for \mathbf{q} and \mathbf{q}_Ω :

$$\dot{V}(\mathbf{x}_1, \Omega) = \mathbf{q}_v^T \mathbf{M} \left(\begin{bmatrix} q_0 \\ \mathbf{q}_v \end{bmatrix} \otimes \begin{bmatrix} 0 \\ \Omega/2 \end{bmatrix} \right) - k_2 \mathbf{x}_1^T \mathbf{x}_1 \quad (22)$$

Applying the quaternion multiplication rule:

$$\dot{V}(\mathbf{x}_1, \Omega) = \frac{1}{2} \mathbf{q}_v^T \mathbf{M} \left(\begin{bmatrix} -\mathbf{q}_v^T \Omega \\ q_0 \Omega + \mathbf{q}_v \times \Omega \end{bmatrix} \right) - k_2 \mathbf{x}_1^T \mathbf{x}_1 \quad (23)$$

Simplifying, it can be shown that:

$$\dot{V}(\mathbf{x}_1, \Omega) = \frac{1}{2} \mathbf{q}_v^T (q_0 \Omega + \mathbf{q}_v \times \Omega) - k_2 \mathbf{x}_1^T \mathbf{x}_1 \quad (24)$$

Further simplification leads to:

$$\dot{V}(\mathbf{x}_1, \Omega) = \frac{q_0}{2} \mathbf{q}_v^T \Omega - k_2 \mathbf{x}_1^T \mathbf{x}_1 \quad (25)$$

Let $\Omega = \Omega_p + \Omega_n$, where Ω_p is the component of Ω that is parallel to the vector \mathbf{q}_v , and Ω_n is the component of Ω that is normal to \mathbf{q}_v . Control parameters k_1 and k_2 are assumed to be positive constants. Therefore, Ω_p and Ω_n can be defined as follows:

$$\Omega_p = \alpha_1 k_1 \mathbf{q}_v \quad (26)$$

$$\Omega_n = \Omega - \Omega_p \quad (27)$$

where α_1 is a positive constant. Substituting by Ω_p and Ω_n in Eq (25):

$$\dot{V}(\mathbf{x}_1, \Omega) = 0.5 \alpha_1 k_1 q_0 \mathbf{q}_v^T \mathbf{q}_v - k_2 \mathbf{x}_1^T \mathbf{x}_1 \quad (28)$$

Substituting for \mathbf{x}_1 :

$$\dot{V}(\mathbf{x}_1, \Omega) = 0.5 \alpha_1 k_1 q_0 \mathbf{q}_v^T \mathbf{q}_v - k_2 (\Omega + k_1 \mathbf{q}_v)^T (\Omega + k_1 \mathbf{q}_v) \quad (29)$$

To simplify Eq. (29), the angular velocity Ω is decomposed into its parallel and normal components, as follows:

$$\begin{aligned} (\Omega + k_1 \mathbf{q}_v)^T (\Omega + k_1 \mathbf{q}_v) &= (\Omega_n + \Omega_p + k_1 \mathbf{q}_v)^T (\Omega_n + \Omega_p + k_1 \mathbf{q}_v) \end{aligned}$$

\mathbf{q}_v and Ω_n are normal, therefore

$$(\Omega + k_1 \mathbf{q}_v)^T (\Omega + k_1 \mathbf{q}_v) = \Omega_n^T \Omega_n + (\alpha_1 + 1)^2 k_1^2 \mathbf{q}_v^T \mathbf{q}_v$$

Substituting by the last equation into Eq. (29), it can be shown that:

$$\dot{V}(\mathbf{x}, \Omega) = 0.5 \alpha_1 k_1 q_0 \mathbf{q}_v^T \mathbf{q}_v - k_2 \Omega_n^T \Omega_n - k_2 (\alpha_1 + 1)^2 k_1^2 \mathbf{q}_v^T \mathbf{q}_v \quad (30)$$

which can be simplified as follows:

$$\dot{V}(\mathbf{x}, \Omega) = -k_1 \left(k_1 k_2 (\alpha_1 + 1)^2 - 0.5 \alpha_1 q_0 \right) \mathbf{q}_v^T \mathbf{q}_v - k_2 \Omega_n^T \Omega_n \quad (31)$$

