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Research Article

Attitude Determination and Control System design of KufaSat

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Abstract

In this paper the design of attitude determination and control subsystem of KufaSat Nanosatellite is presented. A three axis magnetometer, six single axis sun sensors, three axis gyroscope and GPS receiver are used as the sensors for attitude determination. TRIAD, algorithm are used for determining attitude estimate from two vector measurements This estimate is then passed to extended Kalman Filter, along with the gyroscope measurements, to obtain a finer attitude. The attitude controller is designed to achieve desired attitude with an accuracy of 5 degrees in nadir pointing using three orthogonal magnetic coils. Two attitude control modes has been considered , detumbling mode and stabilization mode , B-Dot control algorithm is used in detumbling mode while quaternion feedback regulator algorithm is used in stabilization mode. Performance of the control system is verified through closed loop simulations involving models of satellite kinematics and dynamics, space environment, sensors, control law and actuators. Simulations and results of both detumbling mode, and stabilization mode are presented. The simulations show that the satellite will detumble in 60 minutes after separation from the launcher and the stabilization mode controller able to point the satellite with a maximum error of 5 degrees.

Keywords: *Attitude Determination, Attitude Control, Quaternion, B-dot algorithm, Nanosatellite, KufaSat*

1. Introduction

The Attitude Determination and Control System (ADCS) can be divided into two subsystems, attitude determination subsystem (ADS) and attitude control subsystem (ACS). Attitude determination is the process of computing the orientation of the spacecraft relative to either an inertial reference or some object of interest, such as the earth.

Attitude determination uses a combination of sensors and mathematical models to collect vector components in the body and inertial reference frames, typically in the form of a quaternion, Euler angles or rotation matrix (Christopher D. Hall, 2003). Sensors are needed to sense the orientation of the satellite. Several types of sensors are available on satellite area such as Earth Sensors, Sun Sensor, Star Trackers, Magnetometer, Gyroscopes, GPS receiver. Attitude control is the process of orienting the spacecraft in a desired attitude with sufficient accuracy in the space environment. It consists of two areas, attitude stabilization, and attitude maneuver control. Attitude stabilization is the process of maintaining an existing attitude relative to some external relative frame. Attitude maneuver control is the process of reorienting the spacecraft from one attitude to another (James R. Wertz, 1978). Actuator and controller are the main parts of attitude control system; actuator which may be active or passive applies the desired torque to adjust the attitude.

The attitude sensors provide vector measurements that are passed through the determination algorithm to determine attitude estimate. The control algorithm compares the estimated attitude with the desired attitude and calculates appropriate control torques to minimize this error. This torques is sent to the appropriate torque coils to exact a moment on the spacecraft.

The combination of attitude determination system and attitude control system can be summarized in a block diagram of attitude determination and control system ADCS as shown in Fig. 1

A quick review to the literature of various ADCS designs to know some important parameters of these systems like types of sensors used to attitude determination, types of magnetic control active or passive used to attitude control. A survey of 94 publicly known nanosatellite projects which are done by (J. Bouwmeester, J. Guo,2010) for statistical analysis show that,

- The most common used sensors are sun sensors and magnetometers. Earth sensors and gyros are also used. About 16% of the nanosatellites are equipped with a GPS receiver, thereby having a direct means of onboard navigation.
- Magnetic control, either passive or active, is very popular in nanosatellites. Since almost all nanosatellites operate in LEO, magnetic control is a simple and effective means of attitude control. About 40% of the nanosatellites have active attitude control, and 40% passive control, mostly by means of magnetic material and 20% does not have any attitude control at all, leaving the satellites tumbling free in space. Spin-stabilization and a gravity gradient boom

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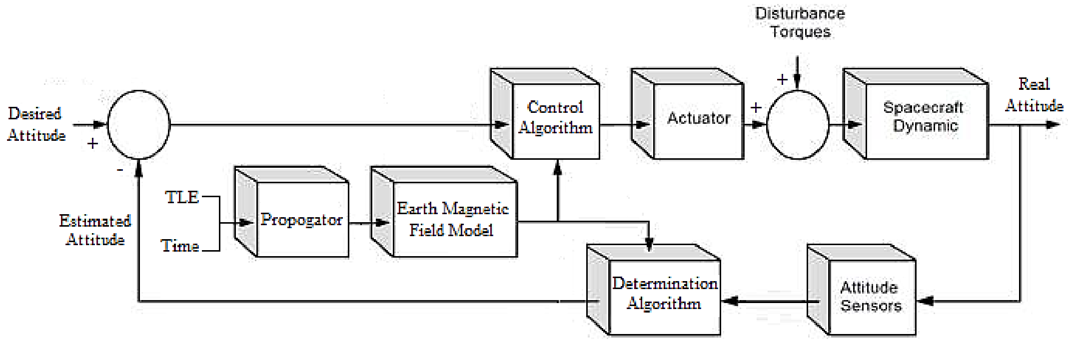


Fig.1 Block diagram of ADCS

are also simple but effective means of attaining static attitude (J. Bouwmeester, J. Guo,2010) .

Another survey done by (Nagarjuna Rao Kandimala, 2012) to 24 nanosatellites missions that have been launched, which covers all kinds of attitude sensors show that:

- 16 nanosatellites (CanX-1 University of Toronto, DTUsat Technical University of Denmark, AAU Cubesat Alborg University, Quake Sat Stanford University, NCube2 Norwegian University of Science and Technology, CUTE 1.7 + APD Tokyo Institute of Technology , KUTEsat Pathfinder University of Kansas, CP2 California Polytechnic Institute , CSTB-1 Boeing Company , Compass One Fachhochschule Aachen , Polysat CP6 California Polytechnic State University , AtmoCube University of Trieste , Goliat University of Bucharest , PW-Sat Warsaw University of Technology , Swiss Cube Polytechnical School of Lausanne , and SRMSAT SRM University) used magnetometer in attitude determination subsystem,
- 10 nanosatellites (CUTE-I Tokyo Institute of Technology, DTUsat Technical University of Denmark, AAU Cubesat Alborg University, CUTE 1.7 + APD Tokyo Institute of Technology, KUTEsat Pathfinder University of Kansas, CP1 California Polytechnic Institute, CSTB-1 Boeing Company, Delfi-C3 Delft University of Technology, Compass One Fachhochschule Aachen , and Swiss Cube Polytechnical School of Lausanne) used sun sensor in attitude determination subsystem.
- 4 nanosatellites (CUTE-I Tokyo Institute of Technology, CUTE 1.7 + APD Tokyo Institute of Technology, PW-Sat Warsaw University of Technology , and Swiss Cube Polytechnical School of Lausanne) used gyroscopes in attitude determination subsystem.
- 7 nanosatellites (CanX-1 University of Toronto, Libertad-1 University of Sergio Arboleda, AtmoCube University of Trieste, Goliat University of Bucharest , PW-Sat Warsaw University of Technology, SRMSAT SRM University, and Jugnu Indian Institute of Technology) used GPS receiver in attitude determination subsystem (Nagarjuna Rao Kandimala, 2012) .

