Design of High Pointing Accuracy NPSAT-1 Satellite Attitude Systems of Armature Controlled DC Motor with utilization for PD Controller

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Abstract: *An Attitude control system plays the important role to maintain the satellite to desired attitude orientations. The intended application of NANO satellite in low earth orbits (LEO) helps find transient responses with and without controllers. LEO satellites typically orbit at an altitude ranging between160-2000 km. LEO satellites are widely used for remote sensing, navigation, and military surveillance applications. The Nano NPSAT-1 satellite attitude control systems (ACS) are described in this research work. The high pointing accuracy attitude estimation and feedback control systems are presented. The design specifications have been taken to meet the accuracy requirements (desired value ≤ 0.2 seconds) of Nano satellite attitude control. The feedback signal from on-board sensors compared with reference orbit trajectory and implementation of the Proportional Derivative (PD) controller is constructed. An algorithm of Nano satellite (NPSAT-1) attitude control is implemented using MATLAB Tools. In addition, the closed loop poles help find the gain of the system using* ***Root Locus*** *(RL) methods. The satellite control system is used to improve the transient response like overshoot and settling time of the system. Thus, the design of attitude control to improve the rise time, the settling time, the maximum overshoot, and no steady state error was carried out.*

Key Words: *NANO-Satellite, LEO, PD Controller, Root Locus Response*

1. INTRODUCTION

The Attitude Control System (ACS) is important to maintain the satellite into prescribed/determined orbit from perturbed/disturbed orbit. The attitude (Roll angle, Pitch angle, and Yaw angle) of satellite changes due to the orbital perturbation. It is mandatory to control & correct the attitudes of the satellite into the actual orbit [1], [2]. This paper presents the design of the attitude controller for NANO satellites as given below

- NPSAT-1 Satellite

The controller used to reduce the oscillation due to the perturbation forces affects the attitude of the satellite.

The effects of satellite dynamics without controller and with controllers are compared.

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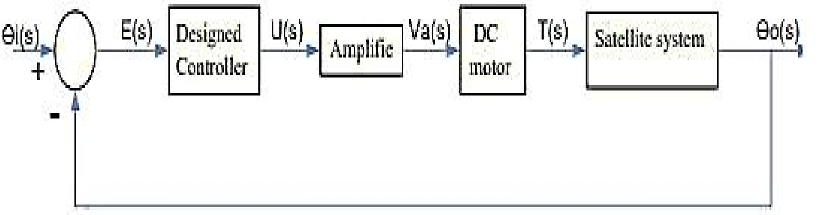


Fig. 1 Satellite Attitude Control System [3]

The design of Proportional – Derivative (PD) controller has been introduced for various transient responses of the NANO satellite attitude corrections. Figure 1, shows the satellite attitude control system, with actual attitudes measure by Gyros, which can be compared with reference attitudes [3]. The comparator produces the error signals E(s) fed to the controllers. PD controllers are introduced in the satellite control system. The steady state is settled exactly at 1 (zero steady-state error). Hence, there is no need for integral control. The PD controller generates the Armature voltage (**V_a**) supplying the DC motor to control armature current (**I_a**) in the input circuit. The DC motor acts as an actuator, DC motor output is Torque (**T**). The armature coil consists of shaft used to drive the load. The Attitude determination and control system (ADCS) has considered the satellite inertia load [3]. In the input circuit, due to variation in the armature voltage, the output circuit of satellite inertia (Load) induces the angular velocity (**ω**). This angular velocity has generated, because of the mutual inductance in the secondary circuit. In secondary coil is induces the back-e. m. f (***b***) due to the changes in the current in the primary coil. The torque is inversely proportional to the armature current shown in equation (1)

$$\mathbf{T = K\, I_a} \tag{1}$$

K = Torque constant; $\mathbf{T} \propto \mathbf{I_a}$

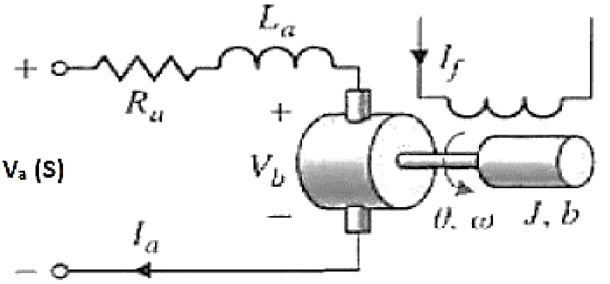


Fig. 2 Armature controller DC motor [3]

Figure (2) shows the armature-controlled DC motor, where **R_a** is armature resistance, **L_a** is armature inductance, **V_b** is back e. m. f voltage, **J** is the moment of inertia, **b** is the back e. m. f constant, **I_f** is field current [3]. The magnitude of back e. m. f voltage is opposite to the input voltage.