$\dot{V}(\mathbf{x}, \Omega)$ is desired to be negative definite to satisfy the Lyapunov stability criterion and controller parameters k_1, k_2 are assumed to be always positive; hence, $\dot{V}(\mathbf{x}, \Omega)$ is positive definite if $k_1 k_2 (\alpha_1 + 1)^2 - 0.5 \alpha_1 q_0 > 0$, which can be simplified as follows:

$$\alpha_1^2 + \left(2 - \frac{q_0}{2k_1 k_2} \right) \alpha_1 + 1 \quad (32)$$

The discriminator of this equation is given by:

$$D = \left(1 - \frac{q_0}{4k_1 k_2} \right)^2 \quad (33)$$

q_0 is constrained to be positive; hence, if $0 < \frac{q_0}{4k_1 k_2} < 2$, the value of D is less than zero ($D < 0$), which means the condition stated by Eq. (32) is true for any value of α_1 . Given that $q_0 < 1$, the system can be stabilized for all values of \mathbf{q} and Ω by imposing the condition $k_1 k_2 > \frac{1}{8}$. Finally, it is shown that:

$$\dot{V}(\mathbf{x}_1, \Omega) = -k_{12} \mathbf{q}_v^T \mathbf{q}_v - k_2 \Omega_n^T \Omega_n \quad (34)$$

where the constant $k_{12} = k_1(k_1k_2(\alpha_1 + 1)^2 - 0.5\alpha_1q_0)$ is positive whenever the constants k_1 and k_2 satisfies the condition $k_1k_2 > \frac{1}{8}$; hence, $\dot{V}(\mathbf{x}, \boldsymbol{\Omega}) \leq 0$. It can be also shown that $\dot{V}(\mathbf{x}, \boldsymbol{\Omega}) = 0$ only if $\mathbf{x} = 0$ and $\boldsymbol{\Omega} = 0$. These are sufficient conditions for global asymptotic stability.

C. ALTITUDE CONTROLLER

The dynamic equation for quadcopter altitude is derived from Eq. (8). We are interested only in controlling the altitude z ; thus, the equation of quadcopter altitude dynamics is:

$$m\ddot{z} = -mg + r_{33}F_b \quad (35)$$

where r_{33} is the last term of the rotation matrix \mathbf{R} . The feedback linearization technique can be used to linearize Eq. (35) by proposing the control law $F_b = \frac{mg+v}{r_{33}}$, where v is an introduced control input. This substitution is valid whenever $r_{33} \neq 0$. Altitude dynamics can be written in the modified form as:

$$m\ddot{z} = v \quad (36)$$

Eq. (36) represents an ordinary second-order linear system with control input v . A PD controller can be used to stabilize such a system:

$$v = -k_3z - k_4\dot{z} \quad (37)$$

The system described by Eq. (37) can be stabilized by a suitable choice of k_3 and k_4 . The complete control law is given by:

$$F_b = \frac{mg - k_3z - k_4\dot{z}}{r_{33}} \quad (38)$$

V. SIMULATION RESULTS AND DISCUSSION

This section presents a comparison study between the proposed controller and two candidates state-of-the-art controllers (Non linear1, Non linear2) that were presented in [18], [19] respectively. A case study is conducted on a quadcopter with physical parameters provided in [41]. Hence the initial conditions and nominal parameters have been presented in Table 1. Simulations are conducted on a PC with 6.00 GB RAM, 64-bit operating system, Intel(R) Core (TM) i5-2410M CPU @2.30 GHz. A MATLAB–Simscape multibody toolbox is used to model the simulation case [42].

The equation of the first controller (Non linear1) is shown as follows \mathbf{M}_B and F_b , as shown at the bottom of the page,

TABLE 1. Quadcopter parameters and constant.