Kufasat is the first Iraqi student satellite project at University of Kufa and has mission to imaging purposes (Mahdi Mohammed Chessab, *et al*, May, 2014). In accordance with CubeSat specifications, it is 1U Nanosatellite with a total mass of 1kg, and its size is restricted to a cube measuring 10×10×10 cm3. It also contains 1.5 m long gravity gradient boom, which will be used for passive attitude stabilization in addition to three magnetic coils used as active attitude stabilization (Mahdi Mohammed Chessab, *et al*, July, 2014).

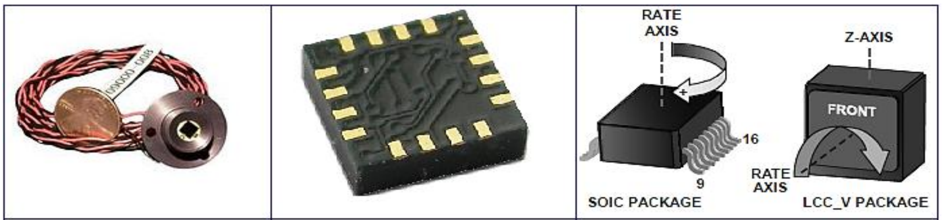
2. ADCS hardware

A. ADS hardware

The attitude determination hardware includes sensors to measure the spacecraft body attitude with respect to inertial space, as well as the spacecraft's angular velocity. The sensors to be used on this satellite are six sun sensors to indicate the direction of the sun, three-axis magnetometer to measure the Earth's magnetic field intensity and direction, and three-axis gyroscope to measure the spinning rate for each axis. The process of selecting the sensors to use for KufaSat included weight and volume considerations, power consumption, and space heritage.

(1)Sun Sensor: Sun sensors are used for providing a vector measurement to the Sun by measuring the angle of the sun vector in respect to the plane on which the sensor is placed. Sun sensors are available in various designs, small sizes and low mass of just a few grams. There are two types of sun sensors, both of them relying on photocells. The first one, the analog sun sensor, also called cosine sensor, it is a simple type which uses the fact, that the output current from a silicon solar cell has a sinusoidal variation with the angle of incoming sun light (James R. Wertz, 1978) . One cosine sensor is a single axis sun sensor. The second one, the digital sun sensor, uses a pattern where different photocells are exposed depending on the direction of the sun (S. T. El .M .Brembo, 2005). By installing more sensors in different directions on the planes of the satellite's body it is possible to determine the sun vector with respect to the center of mass, which can be used to gain the exact attitude of the CubeSat in relation to the sun.

The sun sensor designs considered are categorized as three- axis, two- axis, and single-axis sensors. Because of the simplicity in design it has been decided to use six single axis sun sensors



(a) Coarse Sun Sensors SPACE MICRO	(b) 3-Axis Digital Compass Honeywell	(c) Digital Output Gyroscope ANALOG DEVICES

Fig.2 (a) Sun Sensor (b) Magnetometer (c) Gyroscope

CSS-01, 02 Coarse Sun Sensors from SPACE MICRO Fig. 2a, one on each face of KufaSat .This type of sensors are temperature dependent and require thermistor for temperature measurements, so LM75B temperature sensor, from NXP Semiconductors, are used to monitor the temperature of the sun sensor.

(2)Magnetometer: Magnetometers are inexpensive, lightweight, and highly reliable sensors that are carried on most low Earth orbit spacecraft. As they provide us with information about the attitude of the spacecraft and therefore, they become interesting for small satellite systems (T. Bak, 1999). A magnetometer measurers the flux density of the magnetic field it is placed in. By connecting several of these devices in different directions, the information about the angle and strength of the Earth's magnetic field is gained. A main drawback of this system is the fact that the onboard electronic circuits and the ferromagnetic materials used in the orbiting body can influence the measurements in a way that the error of the measurements is extremely high. So that these devices are installed externally most of the times, in order to minimize these effects of measurements' distortion (WARR Antimatter Research Platform, 2012). A three-axis magnetometer is used to measure the Earth's magnetic field, B and outputs three voltages, each corresponding to the magnitude along a component axis. This measurement is taken by Honeywell HMC5883L 3-axis sensor Fig. 2b which was selected as the most viable option because of its accuracy, weight, and flight heritage that provides a compass heading accuracy within(1- 2)degrees (Honeywell International Inc, 2013).

(3)Gyroscope: Gyroscopes determine the attitude by measuring the rate of rotation of the spacecraft. Gyroscopes have a high accuracy for limited intervals. Some disadvantages exist with gyroscopes .Since they measure a change instead of absolute attitude, gyroscopes must be used with other attitude hardware to obtain full measurements, (K. L.Makovec, 2001).

A gyroscope measures the angular velocity around a firm axis. We can estimate the angle of the gyro's rotation by integrating. A combination of 3 gyros, in three orthogonal axis, gives us information about the total angle of steering in a given time interval.

The angular velocity is measured using three ADXRS453 from Analog Devices, Inc. - Fig. 2c which is high performance, digital output Gyroscope (Analog Devices, 2011). Each one measures the angular velocity about a single axis, and together they form a complete angular velocity vector.

(4)GPS receiver: There are two commonly proposed solutions for finding orbital position, Global Positioning System (GPS) and position estimated from a Two-Line Element set using Simplified General Perturbations (SGP4) propagator. The combination of both solutions leads to a rising accuracy and reliability because if one solution fails, you can still use the information of the other system to complete the mission. Accurate position determination is accomplished using a low-cost commercial Global Positioning System (GPS) receiver that has been modified to work in low Earth orbit.

It has been decided to use OEM4-G2L receiver from NovAtel Inc. The receiver board shown in Fig. 3 has a size of 100 × 60 × 16 mm at a mass of 56 g. The small form factor, and more importantly the low power consumption of only 1.6 W (NovAtel Inc, 2006), made the OEM4-G2L the first dual-frequency receiver that could potentially be accommodated and operated on a CubeSat bus.

