Attitude Stabilization for CubeSat

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Concepts and Technology

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To my beloved parents

To my wife Khulood

To my sons:

Roaa

Jannat

Ahmed

Ali

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LIST OF SYMBOLS

A_x ,A_y ,A_z	Area of the magnetic coil (m^2).
ω	Argument of perigee.
x	Attitude and its rate of change state vector.
F^b	Body frame
G_x, G_y, G_z	Components of gravity gradient vector
K	Control gain.
k_x,k_y,k_z	Direction cosines of the Euler axis relative to reference frame.
B_φ ,B_θ ,B_ψ	Earth's magnetic field as it affects the Roll, Pitch, Yaw axis.
F	Force effect on particle (N).
g_n^m , h_n^m	Gaussian coefficients.
m^b	Generated magnetic moment inside the body
F	Gravitational force corresponding to a differential element (N).
G	Gravity gradient vector.
X_i,Y_i,Z_i	Inertial coordinate frame.
u	Input magnetic moment vector.
B	Input matrix representing by the Earth's magnetic field.
B_b	Local geomagnetic field vector.
Ω	Longitude of ascending node**.**
m_x,m_y,m_z	Magnetic moment for each principal axis (Nm).
m	Mass of particle (g).
M	Mass of the Earth.
a	Mean distance between two masses.
ω_o	Mean orbital motion.
μ	Membership function.
I_x,I_y,I_z	Moments of inertia for Roll, Pitch, Yaw axes (kg.m^2).
P	Mutual period of revolution.

G	Newton's gravitational constant.
N_k	Number of windings in the magnetic coil.
F^o	Orbit frame
X_o,Y_o,Z_o	Orbital coordinate frame.
C	Output matrix.
y	Output vector.
μ_o	Permeability of free space.
A	Plant matrix of the attitude dynamic system of the satellite.
a	Practical acceleration vector (N/m^2).
q	Quaternion parameters vector.
φ_b	Roll angle bias due to any z-spin rate (degree).
φ, θ, ψ	Roll, Pitch and Yaw angles (degree).
X_b,Y_b,Z_b	Satellite coordinate frame.
V	Scalar potential function.
P_n^m	Schmidt quasi-normalized.
Ω	Skew-symmetric matrix.
α	The angle between the magnetic moment and the Earth's magnetic field (rad).
$\omega^{b/i}$	The angular velocity of body frame relative to an inertial frame (rad/sec).
$\omega^{b/o}$	The angular velocity of body frame relative to orbital frame (rad/sec).
$\omega_x,\omega_y,\omega_z$	The angular velocity of Roll, Pitch, and Yaw axes (rad/sec).
$\omega^{o/i}$	The angular velocity of the orbital frame with respect to the Earth (rad/sec).
i_x, i_y, i_z	The currents passing through magnetic coils (Amp).
M_x,M_y,M_z	The external torques for each principal axis (Nm).
$T_{Gx}T_{Gy}T_{Gz}$	The gravity gradient torque about each principal axis (Nm).
T_G	The gravity gradient torque vector (Nm).
i_m	The inclination of the spacecraft's orbit with respect to magnetic equator (degree).

I	The inertia tensor matrix of the spacecraft ($kg.m^2$).
T_m	The magnetic torque vector (Nm).
T_{mx}, T_{my}, T_{mz}	The magnetic torques about the Roll, Pitch, and Yaw axes (Nm).
N	The period of one orbit that the satellite needs to complete one revolution around the Earth.
r	The position vector of the mass measured from the center of the spacecraft mass(km).
R_o	The radius vector from the center of the Earth to the center of the spacecraft (km).
M	The summation of the external moments exerted about the center of mass of the rigid body (Nm).
t	Time index (sec).
H	Total angular momentum vector of the rigid body. ($kg.m^2.rad/sec$).
v	Velocity of the particle (m/sec).
γ	Vernal equinox.

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PREFACE

A CubeSat is a miniature satellite for space research applications. It has a cubic shape with 10 cm to each face, and mass of no more than 1.33 kilograms. CubeSat typically uses commercial off-the-shelf parts and electronic components so that it can be deployed quickly and cost-effectively. Because of its small size CubeSat can be carried into orbit cheaply as a secondary passenger on a launch vehicle.