Parameter	Description	Value
J_x	MOI about x-axis	$7.5e-3 \text{ kg.m}^2$
J_y	MOI about y-axis	$7.5e-3 \text{ kg.m}^2$
J_z	MOI about z-axis	$1.3e-3 \text{ kg.m}^2$
J_r	Inertia of motor	$6e-5 \text{ kg.m}^2$
l	Moment arm	0.23 m
m	Quadcopter mass	0.65 kg
k_f	Thrust coefficient	$3.13e-5 \text{ N s}^2$
k_M	Moment coefficient	$7.5e-7 \text{ N ms}^2$
\mathbf{k}_r	Rotation drag coefficient	$\text{diag}(0.1, 0.1, 0.15) \text{ Nm. s}$
\mathbf{k}_t	Translation drag coefficient	$\text{diag}(0.1, 0.1, 0.15) \text{ Ns/m}$
g	gravitational acceleration	9.81 m/s^2

where (ϕ, θ, ψ) roll, pitch, and yaw angles denote rotations about the X, Y, and Z axes, respectively, to characterize the quadcopter's orientation, and $c_1 \sim c_8$ are positive constants. $e_1 \sim e_8$ define errors associated with the attitude, and altitude dynamics that calculated as in the following equations:

$$\begin{aligned} e_1 &= \phi - \phi_d \\ e_2 &= \dot{\phi} - \dot{\phi}_d + c_1 e_1 \\ e_3 &= \theta - \theta_d \\ e_4 &= \dot{\theta} - \dot{\theta}_d + c_3 e_3 \\ e_5 &= \psi - \psi_d \\ e_6 &= \dot{\psi} - \dot{\psi}_d + c_5 e_5 \\ e_7 &= z - z_d \\ e_8 &= \dot{z} - \dot{z}_d + c_7 e_7 \end{aligned}$$

$(\dot{\phi}_d, \dot{\theta}_d, \dot{\psi}_d, \dot{z}_d)$ are equal to zero in the case of regulation controller.

The equation of the second controller (Non linear2) is shown as follows:

$$\mathbf{M}_B = -\alpha_1 \mathbf{q}^e - \alpha_2 \tilde{\mathbf{q}}$$

where α_1 , and α_2 are positive constants. The vectors \mathbf{q}^e and $\tilde{\mathbf{q}}$ are the vector parts of the unit quaternion \mathbf{Q}^e and $\tilde{\mathbf{Q}}$. \mathbf{Q}^e is the unit-quaternion tracking error, which describes the discrepancy between the actual unit-quaternion

$$\begin{aligned} \mathbf{M}_B &= \begin{bmatrix} \frac{J_x}{I}(-\dot{\theta}\dot{\psi}\left(\frac{J_y-J_z}{J_x}\right) + \frac{J_x}{J_x}\dot{\theta}\Omega + (c_1^2 - 1)e_1 - (c_1 + c_2)e_2) \\ \frac{J_y}{I}(-\dot{\phi}\dot{\psi}\left(\frac{J_z-J_x}{J_y}\right) - \frac{J_x}{J_y}\dot{\phi}\Omega + (c_3^2 - 1)e_3 - (c_3 + c_4)e_4) \\ \frac{J_x}{I}(-\dot{\phi}\dot{\theta}\left(\frac{J_y-J_z}{J_x}\right) + (c_5^2 - 1)e_5 - (c_5 + c_6)e_6) \end{bmatrix} \\ F_b &= \frac{m(g + (c_7^2 - 1)e_7 - (c_7 + c_8)e_8)}{\cos(\phi)\cos(\theta)} \end{aligned}$$

Q and the desired unit-quaternion Q^d . Q^{\sim} describe the discrepancy between the unit-quaternion tracking error Q^e and the auxiliary feedback unit-quaternion signal Q^- . Q^- is the output of the auxiliary system in this model.

Non linear1 controller uses Euler angle representation that has singular configurations, in which the angular velocity loses one degree of freedom that is called gimbal lock problem. These drawbacks can be avoided using unit quaternions that have proven good performance in comparison to the other representations. The efficiency of the Nonlinear 1 controller is noticeably less when the quadcopter is in or near a critical angle.