Fig.3. NovAtel OEM4-G2L dual-frequency receiver board

B. ACS hardware

(1)Actuators (Magnetic coils): To be able to use magnetic coils as actuators for attitude control in a satellite, three coils are placed perpendicular to each other around the satellite's XYZ axes. The principle is to produce a controllable magnetic moment by feed a constant current in the positive or negative direction through the coils. This magnetic moment interacts with the Earth's magnetic field to produce a mechanical torque onto the satellite which will make it rotate (Mahdi M. Chessab, 2013). This is an attractive mode of actuation for a Nano satellite as no moving parts are needed, low, and simplicity of design and construction.

KufaSat coils were designed to meet the requirements of Nano-satellites in addition to take into account the constraint of the power and volume Fig. 4, Table (1) describes coil design specifications for Kufasat. Structure and magnetic coil.

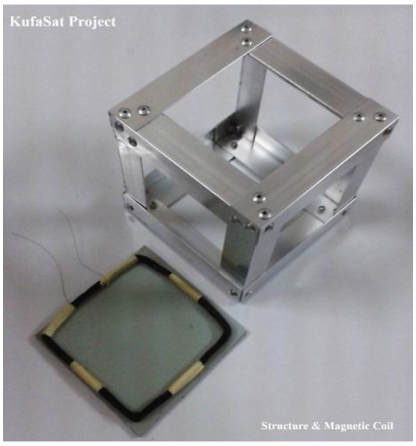


Fig.4 Magnetic coil

Table 1 KufaSat Coil Design specifications

Parameter	Symbol	Value	Unit
Coil length	a	85	mm
Coil width	b	75	mm
Wire diameter	D	0.1016	mm
Wire resistivity	ρ	1.68×10-7	Ω.m
No of turns	N	305	turn
Allowed mass per coil	M	20	g
Max power at full load	P	100	mW
Coil Voltage at full load	V	4.5	Volt
Coil resistance	R	211	Ohm
Coil current	I	21.5	mA
Min Temperature	Tmin	-60	Co
Max Temperature	Tmax	80	Co

(2)Microcontroller: PIC24FJ256GA110 has been decided to be the microcontroller of the ADCS of KufaSat .It is 16-bit general purpose microcontrollers and has the form factor of a 100-Pin TQFP package with physical dimensions 14mm by 14mm. PIC24FJ256GA110 is a low power and high performance micro controller from Microchip with a maximum clock frequency of 32MHz. PIC24FJ256GA110 features 256KB of program memory, 16KB of SRAM, 5(16-Bit) timers, 16 (10-Bit) A/D converters, and 3 comparators (Microchip Technology Inc,2010).It can be summarized the attitude determination and control hardware of KufaSat in Table 2.

Table 2 KufaSat ADCS Hardware

Components	Model	Manufacturer
Sun Sensor	CSS-01, 02 Coarse Sun Sensors	Space Micro
Temperature Sensor	LM75B	NXP Semiconductors
Magnetometer	HMC5883L	Honeywell
Gyroscope	ADXRS453	Analog Devices, Inc.
GPS Receiver	OEM4-G2L	NovAtel Inc
Microcontroller	PIC24FJ256GA110	Microchip Technology Inc

3. ADCS Software

A. ADS algorithms

It can be divide algorithms for estimating three-axis attitude into two classes, Deterministic methods and Recursive estimation algorithms. Deterministic methods need at least two vector measurements obtained at a single point in time to determine a three-axis attitude. If a vector measurement is missing the deterministic solutions cannot provide an attitude. Some common deterministic solutions are: TRIAD, SVD, Q-method, FOAM, QUEST and ESOQ (F. Landis Markley and Daniele Mortari, 1999).

The recursive estimation algorithms use both present and past measurements for determining the attitude. The Kalman filter or the extended Kalman filter is recursive estimation algorithm utilizing a state-space model of the system (Grewal, Andrews, 2001).

(1)TRIAD algorithm: The TRIAD algorithm provides a fast and simple deterministic solution for the attitude. The solutions are based on two vector observations given in two different coordinate systems. TRIAD only accommodates two vector observations at any one time instance. The simplicity of the solution makes the TRIAD method interesting for on-board implementations (T. Bak, 1999). Initially TRIAD assumes that one of the vector measurements is more exact than the other. The vector measurements in the spacecraft body frame are named (b_1 and b_2), and the vectors in the reference frame (r_1 and r_2). It is assumed that the first vector measurement b_1 is the most reliable. Based on this three triads are set up as in (1), (2), and (3) (Christopher D. Hall, 2003).

$$t_{1b} = \frac{b_1}{|b_1|} \qquad t_{1r} = \frac{r_1}{|r_1|} \tag{1}$$
$$t_{2b} = \frac{b_1 \times b_2}{|b_1 \times b_2|} \qquad t_{2r} = \frac{r_1 \times r_2}{|r_1 \times r_2|} \tag{2}$$
$$t_{3b} = t_{1b} \times t_{2b} \qquad t_{3r} = t_{1r} \times t_{2r} \tag{3}$$

Finally the attitude matrix A_{triad} based on the three triads can be written as shown in equation (4)

$$A_{traid} = [t_{1b} t_{2b} t_{3b}][t_{1b} t_{2b} t_{3b}]^T \tag{4}$$

The TRIAD algorithm fails when the two vector measurements are co-aligned or there only one sensor measurement available likes the sun sensor reading during Eclipse period.

(2)Extended Kalman filter: Kalman filter (KF) is an algorithm that uses a series of measurements observed over time, containing noise (random variations) and other inaccuracies, and produces estimates of unknown variables that tend to be more precise than those based on a single measurement alone(Wikipedia).

When the system is linear, KF may be the suitable estimator but the application of KF to nonlinear systems can be difficult.