$$\mathbf{V_b = k\, \omega} \tag{2}$$

Hence, equation (2) the angular velocity is inversely proportional to back e. m. f voltage. The back e. m. f is expressed as V/(Radian/Sec). In the proposed design considered back e. m. f. constant is 0.85. In ACS, V_b = 0.85 V generates the angular velocity ω (1 Radian/Seconds). The one Ampere current (I) produces the torque 0.85 (Nm). Since, the torque is proportional to the armature current. This is highly used to control the attitudes of Nano satellite [4]. The transfer function of armature-controlled DC motor is given in equation (3)

$$\frac{T(s)}{V_a\,(s)} = \frac{K}{L_a\ s + R_a} \tag{3}$$

Table 1. ACS DC Motor design Parameters [3]

Parameters	Value
Torque constant, K	0.85
Armature Inductance, L_a	0.003H
Armature Resistance, R_a	1 Ohms (Ω)
Back e. m. f. constant, b	0.85

Table 1 shows the ACS design parameters; the sensitivity of Gyro is (V/ (deg/Sec). In order to measure the actual attitude from satellite the Gyro gain chosen is one. (Gyro Gain = 1). The second order characteristics equation is consider for satellite attitude dynamics. The Nano-satellite time domain transient response signal has reached the maximum value. The transient response of satellite such as rise time, overshoot, settling time has improved with controller [4]. This control torques is use to stabilize the orientation of a satellite into actual path in orbit [5].

2. LITERATURE REVIEW

A perturbation is a deviation from some typical or expected movement. These irritations, or varieties in the orbital components, can be arranged having in view how they influence the Keplerian components. In orbit, the common varieties in the component, brief period varieties are occasional in the component with a period no greater than the orbital period. The satellite orbit changes due to the perturbation forces affecting the highest pointing accuracy attitude control and the determination system, which is used to keep the satellite into the predicted trajectory. The attitude control design needs a suitable controller to trigger the actuator for the required demands. At LEO, satellite control relies on magnetic torquers, which can interact with GEO (Earth) magnetic field as the resultant torque is used to control the Nano or Small satellites. Fischel1, R. E, (1963) “Passive Magnetic Attitude Control for Earth Satellites”, Advances in the Astronautical Sciences, Volume 11, Western Periodical Company Hollywood, Calif. The research worked on the vertical stabilization scheme, which has been incorporated in the low earth orbit [6]. The research discussed about the satellite attitude control using magnetic actuator (or) magnetic torquers in low earth orbiting satellite. The magnetic torquer consists of solenoid coil, the current ‘I’ in the coil produce the magnetic flux that interacts with GEO magnetic field vectors. The research work, how this magnetic field generates the control torque to controlling the satellite with suitable algorithm/control laws [6]. The attitude control system of small satellite relies on magnetometer signal. WH Steyn, (2001), “Comparison of Low-Earth Orbiting Satellite Attitude Controllers Submitted to Controllability Constraints”, [7]. The works present the magnetic torque control of the satellite dynamics. The attitude control is introduced mainly in the satellite orbital plane not only in the equatorial plane. The work has further proposed that the magnetic field can be taken as periodic changes in the earth’s atmosphere. The stabilization platform is achieved by the magnetic moments produced by the satellite body, which, consists of magnetic torque rods. This field interacts with a GEO magnetic field and produces the control signal to the actuator. In the orbit, Dipole is considered as a non-rotating platform with the combined earth magnetic field [7]. By applying the above-mentioned theory on the stabilization of small/Nano satellite, the attitude control is achieved by magnetic torques or magnetic moments. Jonas Elfving, (2002), the author of “Attitude and Orbit Control for Small Satellites”, presents various sensors and estimations, which are used to predict the satellites current position, velocity, attitude and angular velocity [8]. The current work discusses the designs pointing accuracy of a satellite when using different sensors and actuators so that a craft does not get too expensive. The work based on experiments analyzed the satellite minimum solar radiation pressure at higher altitude, which has more effect on satellite, but influences aerodynamic drag that we have discussed in the atmospheric drag [8]. The orbital elements in the satellite change due to the periodic variations in solar pressure. For calculating the radiation in solar pressure, it is more important that the satellite focus on the SUN.

3. METHODOLOGY

The design the Nano satellite attitude control transient response of using PD controllers.