Non linear2 controller takes into consideration attitude tracking problem but doesn't consider the altitude tracking problem. Also, the gyroscopic moment is not included in this controller.

A. DYNAMIC RESPONSE

An initial disturbance from the equilibrium point is set in the simulated model, and the system's response to the disturbance is studied for each of the three controllers. Eight cases are studied: (1) response to an initial disturbance that has $\theta_z(t = 0) = 20^\circ$; (2) response to an initial disturbance with $\theta_z(t = 0) = 50^\circ$; (3) response to an initial disturbance with $\theta_x(t = 0) = 20^\circ$; (4) response to an initial disturbance with $\theta_x(t = 0) = 50^\circ$; (5) response to an initial disturbance with $z(t = 0) = 1$; (6) response to an initial disturbance with $\theta_z(t = 0) = 50^\circ$, and $z = 1$; (7) response to an initial disturbance with $\theta_x(t = 0) = 85^\circ$; and (8) response to an initial disturbance with $\theta_x(t = 0) = 50^\circ, \theta_y(t = 0) = 20, \theta_z(t = 50^\circ) = 50^\circ$, and $z(t = 0) = 1$. The quadcopter model has an initial angular velocity $\Omega_z(t = 0) = 100^\circ/s$ for all eight cases. The simulation is performed with MATLAB–Simscape for a time span of 10s. We applied the control of attitude and altitude with a constant ($k_1 = k_2 = k_3 = k_4 = 4$). The orientation angle is represented in the simulation model by axis angle representation. To assess the efficiency and performance of the considered controllers, simulation results are provided, followed by comparisons between the various controllers.

Simulations in Figures. 2 and 3 represent Cases (1) and (2) for $\theta_z = 20^\circ$ and 50° . The transient performance of the proposed closed-loop system with respect to the altitude is the lowest overshoot for $\theta_z = 20^\circ$, and $\theta_z = 50^\circ$, compared with Non linear1 candidate controller. For both cases, Non linear2 candidate controller has the longest response time of the orientation angle. No change happens to the orientation angle around the x- and y-axes.

Figures. 4, and 5 represent Cases (3) and (4) for $\theta_x = 20^\circ, 50^\circ$, respectively. For both cases, Non linear1 candidate controller has high overshoot than the proposed controller in altitude. However, the θ_x disturbance induces transient responses in θ_y and θ_z , in contrast to Cases (1) and (2) response.

Figures. 6, and 7 represent Cases (5) and (6) for ($z = 1$) and ($\theta_z = 70^\circ$, and $z = 1$), respectively.

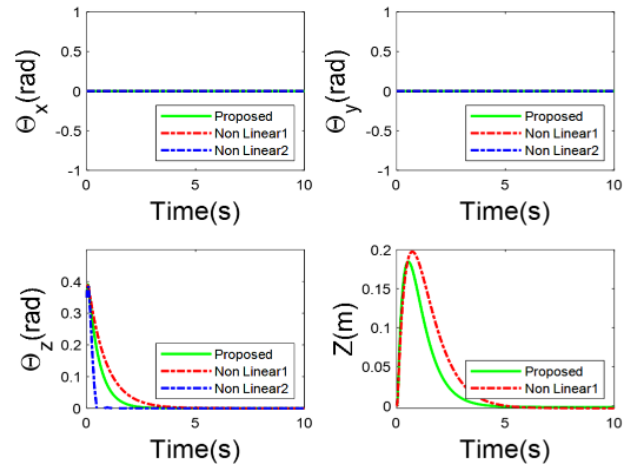


FIGURE 2. Transient response for case (1) in response to an initial condition on $\theta_z = 20$ for the proposed system and the controllers of [18], [19].