The most widely used estimator for nonlinear systems is the extended Kalman filter (EKF). The EKF applies the KF to nonlinear systems by simply linearizing all the nonlinear models so that the traditional linear KF

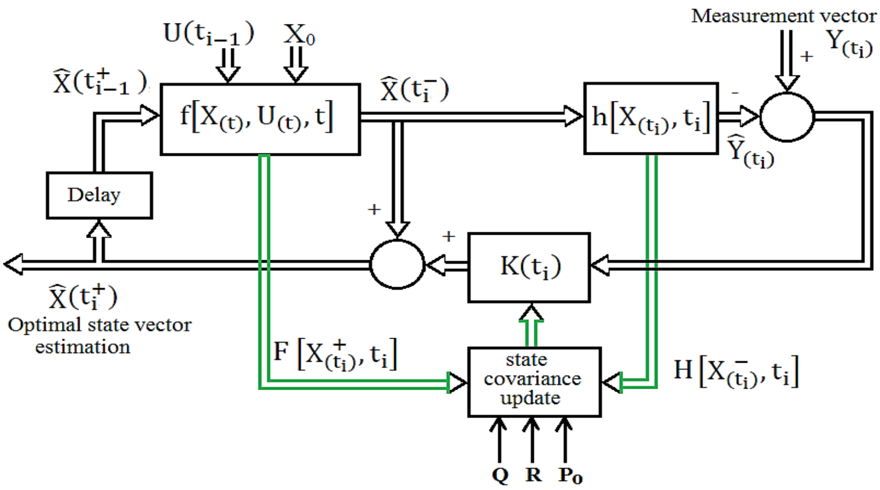


Fig.5 Block diagram of extended Kalman filter (Dhaouadi. Rached, Mohan. Ned, 1991)

equations can be applied. The EKF algorithm can be summarized as follows (Dhaouadi. Rached, Mohan. Ned, 1991): If our system represented by the nonlinear dynamic state space model.

$$\dot{X}_{(t)} = f[X_{(t)}, U_{(t)}, t] + w_{(t)} \qquad (5)$$

where the initial state vector $X(t_0)$ is modeled as a Gaussian random vector with mean X_0 and covariance P_0, $U(t)$ is the deterministic control input vector , and $w(t)$ is a zero mean white Gaussian noise independent of $X(t_0)$, and with a covariance matrix $Q(t)$.

$$Y_{(t_i)} = h[X_{(t_i)}, t_i] + v_{(ti)} \qquad (6)$$

where $v\ (t_i)$ is a zero-mean white Gaussian noise that is independent of $X\ (t_0)$ and $w\ (t)$, and with a covariance matrix $R\ (t_i)$. The optimal state estimate $\hat{X}_{(t)}$ generated by the filter is a minimum variance estimate of $X_{(t)}$, and is computed in recursive manner as shown in Fig. 5.The filter has a predictor –corrector structure as follows (– , + refer to the time before and after the measurements have been processed.

Step 1: Prediction (from t_{i-1}^{+} to t_i^{-})

The optimal state estimate $\hat{X}$ and the state covariance matrix P are propagate from measurement time (t_{i-1}) to measurement time (t_i), based on previous values, the system dynamics, and the previous control inputs and errors of the actual system . This is done by numerical integration of the following equation:

$$\dot{X}_{(t)} = f[X_{(t)}, U_{(t)}, t] \qquad (7)$$
$$\dot{P}_{(t)} = F^T P(t) + P(t)F + Q \qquad (8)$$
$$t \in [t_{i-1}^{+}, t_i^{-}]$$

Starting from initial conditions: $X(t_{i-1}^{+})$, $P(t_{i-1}^{+})$

where $F = \frac{\partial f(X,U,t)}{\partial x}$ (9)

Evaluated at $X = X(t_{i-1}^{+})$

Step 2: Filtering (from t_i^{-} to t_i^{+}) .

By comparing the measurement vector, Y, to estimated one,$\hat{Y}$, a correction factor is obtained and is used to update the state vector. The filter gain matrix $K\ (t_i)$ is defined as

$$K(t_i) = P(t_i^{-})H^T[HP(t_i^{-})H^T + R(t_i)]^{-1} \qquad (10)$$

where $H(t_i) = \frac{\partial h(X,t_i)}{\partial x}$ (11)

Evaluated at $X = X(t_i^{-})$.

The measurement update equations for the state vector and the covariance matrix are

$$X(t_i^{+}) = X(t_i^{-}) + K(t_i)[Y(t_i) - h\{X(t_i^{-}), t_i\}] \qquad (12)$$

$$P(t_i^{+}) = P(t_i^{-}) - K(t_i)H(t_i)P(t_i^{-}) \qquad (13)$$

where, $X(t_i^{+})$ is the optimal state vector estimateAttitude determination process can be dividing into two modes:

Mode 1 sunlight part of the orbit, in which the sun sensor, magnetometer, and gyroscope information are available. Information of sun sensor and magnetometer used first by TRIAD and the resulting quaternion is then passed, along with the rate gyroscope measurements; to extended Kalman filter .This mode provides an accurate attitude measurement.

Mode 2 eclipse part of the orbit, in which only the magnetometer and rate gyroscope information are used by extended Kalman Filter takes longer to converge and accuracy is decreased. This mode provides less accurate attitude measurement.

B. ACS algorithms

Two attitude control modes will be taking into account, detumbling mode and stabilization mode. B-Dot control algorithm is used in detumbling mode while quaternion feedback regulator algorithm is used in stabilization mode.

(1) B-Dot control algorithm: B-Dot control algorithm use magnetometer measurements and the magnetorquers as

control actuators to reduce high rotational rates, resultant from the separation of cubesat from the launcher. This method applies a magnetic dipole via the magnetorquers in the direction opposite to the change in magnetic field which is estimated by magnetometer measurements each few seconds. The control law is:

$$m_b = -k\dot{B^b} - m_c \tag{14}$$

where $m_c = (0\ 0\ m_c)^T$ will dissipate the kinetic energy of the satellite and align it with the local geomagnetic field, B^b is a magnetic field vector (Gravdahl J. Tommy, *et al*, 2003). The magnetorquers apply a torque according to the equation,

$$\tau_m^b = m^b \times B^b \tag{15}$$

(2) Quaternion feedback regulator algorithm: The quaternion feedback regulator algorithm used to calculate the required torque to control the satellite. The control law of the quaternion feedback regulator is formulated as follows.

$$T_{control} = -dI\omega_e - kIq_e + \Omega i\omega \tag{16}$$

where d and k are gain parameters, I is the moment of inertia matrix of the satellite, $\boldsymbol{\omega_e}$ is the error between the desired rotational and the estimated rotational rate vector, $\boldsymbol{q_e}$ is the vector part of the quaternion that describes the error between the desired and the estimated attitude quaternion, $\boldsymbol{\omega}$ is the estimated rotational rates of the satellite and Ω is a skew symmetric matrix (M. Vos, 2013). The term $\Omega\ i\ \omega$ can be discarded, because it only adds computational complexity without providing much more control accuracy (B. Wie, 2008). By canceling this term equation (7) becomes:

$$T_{control} = -dI\omega_e - kIq_e \tag{17}$$

The gains d and k determine the settling time and the damping of the control algorithm.