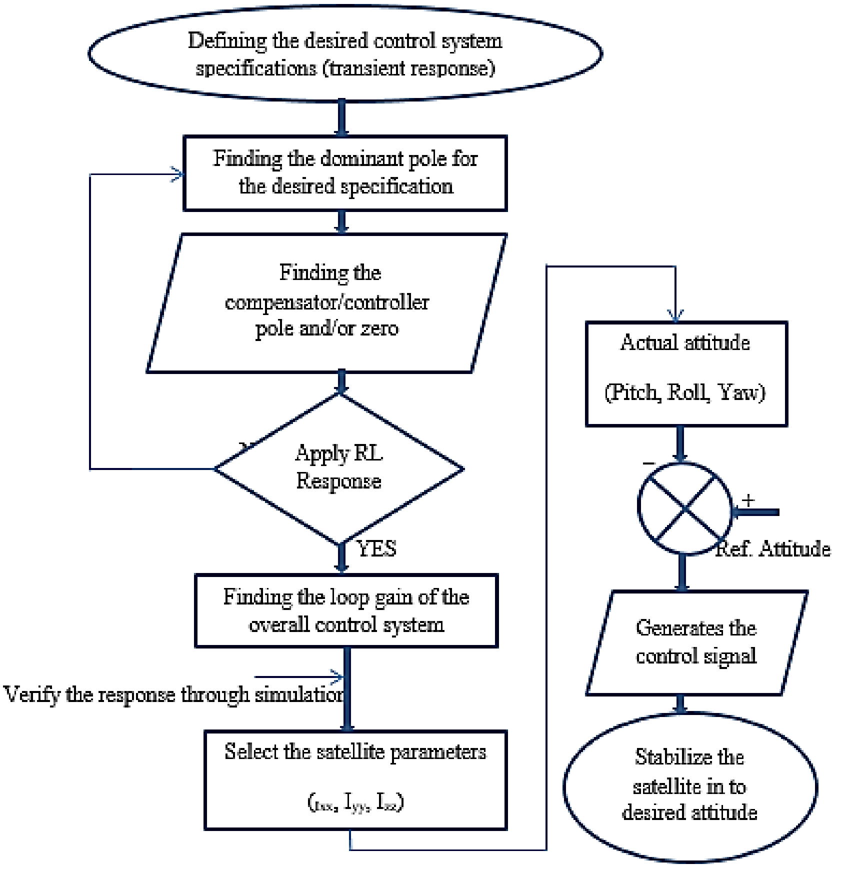


Fig. 3 Flow chart for design of satellite attitude controller

Discussed Control System Design Response Specifications:

- Settling time ≤ 0.2 seconds,
- Improve the Peak time, Rise time,
- Reduces the overshoot (Damping),
- Zero steady state error.

4. DESIGN ALGORITHM FOR PD CONTROLLER

The Proportional-Derivative (PD) controllers has been introduced for Nano satellite/plant attitude control. In flowchart (See Figure 3) illustrate a design steps of PD controller. Theoretically, the damped frequency of oscillation is $\omega_d = \omega_n \sqrt{1-\zeta^2}$, ω_n is the natural frequency of oscillation (rad/ seconds). The plant **is** referred as satellite dynamics. PD controller proportional gain $\mathbf{K_p}$ and derivative Gain $\mathbf{K_d}$ in the system [9]. The design of satellite control systems consists of two parts, one is Attitude determination (AD) another Attitude control (AC). The satellite subsystem is used to determine the attitude, predict the future attitude, and control the attitude of satellite. This paper includes the attitude control parts of the Nano spacecraft/satellite attitude corrections [10]. The main part of ADCS is actuator and spacecraft dynamics, controllers and attitude sensors. The satellite attitude sensor such as Gyro is used to determine the actual attitudes. The satellite reference input signal is compares with feedback signal from the attitude sensor. If there is any variation in the feedback signal with respect to reference/pre-determined attitude, then initiate the actuator to generate the control torque/forces. The damping ratio is desired types of oscillation. The first part of equation $\zeta\omega_n \pm j\omega_d$ indicates the real value; second part indicates the imaginary value. As per the design specification the settling time less than 0.2 seconds is calculated by $\zeta\omega_n = \frac{4}{0.2} = \mathbf{20}$ [9].

$$\tan(34.3) = \frac{\omega_d}{20} \Rightarrow \omega_d = \mathbf{13.64}$$

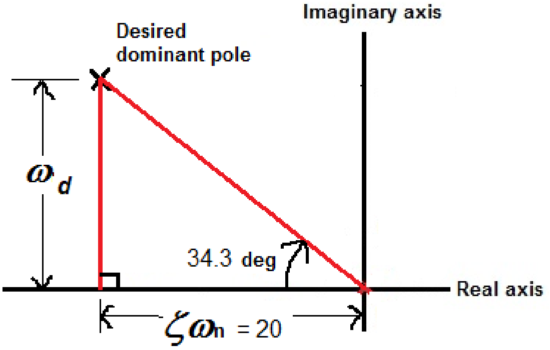


Fig. 4 Location of S-Plane

Moment of Inertia of NPSAT-1

$$= \begin{bmatrix} 24.67 & 0 & 0 \\ 0 & 22.63 & 0 \\ 0 & 0 & 11 \end{bmatrix} \text{kg-m}^2$$

To determine the imaginary part value it is required to find the angles from closed dominant pole, therefore draw the line from closed dominant pole to origin of the s-plane (or) frequency plane. $\Theta = \cos^{-1}(\zeta) = \cos^{-1}(0.826) = \mathbf{34.3}$ as shown in Figure 4 (Here, for design point of view the **overshoot percent is taken as 1%** (equal to a damping ratio of **0.826**) [9].