Maintaining the orientation of CubeSat in space in the desired attitude is called attitude stabilization. Attitude stabilization systems are classified as active or passive. Passive attitude stabilization does not require power. Gravity gradient has been used as a means of passive attitude stabilization since the early 1960s. Gravity gradient stabilization alone cannot achieve accurate three-axis control. The simplicity and low cost of active magnetic control make it an attractive option for CubeSat in Low Earth Orbit (LEO). Magnetic torquing combined with gravity gradient stabilization represents an attractive method of attitude stabilization for CubeSat.

The purpose of this book is to develop a nonlinear mathematical model of a spacecraft with the assumption that the satellite is a rigid body, and so design a suitable attitude stabilization system by combining gravity gradient stabilization with magnetic torqueing. It will study magnetic coils which need to be added in order to improve the accuracy of attitude stabilization, and investigate, simulate and compare possible controller configurations (PID, LQR, FLC) in order to control the currents in the magnetic coils when these coils behave as the actuator of the system.

Chapter One gives a general introduction to define the terms used in the CubeSat field: CubeSat overview, CubeSat subsystems, the requirements of an attitude control system, orbit dynamics, orbital perturbations, components of attitude determination, and control systems. It also gives a survey of the available literature.

Chapter Two explains attitude stabilization of spacecraft and divides it to active and passive stabilization. The chapter presents gravity gradient stabilization, aerodynamic stabilization, and permanent magnets stabilization as passive stabilization methods, and spin stabilization and three-axis stabilization as active stabilization methods. In addition it describes modes of attitude stabilization of a spacecraft.

Chapter Three presents satellite attitude dynamic and modeling, introduces the equation of motion of a three-axis stabilized satellite under the effect of gravity gradient torque, and the magnetic actuators used with the proposed model.

Chapter Four describes the design of three controller configurations: Proportional–Integral–Derivative controller (PID), Linear Quadratic Regulator (LQR) and Fuzzy Logic Controller (FLC). Then these controller configurations are used to design the attitude control of three-axis stabilization of CubeSat. Chapter Five includes simulation of attitude control techniques and results for the three controller configurations (PID, LQR, and FLC) using Matlab. It makes a comparison between these control techniques and discusses the stability problem. Finally, all necessary mathematical details and all Simulink block diagrams, in addition to an initial values Matlab file, are presented in the appendices.

ACRONYMS AND ABBREVIATIONS

ADCS	Attitude determination and control system.
ADS	Attitude Determination System.
ACM	Attitude Control Maneuver
ACS	Attitude Control System.
AOCS	Attitude and Orbit Control System
ARE	Algebraic Riccati Equation
CG	Center of gravity.
ECI	Earth Centered Inertial
ECEF	Earth Centered Earth Fixed
FLC	Fuzzy logic controller.
GG	Gravity gradient
GPS	Global Positioning System.
IAGA	International Association of Geomagnetism and Aeronomy
IGRF	International Geomagnetic Reference Field.
LEO	Low Earth Orbit.
LQR	Linear Quadratic Regulator.
MIMO	Multi Input Multi Output.
MISO	Multi Input Single Output.
NED	North East Down.
PID	Proportional Integral Derivative.
SCO	Spacecraft Orbit.
SISO	Single Input Single Output.

CHAPTER ONE

CUBESAT OVERVIEW

1.1 Introduction

CubeSat is a nanosatellite, a type of spacecraft used by scientists and researchers for research. The standard dimensions of CubeSat are called Units (U). The dimensions of each unit are 10x10x11 cm. The size of the CubeSat can be 1U, 2U, 3U, or 6U, and typically the weight is less than 1.33 kg for each 1U. The CubeSat is deployed from a P-POD (Poly-Picosatellite Orbital Deployer). The CubeSat is designed at a miniature scale to reduce the cost of deployment. The missions of CubeSats are made for Low Earth Orbits (LEO), so that radiation can be ignored and commercial off-the-shelf electronics components can be used.

1.2 CubeSat systems

The various systems of a satellite are responsible for tasks that are necessary for proper functioning of the system. The typical nanosatellite bus consists of the following systems.