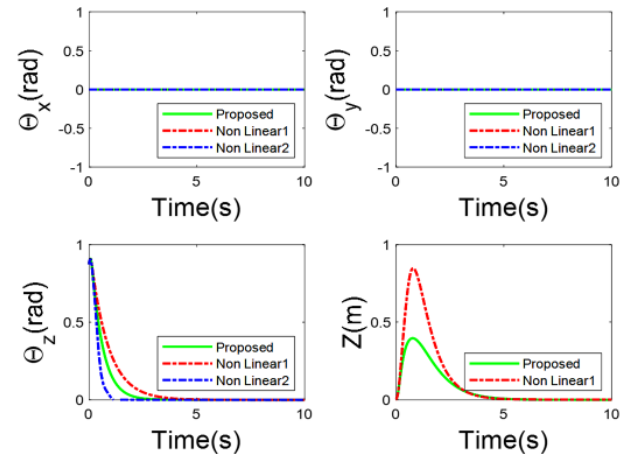


FIGURE 3. Transient response for case (2) in response to an initial condition $\theta_z = 50$ for the proposed system and the controllers of [18], [19].

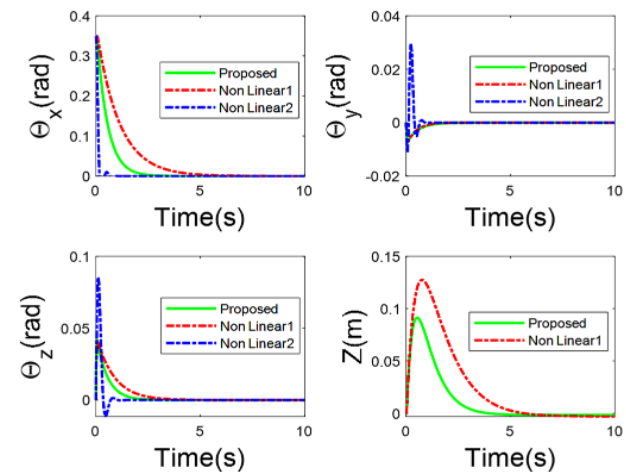


FIGURE 4. Transient response for case (3) in response to an initial condition $\theta_x = 20$ for the proposed system and the controllers of [18], [19].

The proposed controller is the best with respect to the altitude response, but Non linear2 candidate controller has

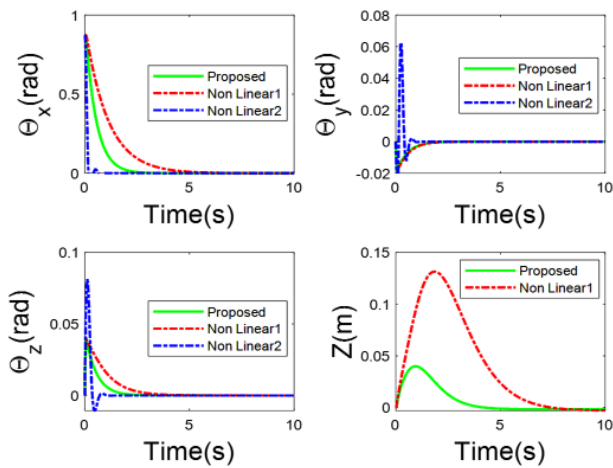


FIGURE 5. Transient response for case(4) in response to an initial condition $\theta_x = 50$ for the proposed system and the controllers of [18], [19].

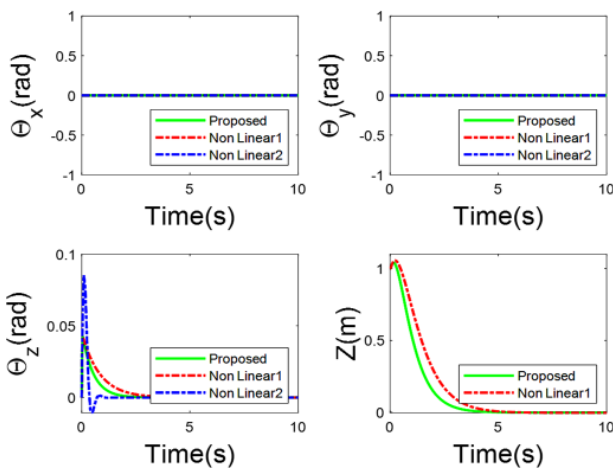


FIGURE 6. Transient response for case (5) in response to an initial condition $z = 1$ for the proposed system and the controllers of [18], [19].