Various steps to be followed for design of Attitude controller

STEP 1. Define the desired control system specification (transient response and steady-state error).
STEP 2. Find the dominant pole for the desired specifications.
STEP 3. Design the compensator/controller pole and/or zero.
STEP 4. Calculate the loop gain of the overall control system.
STEP 5. Verify the response through simulation.

An algorithm of Nano satellite (NPSAT-1) attitude control is implemented using MATLAB Tools. In addition, the closed loop poles help to find the gain of the system using **Root Locus** (RL) methods. The design simulation considered such as Satellite Attitude determination (SAD), Satellite Attitude Prediction (SAP), and Satellite Attitude Control

(SAC). The SAD is the process of computing the orientation of satellite with pre-determined point accuracy from on board sensors. The SAP is the process of estimating the future attitude of the satellite model. The SAC is the process of controlling the orientation of the satellite. Table (2) shows the design parameter of Nano satellites considered for simulations [11].

Table 2. Design parameters for Nano satellite attitude control

Satellite Details	**NPSAT-1**
Altitude	550 km altitude, Low Earth circular orbit (LEO)
Orbital angular velocity, Ω	0.0011068 rad/s
$\boldsymbol{I_{xx}}$	24.67 kg-m^2
$\boldsymbol{I_{yy}}$	22.63 kg-m^2
$\boldsymbol{I_{zz}}$	11 kg-m^2

Attitude Control: NPS Aurora Satellite (NPSAT-1)

The principle moment of inertia of NPSAT-1 satellite is [24.67, 22.63, 11] kg-m^2. The angular velocity of the satellite is 0.0011068 rad/s at 550 km Altitude [12]. The closed loop response of roll attitude dynamics is $\frac{\phi\ (s)}{T_x\ (s)}$.

The input of the model is torque and output is actual attitudes (Roll angle, Pitch angle, and Yaw angle) of satellite [13]. The sensitivity of Gyro is (V/(deg/Sec).

For getting the actual attitudes from model, the feedback signal is compared with predetermined attitudes. The errors signal as generated from comparators and transfer to Armature control DC motor. The DC motor acts as an actuator.

NPSAT-1 Roll attitude control system

The comparison of reference roll and feedback signals measured from Rate Gyro's (RG) produces (Ref. from figure 1) the errors in the system. The errors are minimized from controllers and generate the control torque to the satellite dynamics.

As per the design, specification of NPSAT-1 settling time is ≤ 0.2 seconds and the considered, closed dominant pole is **–20±13.64*j***. The error signal is expressed in e (t), Satellite input/reference roll signal (ϕ_{ref}) is expressed in degree. The output of the satellite system is ϕ (t).

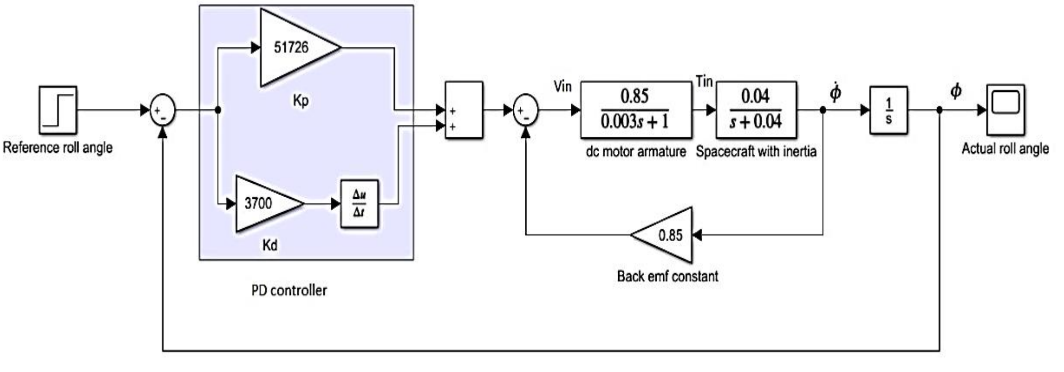


Fig. 5 NPSAT-1 Roll attitude control system

The step response of the roll attitude transfer function without controller given below.