1.2.1 Electrical Power System

The main task of the electrical power system (EPS) is to provide the other subsystems with a reliable and continuous power source. The main components of this system are solar array, batteries and regulators, which lead to a conversion from solar energy to electrical power, energy storage in batteries, regulation of the electrical power, and distribution of the power to another subsystem.

The EPS is considered an essential part of the satellite because lack of power will mean an end to the mission. So it is important to have a stable and reliable power system. The required size of the solar panels and batteries depends on the requirements for the payload(s) and the lifetime of the mission. The EPS provides direct current (DC) power for all the subsystems on board the satellite. When the satellite is in direct sunlight the solar panels will be used, while when the Sun is eclipsed two Lithium Polymer batteries will be applied.

1.2.2 Attitude Determination and Control System

The attitude determination and control system (ADCS) is responsible for keeping the orientation of a spacecraft in space, in addition to achieving the required maneuver. Keeping the orientation of a spacecraft in space is called attitude stabilization. The attitude maneuver is the re-orientation process that changes one attitude to another. The ADCS collects data from the attitude sensors and processes it to determine the current attitude of the spacecraft.

The ADCS then compares the current attitude with the desired attitude and uses the difference between them, using a specified algorithm, to activate the appropriate actuators to remove or reduce the error.

1.2.3 Communications System

One of the main requirements of any satellite is the ability to communicate with the Earth reliably by sending and receiving data from the ground station. This data is the output of sensors, which give details of the health of the satellite, telemetry data which provide the orbital location of the satellite, and commands from the ground station to execute a specific program or function. The communication system also transmits periodically a Continuous Wave (CW) beacon, which carries telemetry data in Morse Code format.

Communication with the Earth can be established using a wide range of radio frequencies, depending on the data rate requirements, Earth station equipment costs, and Federal Communications Commission (FCC) licensing restrictions. The communication system consists of a transceiver which includes a transmitter, receiver, antenna, and terminal node controller (TNC). The TNC consists of a micro control unit, modem, EPROM and software that implement the AX.25 protocol.

1.2.4 Telemetry, Tracking, and Command System

The telemetry, tracking, and command (TT&C) system of a spacecraft provides the most important telecommunication link between it and its ground station. This system collects all data from sensors, and has processes to convert it to continuous data in order to transmit it to the ground station via the downlink. These telemetry data represent the health of the satellite, its orbital location, and the operational configuration data.

At the Earth station, the tracking system provides the information needed to compute orbital elements. The control system at the Earth station uses the data received from the satellite via the telemetry system and orbital data obtained from the tracking system to correct the antenna positioning and the configuration of the communication system. The command system sends commands from Earth to the spacecraft, via an uplink, to carry out specific actions like orbital maneuvers and antenna control.

1.2.5 Structure and Mechanisms System

The structure and mechanisms system is the backbone of the satellite. The structure provides an extremely strong and lightweight chassis for housing all satellite components. It comprises several mechanisms to deploy the solar panels, the boom, and the antennas. The structure must keep its form during all modes of the mission. During the launch, the frame must resist the force and vibration so as to hold the system components in place for proper operation.

The properties required in materials used for the spacecraft structure are stiffness, strength, thermal expansion, thermal conductivity, corrosion resistance and ease of fabrication, and there must be consideration of the materials' cost. The primary materials used for the structure and mechanisms system are aluminum alloys, heat-resistant steel, titanium, and composites.

1.2.6 Thermal Control System

The thermal control subsystem (TCS) keeps the temperatures of the payload and all components of the satellite within specified ranges by effective use of coatings, insulations, and radiators. The main external heat sources are direct sunlight, sunlight reflected from the Earth (albedo) and infrared radiated from the Earth.

TCS can be active, using heaters, coolers, temperature sensors, thermostats, and control electronics, or passive, using coatings, multi-layer blankets, louvres and fixed radiators; or both active and passive TCS may be needed.

1.3 Orbital Dynamics

Orbital dynamics is the study of the motions of artificial satellites and space vehicles moving under the influence of forces such as gravity, atmospheric drag, thrust, and so on. Johannes Kepler developed the first laws of planetary motion to predict the motion of the planets about the Sun or the path of a satellite about the Earth, and his theories were confirmed when Isaac Newton revealed his universal law of gravitation. These laws provide a good approximation of the path of a body in space mechanics.