4. Satellite model

A. Dynamic Equations

The dynamic equation of motion of the satellite about its center of mass can be expressed as (Mahdi M. Chessab, 2013):

$$\tau = d/dt\,(I.\omega) + \omega \times (I.\omega) \tag{18}$$

where τ is summation of external moments exerted about the center of mass of the satellite, $\omega = \quad \omega b/I$. Equation (18) leads to the following three dynamic equations for the roll, pitch, and yaw axes respectively:

$$\tau_x = \dot{\omega_x} I_x + (I_z - I_y)\omega_y\omega_z \tag{19a}$$
$$\tau_y = \dot{\omega_y} I_y + (I_x - I_z)\omega_z\omega_x \tag{19b}$$
$$\tau_z = \dot{\omega_z} I_z + (I_y - I_x)\omega_x\omega_y \tag{19c}$$

These three equations are known as Euler's equations of motion for a rigid body.

B. Kinematics Equations

The quaternion Kinematic equations of motion can be written in terms of the satellite's angular velocity components as follows (Mahdi M. Chessab, 2013):

$$\dot{q} = \frac{1}{2}\,\Omega\,.\,q \tag{20}$$

Where $q = [q_1\ \ q_2\ q_3\ \ q_4]^T$ and Ω is a skew –symmetric matrix defined as:

$$\Omega = \begin{bmatrix} 0 & \omega_z & -\omega_y & \omega_x \\ -\omega_z & 0 & \omega_x & \omega_y \\ \omega_y & -\omega_x & 0 & \omega_z \\ -\omega_x & -\omega_y & -\omega_z & 0 \end{bmatrix} \tag{21}$$

Thus equation (20) will be:

$$\begin{bmatrix} \dot{q}_1 \\ \dot{q}_2 \\ \dot{q}_3 \\ \dot{q}_4 \end{bmatrix} = \frac{1}{2} \begin{bmatrix} 0 & \omega_z & -\omega_y & \omega_x \\ -\omega_z & 0 & \omega_x & \omega_y \\ \omega_y & -\omega_x & 0 & \omega_z \\ -\omega_x & -\omega_y & -\omega_z & 0 \end{bmatrix} \cdot \begin{bmatrix} q_1 \\ q_2 \\ q_3 \\ q_4 \end{bmatrix} \tag{22}$$

C. Sensors Model

The sensors to be used on this satellite are six sun sensors, three-axis magnetometer, and three-axis gyroscope. We assume six solar cells, one in each side of the satellite. Each of the individual Solar sensor readings are proportional to the cosine of the incident angle with some noise, and converted to a unit vector of the sun direction after being processed by the on-board attitude determination software.

$$u_i = cos\theta_i + v_c\ ,\ \forall i = 1{:}\,6 \tag{23}$$

whereu_i measured value, υ_c is solar cell noise. The measured Sun vector can be represented as,

$$s_{m,B} = s_B + \tilde{s}_{m,B} \tag{24}$$

where $\tilde{s}_{m,B}$ is a random zero mean Gaussian variable.

The magnetometer mathematical model is composed by true Earth's magnetic field in body fixed coordinates added to noise and small residual bias. The model is given by:

$$m = B + b_m + \upsilon_m \tag{25}$$

where B is the vector formed with the components of the Earth's magnetic field in the body frame of the reference, b_m is the magnetometer bias, and υ_m is the zero mean Gaussian white noise of the magnetometer. In the gyroscope model the angular velocity is measured from three rate gyroscopes is composed by the true angular rate added to a bias and a white noise. The model is given by:

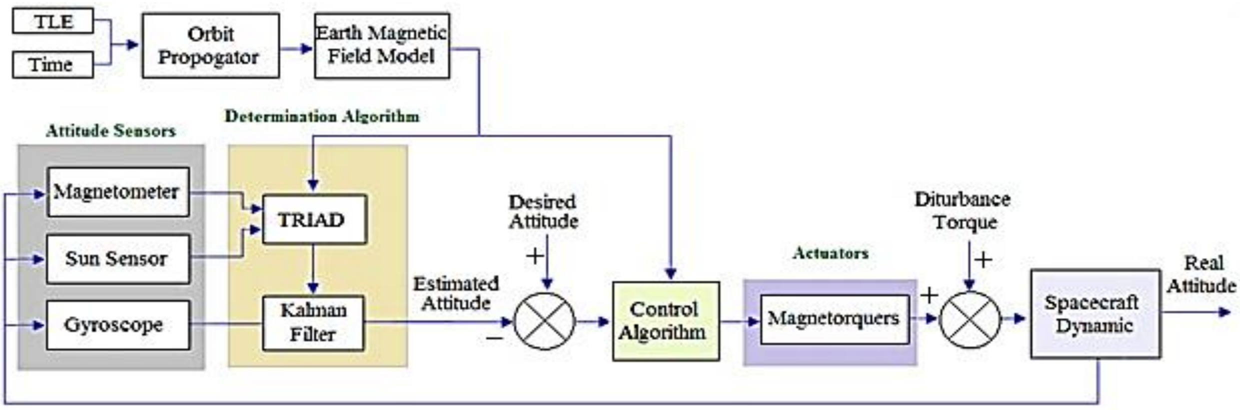


Fig.6 Overall view of the ADCS system

$$\omega_g = \omega + b_g + \upsilon_g \tag{26}$$

where; ω_g is the output of the gyroscope, ω is the real angular rate of the gyroscope, b_g is the gyro bias, υ_g is a zero mean white noise.

D. Magnetic Torquer Model

A current (*I*) flowing through the torque coil with (*N*) turns generates a magnetic dipole moment (*m*), according to the following equation (Mahdi M. Chessab, 2013):

$$m = NIA \tag{27}$$

where; *A* is the cross sectional area of the coil. The interaction of this magnetic dipole moment with the Earth's magnetic field generates a torque, τ_m, according to (28):

$$\tau = m \times B_e \tag{28}$$

where; B_e is the earth's magnetic field at a particular location. By substitution (27) in (28) we get:

$$t_m = NIA \times B_e \tag{29}$$

5. The Orbit Model

To determine the position and motion of a satellite in orbit based on the data from the Two Line Elements, mathematical orbit models are used. Popular examples of orbit models are the Kepler orbit model and the SGP4 orbit model, which will be used in this satellite to calculate the position in the orbit for the purpose of determining the attitude of the satellite. The SGP4 model is the most used and most reliable orbit determination model. It was developed by NORAD (North American Aerospace Defense) in 1970 for the use of near Earth satellites. The SGP4 model is used for calculating orbital descriptions of satellite movement (position and velocity).It is based on the Keplerian orbit calculations but also takes a number of perturbations into account like atmospheric drag and spherical harmonics.