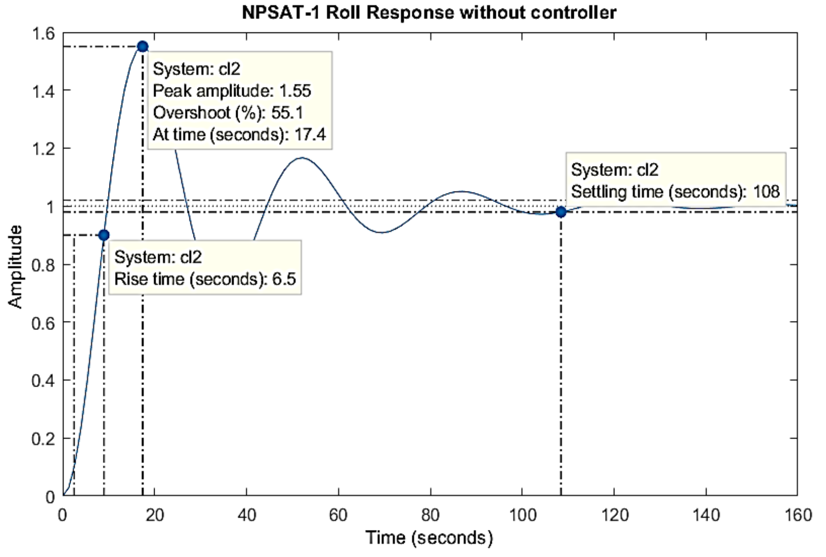


Fig. 6 NPSAT-1 Roll attitude response with step command

Figure 6, clearly shows the damping of the roll attitude system overshoot that is of is 55.1%. The oscillation gradually reduces with settling down at time 108 seconds (we need to reduce it to ≤ 0.2 seconds). The system does not have steady state transient response errors so, the Integral (I) controller is not required [12].

The NPSAT-1 roll attitude dynamics clearly indicates, that there are three poles in the dynamics, namely, one on the origin of the s-plane, another is -0.0639 and -333.264 to find the angles θ_1, θ_2 and θ_3 from closed dominant pole to other poles and zeros. There are no zeros in the pole-zero plots. This method is used to design the PD compensated controller by using root locus analysis and find the zero location. For, finding the PD controller zero (z_c), the line is connected from complex pole to pole at the origin and pole at -0.0639 and -333.264. Let us assume the complex pole considered as '**A**' [13]. The angles from the complex pole (Equation 4) A = 180^0 – (summation of angle measure from complex pole to other poles) + (summation of angle measure from complex pole to other zeros)

$$A = 180 - (\theta_1 + \theta_2 + \theta_3) \tag{4}$$

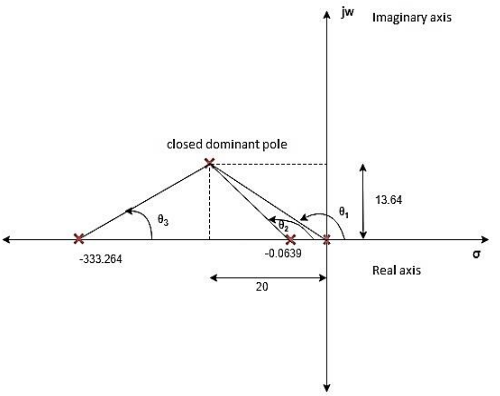


Fig. 7 CDP to Other Poles

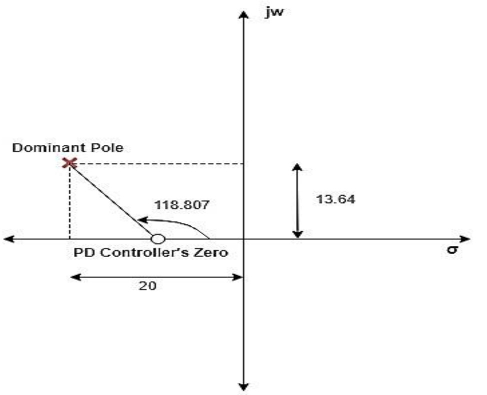


Fig. 8 PD Controller

$$\left.\begin{aligned}\theta_1 &= 180 - \left[tan^{-1}\left[\frac{13.64}{20}\right]\right] = 145.7°\\ \theta_2 &= 180 - \left[tan^{-1}\left[\frac{13.64}{20-0.0639}\right]\right] = 145.614°\\ \theta_3 &= \left[tan^{-1}\left[\frac{13.64}{-333.264-20}\right]\right] = 2.493°\end{aligned}\right\} \tag{5}$$

We substitute the Eqn. (5) angles from closed dominant pole (CDP) to others poles θ_1, θ_2, and θ_3 in equation (4) = 180 – (145.7 + 145.614 + 2.493) = **–113.807**°. Now, we find the PD compensated zero from Figure (8) from angle measured from complex pole A (**-113.807^0**). Using the geometry shown in Figure 8, tan (180^0 - 113.807^0) = $\frac{13.64}{20-z_c}$. To design the PD compensator zero (z_c) from above geometry we calculate the value **–13.98.** Now, the PD controller dynamics is ***K* (s + 13.98)**. The K is the loop gain [14]. Since the system has three poles, so there are three root locus; one travels from origin to negative real axis, another two-root locus start at -159 ± 108i (Break in point) then, travels perpendicular to the real axis. Figure (9) shows, that the NPSAT-1 Roll Root Locus response indicates a gain of 3700 at overshoot (1%),while the damping ratio, ξ is 0.826, undamped natural frequency, $\boldsymbol{\omega_n}$ is 192 (rad/Sec). Here, we considered one percent overshoot (1%) a damping ratio equal to **0.826** to calculate the angle from given equation (5)

$$\Theta = \cos^{-1}(\zeta) = \cos^{-1}(0.826) = \mathbf{34.3°} \tag{6}$$

Draw the straight line at angle 34.3^0 from origin left of the s-plane, when this line is crossing to RL at pole -159 ± 108i record the loop Gain as per damping ration 0.826.