Kepler's First Law: If two objects in space interact gravitationally, each will describe an orbit that is a conic section with the center of mass at one focus. If the bodies are permanently associated, their orbits will be ellipses; if they are not permanently associated, their orbits will be hyperbolas.

Kepler's Second Law: If two objects in space interact gravitationally (whether or not they move in closed elliptical orbits), a line joining them sweeps out equal areas in equal intervals of time.

Kepler's Third Law: If two objects in space revolve around each other due to their mutual gravitational attraction, the sum of their masses multiplied by the square of their period of mutual revolution is proportional to the cube of the mean distance between them. Hence

$$(m + M)P^2 = \frac{4\pi^2}{G}a^3 \tag{1-1}$$

where

P is their mutual period of revolution.

a is the mean distance between them.

m and M are the two masses.

G is Newton's gravitational constant.

Out of the two revolving objects, the one with the greatest mass is called the primary, and the less massive object is called the secondary. If the mass of the satellite is denoted m, and the mass of the Earth is denoted M, the mass of the satellite is considered negligible, $m+M \approx M$

Thus in the case of a satellite orbiting the Earth, it follows from Kepler's laws that the trajectory of the satellite is an ellipse with the center of the Earth at one focus.

1.3.1 Orbital Elements

There are six classical orbital elements (also known as Keplerian elements) that are necessary for us to know about an orbit and a satellite's place in it. These elements help us describe: orbit size, orbit shape, orbit orientation, and orbit location. They also specify the part of the Earth the satellite is passing over at any given time and its Field of View (FOV), which is the angle that describes the amount of the Earth's surface the spacecraft can see at any given time. These six orbital elements shown in Figure1-1 are (Sally Ride EarthKAM):

Semi-major Axis (a): Describes the size of the orbit, which is one-half of the major axis of the orbit.

Eccentricity (e): Specifies the shape of an orbit and is given by the ratio of the distance between the two foci and the length of the major axis. The eccentricity of a circular orbit is zero, and for an ellipse, it can range from zero to less than one.

Inclination (i): Angle between the plane of the equator and the orbital plane.

Right Ascension of the Ascending Node (Ω): It is the angle between the Sun and the intersection of the equatorial plane and the orbit on the first day of spring in the Northern Hemisphere. The day is called the vernal equinox. Looking down from above the North Pole, the right ascension of the ascending node is positive counter-clockwise.

Argument of Perigee (ω): Angle between the ascending node and the orbit's point of closest approach to the Earth (perigee).

True Anomaly (v): True Anomaly is one of three angular parameters ("anomalies") that define a position along an orbit, the other two being the eccentric anomaly and the mean anomaly. True Anomaly represents the angle between the perigee and the vehicle in the orbit plane.

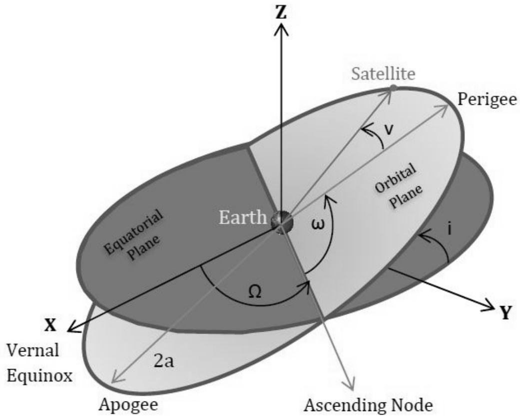

Fig. 1-1 Graphical Representation of Keplerian elements (European GNSS)

