# Attitude Control Final Report

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Documentation Statement:

# Abstract

To be completed with the project’s summary objectives, methods, and findings.

# Nomenclature

To be populated with the necessary symbols and definitions.

# Introduction

The objective of this project is to design and analyze an attitude control system for FalconSAT-9, a next-generation small satellite funded by the Air Force Research Laboratory (AFRL). The spacecraft will conduct advanced propulsion and space maneuvering experiments using an experimental Hall-effect thruster. FalconSAT-9 will be deployed into a 500 km altitude polar orbit as a secondary payload on a Falcon 9 launch vehicle. Upon deployment, the primary mission phases include detumbling the satellite and achieving precise orbit-fixed orientation during propulsion experiments.

The attitude determination and control system (ADCS) must counteract initial angular rates, environmental disturbances, and constant torques induced by propulsion misalignment. The system integrates multiple control modes to meet the spacecraft's mission objectives: Control Mode 0 focuses on detumbling, while Control Mode 1 ensures precise attitude during propulsion experiments. This report outlines the theoretical analysis, control design, and experimental validation necessary to meet these requirements.

# Theory

### Assumptions

The analysis assumes the satellite operates under idealized conditions, including:

1. A rigid-body spacecraft with mass properties derived from CAD body-fixed and principal frames.
2. The Earth is modeled as a perfect sphere with uniform density.
3. External disturbance torques include gravity gradient, atmospheric drag, and magnetic torques; solar pressure is negligible.

### Mathematical Techniques

First, the necessary calculations are made to obtain the spacecraft’s mass properties. We start by assuming the spacecraft consists of two parts: the body and the payload. The given physical parameters used for the following calculations are included in Appendix ##. The total center of mass (COM) of the spacecraft is derived by treating the components as point masses located at their respective COM positions in the CAD frame. We then multiply each component's mass by its position vector to compute its contribution to the total moment about the origin, sum these moments for all components, and divide by the total mass to determine the spacecraft’s overall COM.

|  |  |
| --- | --- |
|  | (1) |

Using the spacecraft’s COM, adjusting the offset position vectors by the COM vector. Using the given inertia matrices for the body and the payload, we apply the linearized parallel axis theorem,

|  |  |
| --- | --- |
|  | (2) |

The resulting moment of inertia matrices are then added together for the total moment of inertia of the satellite in the CAD frame.

|  |  |
| --- | --- |
|  | (3) |

Now we need a way to relate the CAD frame to the principal frame. To find the directional cosine matrix that relates the spacecraft's body frame to a principal frame, solve for the eigenvalues and eigenvectors of the inertia matrix in the body frame.

|  |  |
| --- | --- |
|  | (4) |

Arranging the eigenvectors as the columns of a 3x3 matrix such that the resulting matrix is close to identity gives the DCM  and the corresponding eigenvalues are , , and .

|  |  |
| --- | --- |
|  | (5) |

Taking the transpose of gives . Because any scalar multiple of an eigenvector is also an eigenvector corresponding to the same eigenvalue, multiply by -1 as necessary to get close to the identity matrix. Then, pre and post multiply by the DCM to get the moment of inertia matrix in the principal frame.

|  |  |
| --- | --- |
|  | (6) |
|  | (7) |

The equations of motion (EOM) for the spacecraft are derived in the orbital local vertical/local horizontal (LVLH) reference frame. Nonlinear EOMs account for gravity gradient torque and reaction wheel control, with the LVLH frame origin at the spacecraft's center of mass. These EOMs are linearized about the nominal LVLH orientation for state-space representation.

The linearized state-space form is defined as:

where A, B, C, D matrices represent system dynamics, input, output, and feedthrough, respectively. Input u includes reaction wheel torques, while state x includes angular velocities and attitude deviations.

### Control Mode 0

Micro-thrusters are modeled with on-off (bang-bang) dynamics to detumble the spacecraft. A relay control law using proportional-derivative (PD) gains stabilizes the pitch axis within two degrees of the desired orientation. The thruster delay is modeled as a 0.1-second actuator response time.

### Control Mode 1

Reaction wheels provide continuous 3-axis control for precise slewing and disturbance rejection. Control laws are designed to meet steady-state and transient performance requirements under misalignment-induced and environmental disturbance torques. The ADCS Kalman filter integrates sensor data, including sun sensors, magnetometers, and star trackers, to estimate the spacecraft's attitude in real-time.

Theoretical predictions validate the control modes against nonlinear simulations. State-space models and block diagrams ensure the system meets mission requirements without saturating actuator limits.

# Theoretical Predictions

This section will be completed with the outcomes of the theoretical development once the calculations are finalized.

# Experimental Results

This section will present experimental or computational findings. To be completed post-data collection.

# Discussion

This section will analyze the results, compare experimental data with theoretical predictions, and discuss discrepancies or sources of error.

# Conclusions and Recommendations

To be completed with a concise summary of findings and recommendations for future work.

# Appendices

Include supplementary materials, calculations, and MATLAB Simulink block diagrams here.