6. Magnetic field model

In order to use the magnetic coils as actuators for attitude control, the algorithms need to have knowledge of the earth's magnetic field at the current orbit position. The International Geomagnetic Reference Field model (IGRF) is a standard mathematical model of the earth's magnetic field that is updated every 5 years by the International Association of Geomagnetism and Astronomy. The IGRF model consists of the Gauss coefficients which define a spherical harmonic expansion of the magnetic scalar potential, providing a very accurate estimate of the geomagnetic field.

7. Attitude determination and control scenario

The attitude sensors provide vector measurements that are passed through the determination algorithm to determine attitude estimate. This estimate is then passed to extended Kalman Filter, along with the angular velocity measurements, to obtain a finer attitude solution. The attitude controller compares the estimated attitude with the desired attitude and calculates appropriate control torques to minimize this error. This torques is sent to the appropriate torque coils to exact a moment on the spacecraft. Fig. 6 illustrates the overall view of the ADCS system.

8. Simulation and results

Attitude control simulation model was built in Matlab Simulink to test two attitude control modes; detumbling mode and stabilization mode. The Top level of Matlab simulation, shown in Fig. 7 involves the models of satellite kinematics and dynamics, the space environment, the sensor models, the control law and the torque actuation.

A. Detumbling mode

B-Dot control algorithm is used in detumbling mode to reduce the rotational rates of the KufaSat. The detumbling mode controller was simulated with initial parameters listed in table 3.The simulated magnetometer measurements, shown in Fig. 8 were adjusted by adding noise to the simulated sensors to make the simulation more realistic. Fig. 9 shows that the B-dot controller brings the roll and pitch axes angular velocity (w1, w2) to zero while steadying the yaw axis angular velocity (w3) to constant value.