1.3.2 Two-Line Element (TLE)

A two-line element (TLE) is a special form of mean classical orbital elements that describe the orbit of an earth satellite. TLEs are generated with an orbit determination process based on observations by the United States Space Surveillance Network (SSN), which comprises a number of radar and electro-optical sensors. These elements are periodically updated so as to maintain a reasonable prediction capability on all space objects. The TLE is in a format specified by North American Aerospace Defense Command (NORAD) and used by NORAD and NASA. The TLE can be used directly by all simplified perturbations models (SGP, SGP4, SDP4, SGP8 and SDP8), which are used to calculate the orbital state vectors of satellites and space debris relative to the Earth-centered inertial coordinate system. Orbital elements are determined for many thousands of space objects by NORAD and are freely distributed on the Internet in the form of TLEs. Data for each satellite consists of three lines in the format shown in Figure 1-2.

```
TIGRISAT
1 40043U 14033AK 14237.65733135 .00000701 00000-0 11591-3 0 1958
2 40043 097.9710 131.0170 0062706 350.4599 009.5419 14.72239202 9677
```

Fig. 1-2 TIGRISAT Two Line Element

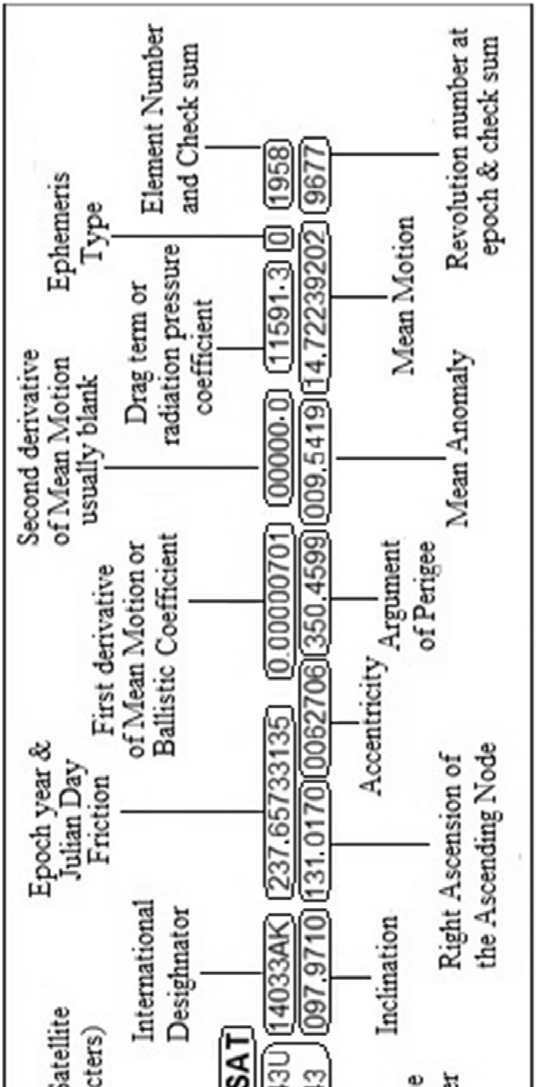

Fig. 1-3 TLE Parameters Explanation

Line 0 is a twenty-four character name (to be consistent with the name length in the NORAD Satellite Catalog SATCAT). Lines 1 and 2 are the standard Two-Line Orbital Element Set Format identical to that used by NORAD and NASA. The format description is as shown in Figure 1-3.

1.3.3 Earth Orbit Classification

Satellites rotate around the Earth in regular, repeating paths, and those paths are called orbits. The orbital path is controlled by two forces: the first is centrifugal force according to the satellite rotating velocity, and the second is the Earth's gravitational pull. The gravitational pull is according to Newton's law of universal gravitation. Centrifugal force and Earth gravitational pull balance each other so the satellite keeps its orbit. There are different satellite orbits that can be used. The satellite's application determines the suitable orbit. The widely used elevation categories are Low Earth Orbit (LEO), Medium Earth Orbit (MEO), Geostationary Orbit (GEO), and High Earth Orbit (HEO).

1.3.3.1 Low Earth Orbit (LEO)

Most satellites, the Space Shuttle, the International Space Station, and the Hubble Space Telescope are in Low Earth Orbit. A spacecraft in Low Earth Orbit is at an altitude between 160 and 2,000 km above the Earth's surface. It is not possible to keep objects below 160km because they would suffer from orbital decay and would quickly enter the atmosphere, which would cause crashing onto the surface or burning up. Orbital period at this altitude is between 88 and 127 minutes.