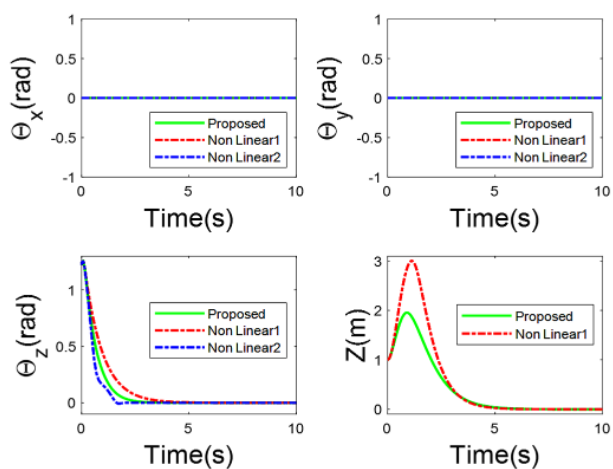


FIGURE 7. Transient response for case (6) in response to an initial condition ($z = 1, \theta_z = 70^\circ$) for the proposed system and the controllers of [18], [19].

a substantial disturbance in the orientation angle around the z-axis θ_z . No change happens to the orientation angle around the x- and y-axes.

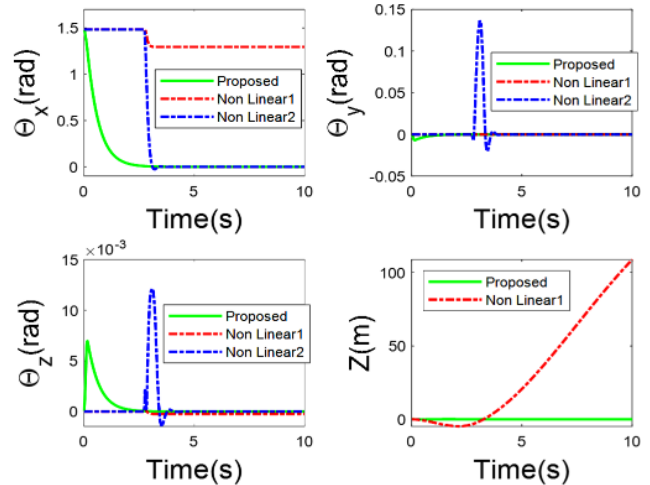


FIGURE 8. Transient response for case (7) in response to an initial condition ($\theta_x = 85$) for the proposed system and the controllers of [18], [19].

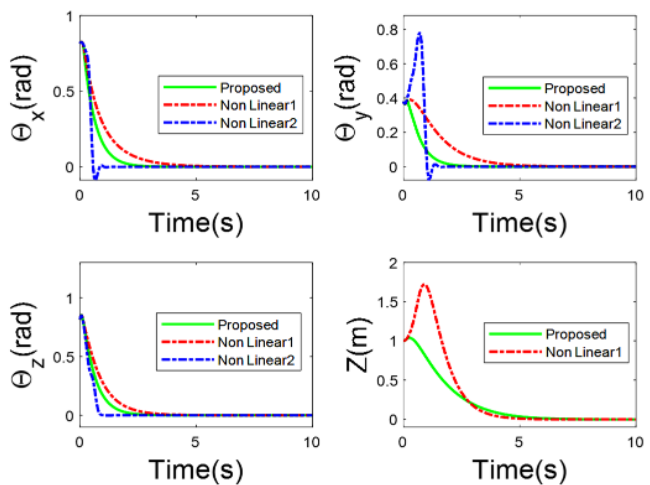


FIGURE 9. Transient response for case (8) in response to an initial condition ($\theta_x = 50^\circ, \theta_y = 20^\circ, \theta_z = 50^\circ$, and $z = 1$) for the proposed system and the controllers of [18], [19].

Figures. 8, and 9 represent case (7), and case (8) responses for ($\theta_x = 85^\circ$), and ($\theta_x = 50^\circ, \theta_y = 50^\circ, \theta_z = 50^\circ$, and $z = 1$) respectively. The proposed controller response is the best response in case (7), and case (8). Non linear2 candidate controller suffers from an additional disturbance in the orientation around Y-axis. The proposed controller has the highest response time to altitude in case (7), and case (8).