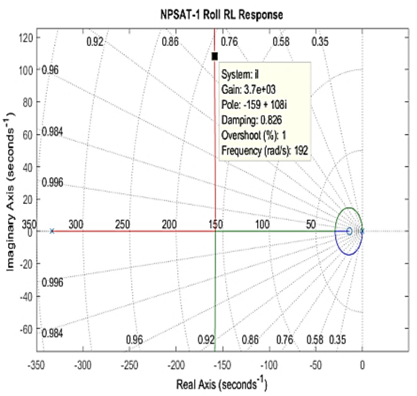


Fig. 9 NPSAT-1 Roll RL response

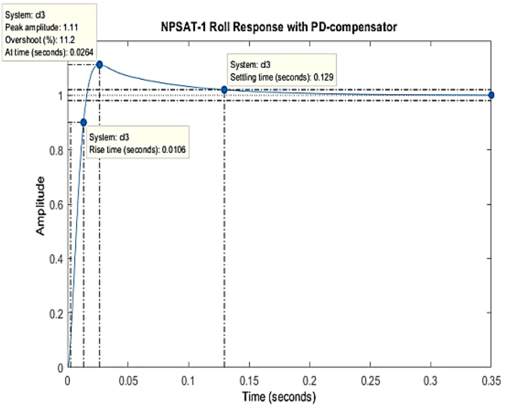


Fig.10 Response with PD

The MATLAB Simulation for closed loop system includes the DC motor dynamics of satellite roll dynamics and PD compensator. As per the design specifications the considered, system settling time is (≤ 0.2), the system responses after implementing the PD compensated controller is 0.129 sec, the rise time is 0.0106, the overshoot is improved by 11.2 %, and the peak time is 0.025 with 1.11 peak amplitude were achieved. The NPSAT-1 SIMULINK responses of the roll attitude control diagram is shown in Figure 11. This model includes the dynamics of armature-controlled DC motor cascade with spacecraft inertia considering the back e. m. f. constant 0.85. To introduce the PD controllers ***K* (s + 13.98),** K is 3700 loop gain calculating from Root locus analysis. The input step signal has considered for satellite control model. The variation in transient response output are measured from scope block. Figure (11) shows the NPSAT-1 Roll response without controller; one can notice that the peak amplitude occurs at 18 seconds peak time [14], [15]. In addition, the response settles down at settling time 108 seconds with zero steady state errors.

NPSAT-1 Pitch attitude control system

The closed loop response of pitch attitude dynamics is given by $\frac{\theta\ (s)}{T_y\ (s)}$.

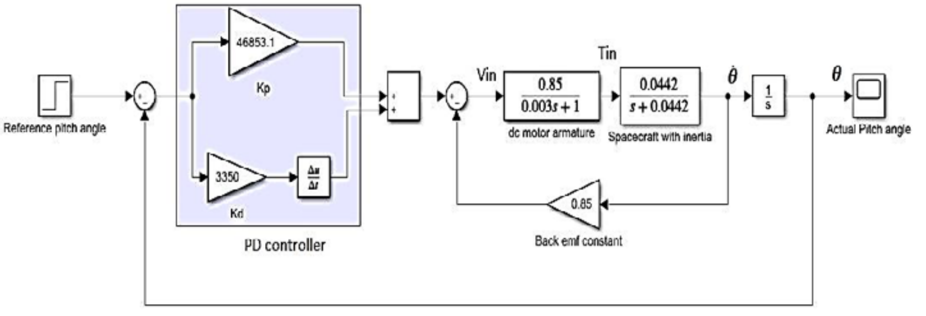
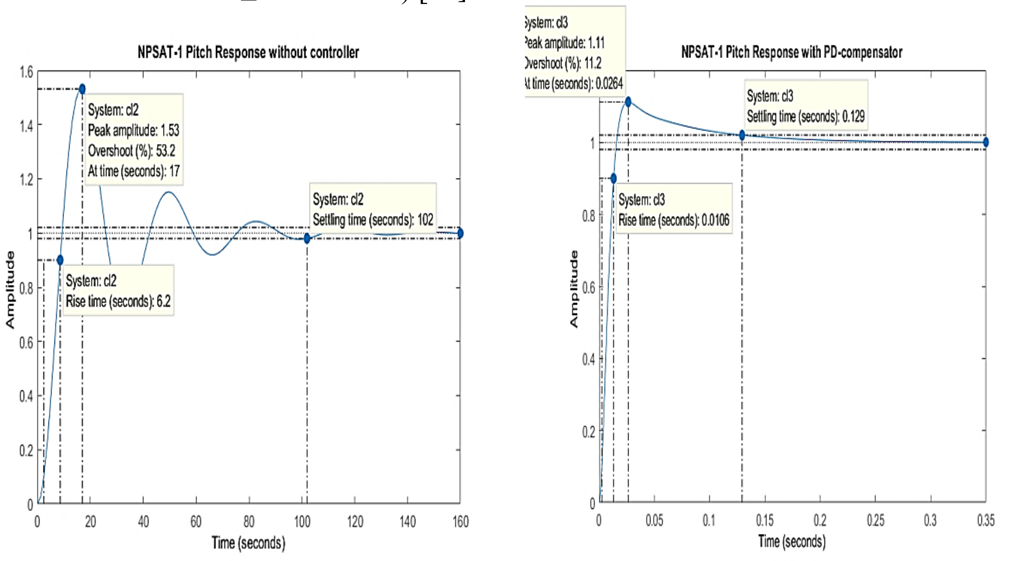