1.3.3.2 Medium Earth Orbit (MEO)

Medium Earth Orbit (MEO) is the region around the Earth above 2,000 kilometers altitude (low Earth orbit) and below 35,786 kilometers altitude (geostationary orbit). Navigation, geodetic/space environment science, and communication are the most common uses for satellites in this region. 20,200 kilometers with a 12 hours orbital period, which is used for the Global Positioning System (GPS), 19,100 kilometers which is used for Glonass constellations, and 23,222 kilometers which is used for Galileo constellations are the most common altitudes in this region. The orbital periods of MEO satellites range from about 2 to nearly 24 hours.

1.3.3.3 Geostationary Orbit

The Geostationary Orbit (GEO) is the most common orbit used for satellite communications. The rotational period of the geostationary orbit is equal to the rotational period of the Earth. It is an equatorial orbit with zero inclination located directly above the equator so the satellite appears fixed facing a point on the Earth. The advantages of the geostationary orbit are that the satellite can provide continuous operation in its field of view, in addition to there being no need to track it from the ground station.

1.3.3.4 High Earth Orbit (HEO)

High Earth Orbit puts the satellite outside the atmosphere and far away from the Earth. It is a geocentric orbit above 35,786 kilometers altitude (geostationary orbit) with orbital periods of more than twenty-four hours. The orbital velocity of the satellite in High Earth Orbit is lower than the Earth's rotational speed, which makes the ground track move westward on the Earth's surface. High Earth Orbit's typical uses are weather observation and space observation.

1.3.4 Orbital Perturbations

A satellite's orbit in an ideal two-body system describes a conic section or ellipse. In reality, there are several factors that cause the conic section to continually change. These deviations from the ideal Kepler's orbit are called perturbations.

Perturbations of the orbit are the results of various forces which are acting on a satellite that perturb it away from the nominal orbit. These perturbations, or variations in the orbital elements, can be classified based on how they affect the Keplerian elements. The principal sources of perturbations are Earth gravity harmonics (deviations from a perfect sphere), the lunisolar gravitational attractions, atmospheric drag, solar radiation pressure, and Earth tides. There are two types of orbital perturbations, gravitational when considering third-body (Sun/Moon) attractions and the non-spherical Earth, and non-gravitational like atmospheric drag, solar radiation pressure, outgassing (fuel tank leaks on the spacecraft), and the effect of tidal friction.

1.3.4.1 Gravitational Perturbations

The gravitational potential of the Earth is broadly symmetrical at a large scale but quite heterogeneous at smaller scales. This is because of the non-spherical shape of the planet, important variations in mass distribution (mountain peaks versus ocean dips) and density (rocks, water and air), as well as the displacement of masses within the system (in the Earth's core, in geophysical processes responsible for plate tectonics, or in atmospheric and oceanic currents, including tides for instance). For a circular orbit, the gravitational force is essentially perpendicular to the velocity vector, while the force that brings the satellite back down is a drag that acts in the opposite direction of the velocity vector.

a. Earth's Oblateness

The Earth is not a symmetrical body and seems to be flat at the poles with a bulge at the equator when compared to a perfect sphere. The difference in force due to the Earth's oblateness is referred to as the J_2 perturbation. This perturbation is the main gravitational force that acts on a satellite in LEO. The other types of gravitational perturbations, like third body effect, have a secondary effect in LEO compared to the other processes and will perturb the orbit, but the impact on the lifetime of the satellite will be minimal.

1.3.4.2 Non-gravitational Perturbations

The non-gravitational perturbations come from space environments such as solar radiation pressure, atmospheric drag, and geomagnetic field. At low altitude orbits, atmospheric drag and geomagnetic field are the principal non-gravitational perturbations acting on a satellite, while solar radiation pressure is the principal non-gravitational perturbation acting on a satellite at a high altitude orbit.

a. Atmospheric Drag

The primary factor that affects the satellite is the atmospheric drag, which itself depends on the atmospheric density and the form factor of the object flying into that atmosphere. Drag forces have an effect on a satellite's motion and change the orbit shape as a result of the presence of molecules of neutral gases in the Earth's upper atmosphere. Atmospheric drag acts in the opposite direction to the velocity and reduces the energy from the