B. RESPONSE TO VARIOUS INPUTS

Several simulation experiments are performed to assess the effectiveness of the proposed controller. Various inputs are applied to show the steady-state error and compare the steady state error of the proposed controller with the two candidate controllers. Initial conditions are taken as follows: ($q_0 = \pm 1, q = 0, \theta = 0$). Three different inputs (step, ramp, and parabolic inputs) are used to check the ability of the three controllers to track the attitude and altitude.

TABLE 2. Steady-state error for the step input and Max error for ramp, and parabolic input.

Input	controller	$E(\theta_x)$	$E(\theta_y)$	$E(\theta_z)$	$E(z)$
Step	Proposed	5e-4	1.1e-4	0	0.001
	Nonlinear1	0.001	0.001	0.001	0.003
	Nonlinear2	0.002	0.003	0.002	Na
Ramp	Proposed	0.002	0.002	0.003	1.05
	Nonlinear1	0.0259	0.0252	0.019	1.39
	Nonlinear2	0.0242	0.0242	0.0248	Na
Para	Proposed	0.024	0.021	0.034	9.70
	Nonlinear1	0.244	0.203	0.196	12.48
	Nonlinear2	0.218	0.217	0.222	Na

1) STEP INPUT

Step input is applied to the attitude $(\theta_x, \theta_y, \theta_z)$, and to the altitude z . The desired attitude and altitude response $(\theta_{dx}, \theta_{dy}, \theta_{dz}, z_d)$ for step input are shown in Figure 10. Transient response for θ_{dx} , θ_{dy} , and θ_{dz} for the three controllers are almost identical. E_{ss} is used to show the efficiency of the proposed controller with the state-of-the-art controllers.

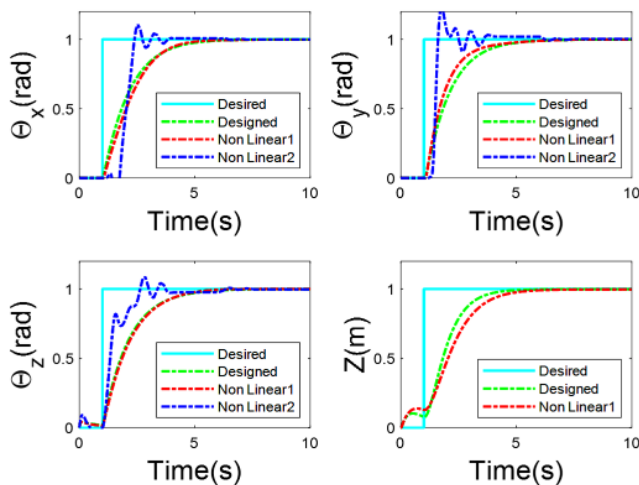


FIGURE 10. Transient response of step input for the proposed system and the controllers of [18], [19].

The three controllers have a low steady state error (almost zero) as shown in Table 2, but Non linear2 candidate controller suffers from some disturbance. For the desired altitude (z_d) the proposed controller and the first controller are almost identical. The error of the step input is almost zero because the step input is equal to regulate the quadcopter.

2) RAMP INPUT

Ramp input is applied to the attitude $(\theta_x, \theta_y, \theta_z)$, and to the altitude z . The desired attitude and altitude response $(\theta_{dx}, \theta_{dy}, \theta_{dz}, z_d)$ for ramp input are shown in Figure. 11.

E_M is used to show the efficiency of the proposed controller in relation to the state-of-the-art controllers.