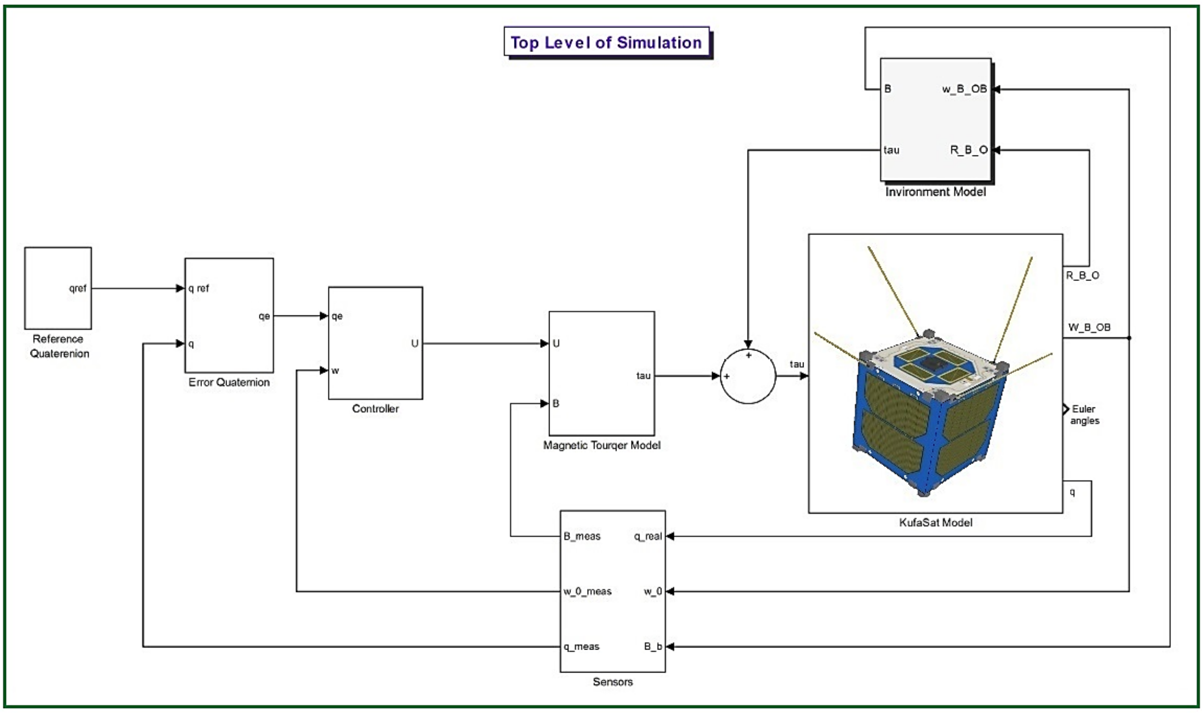


Fig.7 Top level Matlab simulation of KufaSat

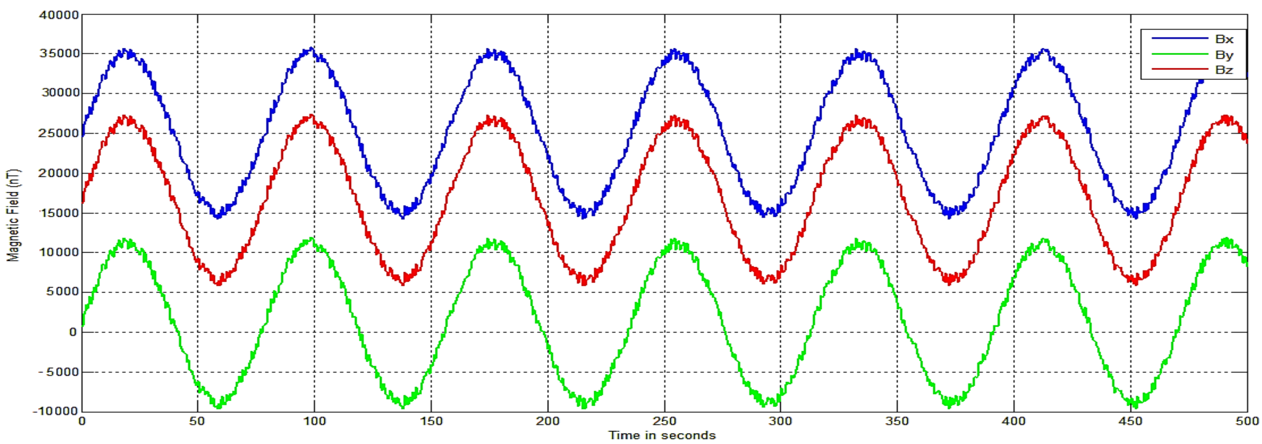


Fig.8 Simulated magnetometer measurements in the satellite body frame

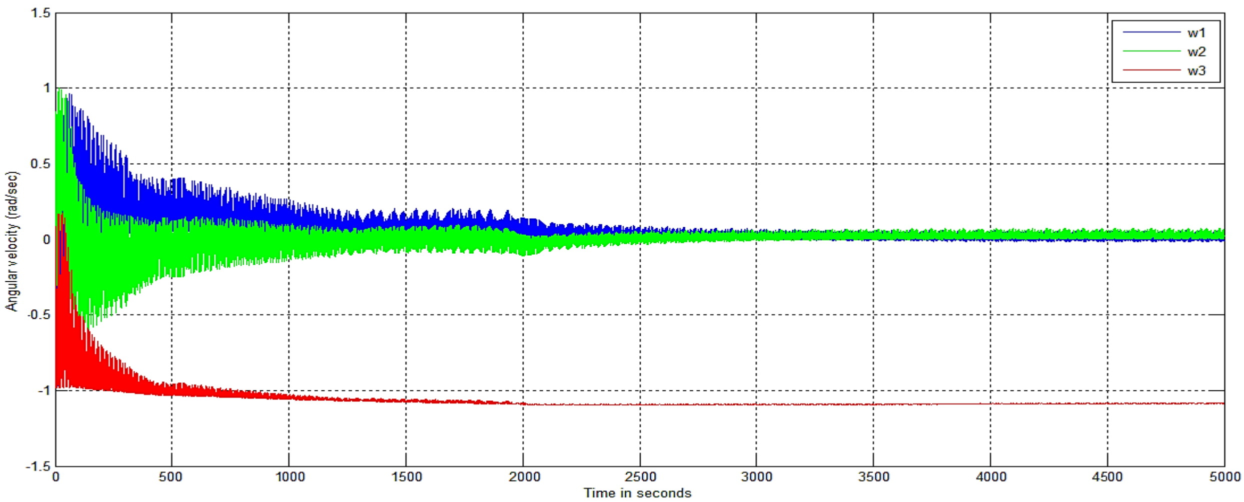


Fig.9 Satellites angular rate during detumbling mode

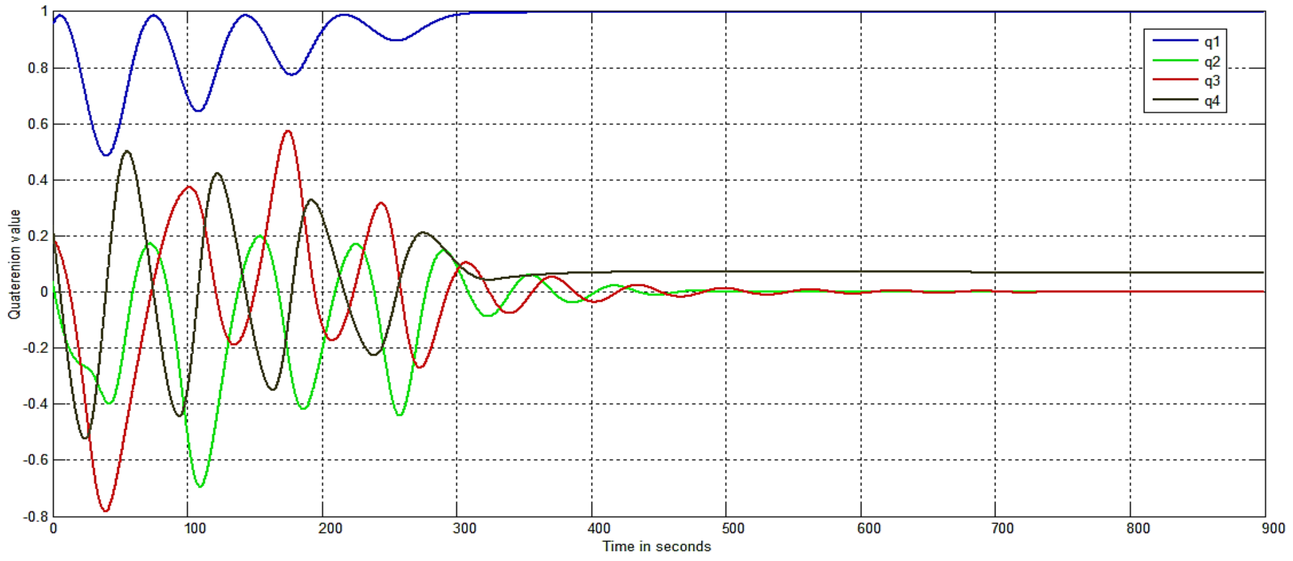


Fig.10 Satellites quaternion representation during stabilization mode

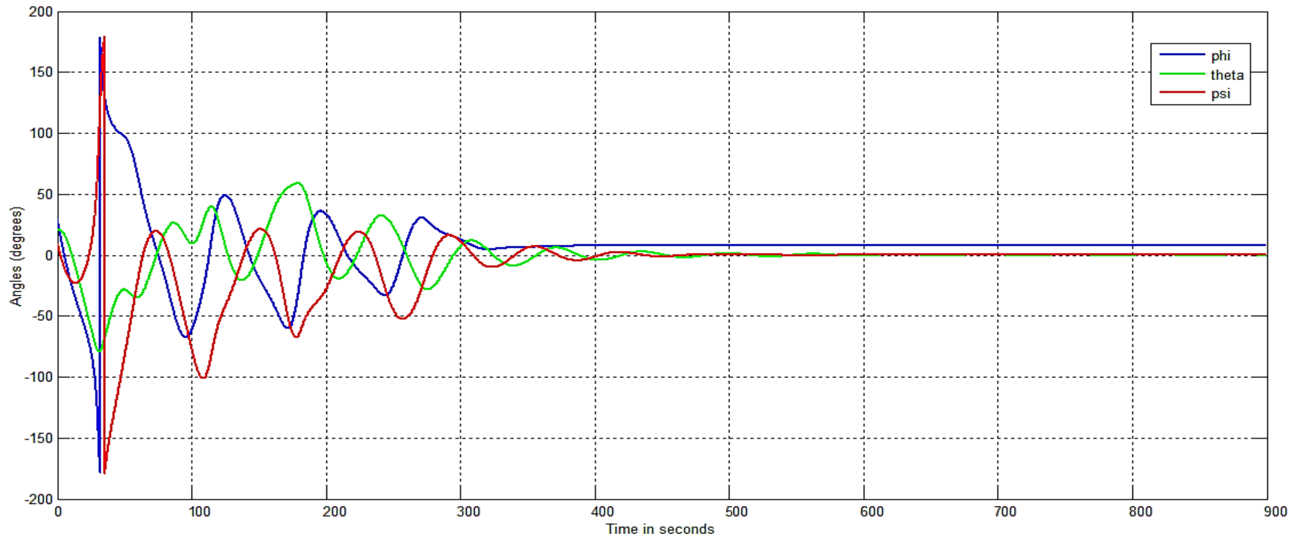


Fig.11 Satellites attitude during stabilization mode

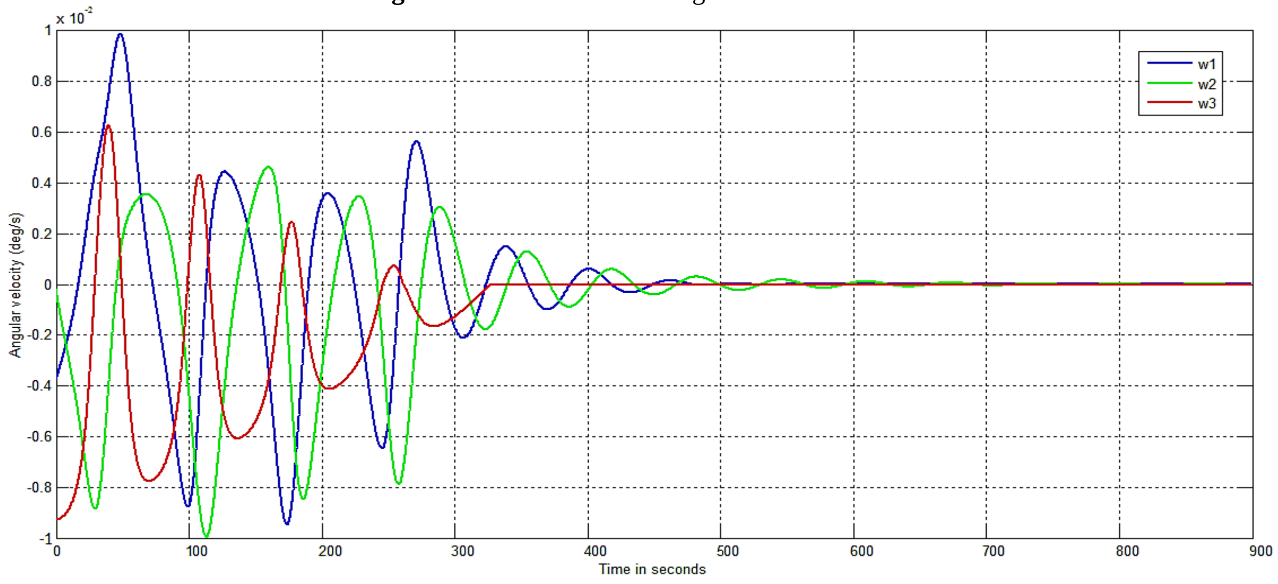


Fig.12 Satellites Angular rate during stabilization mode

Table 3 Initial parameters for detumbling mode

Parameter	Value
CubeSat type	1U
Dimensions	L=W=H=10cm
Mass	1kg
Altitude	600km
Moments of inertia about x	0.1043 kg.m2
Moments of inertia about y	0.1020 kg.m2
Moments of inertia about z	0.0031 kg.m2
Coil cross sectional area	63.75cm2
Initial angular velocity	[0.01, 0.03, 0.02] rad/sec
Initial attitude (roll, pitch, yaw)	[5, 10, 5] degree
B-dot Constant	1e5
Mc	1e-3

B. Stabilization mode

Stabilization mode is considered the default control mode for KufaSat as it is the required control mode for the majority of payload operations. It will allow KufaSat to maintain attitude orientation by using three magnetic coils as the main attitude actuators. The quaternion feedback regulator algorithm used in this mode to calculate the required torque to control the satellite. The stabilization mode controller was simulated with initial parameters listed in table 4.

Figs. 10, 11, and 12 show simulation results (quaternion representation, attitude in Euler angles, and angular rate) for the stabilization mode. Fig. 10 shows that roll and pitch angles go to zero after 450 seconds while steadying yaw angle at constant error about 5 degrees. Fig. 11 shows that angular rate about all axes go to zero after 600 seconds.

Table 4 Initial parameters for stabilization mode

Parameter	Value
CubeSat type	1U
Dimensions	L=W=H=10cm
Mass	1kg
Altitude	600km