Fig. 11 NPSAT-1 Pitch Attitude Control

The NPSAT-1 pitch attitude step response is shown in figure (11) without controller is given below. Figure 12 (a) clearly shows that the damping of the roll attitude system has more overshoot as 53.2%. The oscillation gradually reduces with settling down at time 102 seconds (we need to reduce it to ≤ 0.2 seconds) [15].



(a) (b)

Fig. 12 NPSAT-1 (a) Without controller; (b) With PD controller

The steady state has settled exactly at 1 (zero steady-state error). Hence, there is no need for integral control. The NPSAT-1 roll attitude dynamics clearly indicates that there are three poles in the dynamics : one on the origin of the s-plane, another is -0.0761 and -333.2572 finding the angles θ_1, θ_2 and θ_3 from closed dominant pole to other poles and zeros [15]. There are no zeros in the pole-zero plots. This method is used to design the PD compensated controller by root locus analysis. Figure 12 (b) shows the NPSAT-1 pitch response, as per design specifications Settling time 0.129 Sec, Rise time 0.0106 Sec, Overshoot (%Mp) 11.2 %, and peak time 0.025 Seconds at peak amplitude 1.11 were achieved.

NPSAT-1 Yaw attitude control system

The closed loop response of yaw attitude dynamics is given by $\frac{\psi\,(s)}{T_z\,(s)}$.

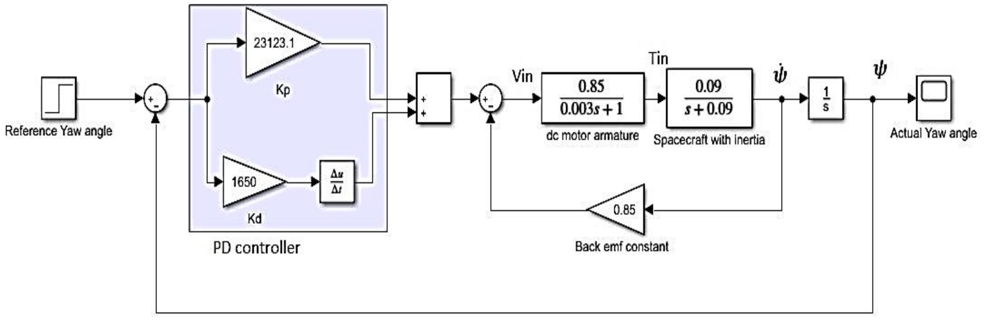
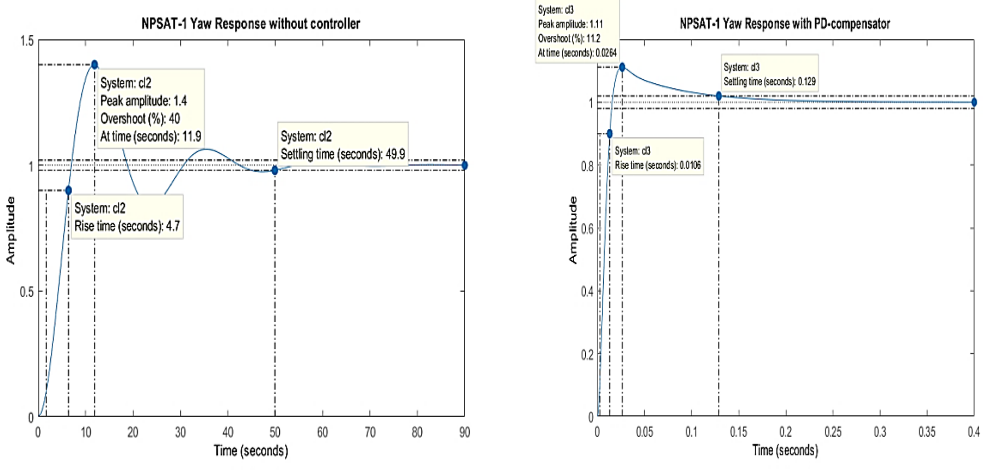


Fig. 13 Dynamics of Yaw attitude control system

Figure (13) shows the NPSAT-1 Yaw attitude step response of the transfer function without controller given below.