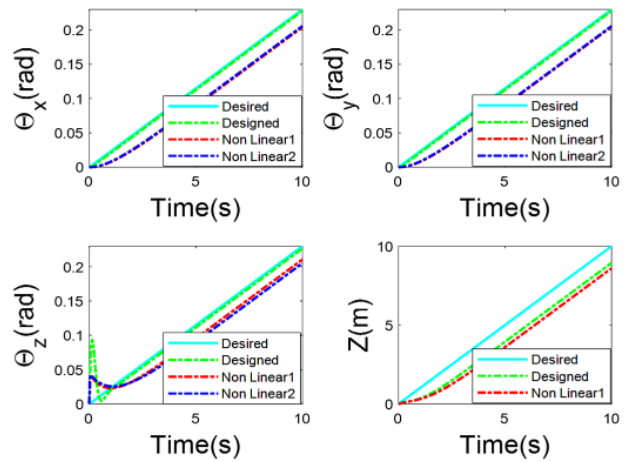


FIGURE 11. Transient response of the ramp input for the proposed system and the controllers of [18], [19].

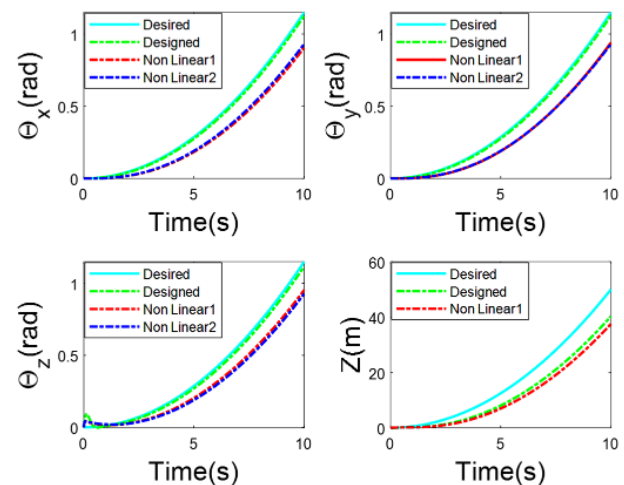


FIGURE 12. Transient response of parabolic input for the proposed system and the controllers of [18], [19].

For the desired attitude, the proposed controller has the lowest error as shown in Table 2. For the desired altitude, the proposed controller has an error less than Non linear1 candidate controller. We noticed that the error for the ramp input is constant with time.

3) PARABOLIC INPUT

Parabolic input is applied to the attitude $(\theta_x, \theta_y, \theta_z)$, and to the altitude z . The desired attitude and altitude response $(\theta_{dx}, \theta_{dy}, \theta_{dz}, z_d)$ for parabolic input are shown in Figure 12. E_M is also used to show the efficiency of the proposed controller in parabolic input.

For the desired attitude, the proposed controller has the lowest error as shown in Table 2. For the desired altitude, the proposed controller has an error less than Non linear1 candidate controller. The error for the parabolic input increases with time. The error for the ramp and parabolic

input is high because the controllers are designed to regulate the quadcopter not to track.

VI. CONCLUSION AND FUTURE WORK

A regulator for stabilizing the attitude and altitude of a quadcopter based on pseudo feedback linearization has been proposed in this paper. The proposed controller is embedded into the quadcopter system. Basic principles of quadcopter control mechanisms have been reviewed and analyzed. To confirm the efficiency and performance of the proposed controller, two state-of-the-art quadcopter controllers are compared with it. Simulations are performed to assess the transient and steady-state performance of the three controllers. Results confirm the validity of the proposed controller design. Simulation experiments covered various cases for the three controllers to prove the stability in each case. The global asymptotic stability of the proposed controller has been verified by the Lyapunov stability theorem. Moreover, the robustness of the control strategy is proved when the system is subject to different disturbances (step, ramp, and parabolic). The steady-state error indicator has been used to evaluate the efficiency in step disturbance input, and the Max error in ramp and parabolic disturbance input. Although the proposed controller was simple, it was the best when the quadcopter was in a critical pose. To the best of our knowledge, a few previous works have achieved global asymptotic stability which primarily achieves a large region of convergence. In this research area, the Feedback linearization regulator is our first attempt, and the future work will be tracking the attitude and altitude of the quadcopter.

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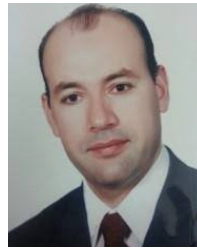
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