Moments of inertia about x	0.1043 kg.m2
Moments of inertia about y	0.1020 kg.m2
Moments of inertia about z	0.0031 kg.m2
Coil cross sectional area	63.75cm2
Initial angular velocity	[0, 0, 0] rad/sec
Initial attitude in Euler angles	[5, 10, 5] degree
Desired attitude in Euler angles	[0,0,0] degree
gain parameters (d, k)	10, 0.01

Conclusion

The objective of this paper was to establish the design of an effective attitude determination and control system for KufaSat based on purely magnetic actuation. The hardware selection comprised of a three-axis magnetometer, sun sensors, gyroscope, GPS, in addition to design and implementation magnetic coils with consideration of the limited resources and the specific performance requirements imposed by the given attitude control task.

The software selection comprised of TRIAD algorithm, extended Kalman filter as attitude determination software and B.dot algorithm, Quaternion feedback regulator algorithm as control software. Simulation results show that the B-dot detumbling controller is capable of reducing the angular rate of the satellite to below the requirement. The performance of quaternion feedback regulator in stabilization mode was satisfied because the simulations show that the controller able to point the satellite with a maximum error of 5∘, so the system stabilization is attained using only electromagnetic coils as actuators.

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Biography

Mohammed Chessab Mahdi had his B.Sc. degree in control and system engineering from University of Technology –Baghdad at 1984 and had his M.Sc. degree in space technology from University of Kufa at 2013. He is full time lecture in Technical Institute of Kufa -Foundation of Technical Education– Iraq and member of KufaSat team - space research unit-Faculty of Engineering University of Kufa. He has good skills in the design and modeling of attitude determination and control systems using Matlab program .

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