(a) (b)

Fig. 14 NPSAT-1 (a) Without Controller; (b) NPSAT-1 with PD controller (Yaw)

From Figure 14 (a), it is clear that the damping of the yaw attitude system has more overshoot as 40%. The oscillation gradually reduces with settling down at time 49.9 seconds (we need to reduce it to ≤ 0.2 seconds) [16]. The steady-state has settled exactly at 1 (zero steady-state error). Hence, there is no need for integral control. As per the design, specification of NPSAT-1 is settling time is ≤ 0.2 seconds at closed dominant pole occurs at **20±13.64*j***. To design the PD compensator zero (z_c,) from the geometry we calculate the value **–14.014.** Now, the PD controller dynamics is ***K*** **(s + 14.014)**. The K is the loop gain. The loop gain found from the root locus at 1% maximum overshoot is 1650. Figure 14 (b) shows the NPSAT-1 yaw attitude response with PD compensator. This includes the dynamics of satellites and orbital perturbation at the low earth orbiting satellite.

Table 3 shows a comparison between the Nano satellite attitudes (Roll, Pitch, Yaw) transient response of control system for NPSAT-1 without controller and with PD compensated controller.

Table 3. NPSAT-1 Nano Satellites Attitude Responses

Specifications	Roll Attitude Dynamics		Pitch Attitude Dynamics		Yaw Attitude Dynamics	
	Without controller	*With controller*	*Without controller*	*With controller*	*Without controller*	*With controller*
Rise time (sec)	6.5	0.0106	6.2	0.0106	4.7	0.0106
Overshoot (%)	55.1	11.2	53.2	11.2	40	11.2
Settling time (sec)	108	0.129	102	0.129	49.9	0.129
Peak time (sec)	18	0.025	8	0.025	12	0.025

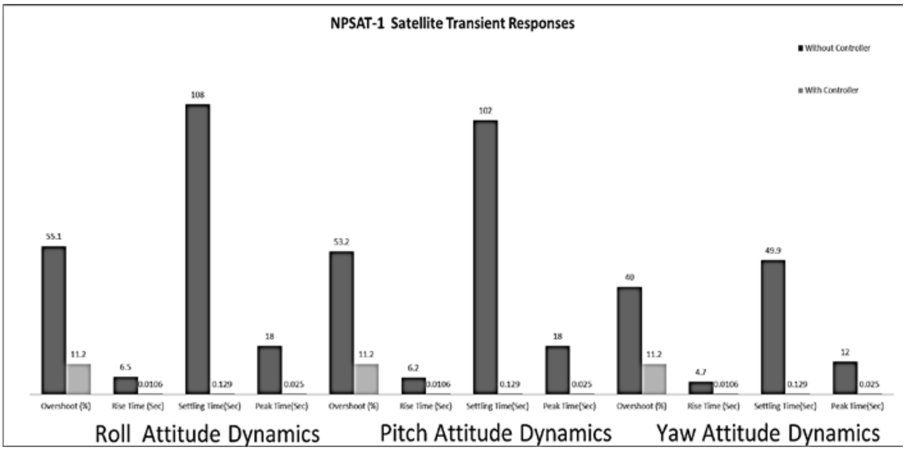


Fig. 15 NPSAT-1 Satellite Transient Response (Comparative analysis)

Thus, PD controllers has introduced ***K* (s + 14.014),** where K is the loop gain calculating from root locus analysis as 1650 at 1% overshoot. The oscillation of satellite due to the perturbation forces measured different time domain specification and variations in maximum overshoot expressed in (%).

This indicates the damping of actual to the critical value [17], [18]. As per the design specifications, settling time is less than 0.2 sec, after implementing the PD controller in the feed forward loop, the settling time 0.129 Sec and Rise time 0.0106 Seconds.

6. CONCLUSIONS

The NPSAT -1 design parameters are [I_{xx} = 24.67 I_{yy} = 22.63 I_{zz} = 11] kg-m^2, Ω = 0.0010239 rad/s (at 550 km Altitude) Low Earth circular orbit (LEO). The compared responses of satellite with controller gain K value 3700 for NPSAT-1 LEO satellite are shown in Figure (15) where the outputs transient response of Nano satellites (NPSAT-1) analysis of without controllers and with PD compensated controller were compared. It is noticed that attitude responses of Nano satellite without controllers has large overshoot and settling time (NPSAT-1 is %Mp = 55.1%, & ts = 108 seconds).

The PD controllers has been introduced in the forward loop of satellite dynamically. It is uSsed to increase the transient response (Overshoot and Settling time) of the system. The output responses of Nano satellite after implemented PD controllers of NPSAT-1 is %Mp = 11.2%, & ts = 0.129 seconds.

The Nano satellites comparative analysis with PD compensated system has achieved as per the design specification, settling time ≤ 0.2 seconds; the NPSAT -1 Nano satellites output attitude transient response meets the design requirements.

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