Attitude Determination and Control Subsystem (ADCS) Review
Contents

• Attitude Determination and Control Subsystem (ADCS) Function
• Impact of Mission Requirements and Other Subsystems on ADCS
• ADCS Design Process
• Spacecraft Coordinate Systems
• Spacecraft Dynamic & Kinematical Equations
• Attitude Determination (AD) System
• Key Design Parameters
• ROCSAT-2 Design Example
• Reference
ADCS Function

- The ADCS stabilizes the spacecraft and orients it in desired directions during the mission despite the external disturbance torques acting on it:
  - To stabilize spacecraft after launcher separation
  - To point solar array to the Sun
  - To point payload (camera, antenna, and scientific instrument etc.) to desired direction
  - To perform spacecraft attitude maneuver for orbit maneuver and payloads operation
- This requires that the spacecraft determine its attitude, using sensors, and control it, using actuators
Impact of Mission Requirements and Other Subsystems on ADCS

**Thermal**
- Special thermal maneuvers required?

**Command & Data Handling**
- Data processing capability

**Propulsion**
- 1. Thruster size
- 2. Propellant load

**Communications**
- Antenna pointing accuracy?

**Mission**
- 1. Earth-Pointing or Inertial-Pointing?
- 2. Control during ΔV burn?
- 3. Separate payload platform?
- 4. Accuracy/Stability needs?
- 5. Slewing requirement?
- 6. Orbit?
- 7. Autonomy?
- 8. Mission life?
- 9. On-board navigation data required?

**Power**
- 1. ADCS load
- 2. Special regulation
- 1. Solar array pointing requirement?

** Structures**
- 1. Center of mass constraints
- 2. Inertia constraints
- 3. Flexibility constraints
- 4. Thruster location
- 5. Sensor mounting

**ADCS Trades**

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PICOSAT ENGINEERING SYSTEM: ATTITUDE CONTROL

Department of Aerospace Engineering, TamKang University System Simulation and Avionics Laboratory

12/15/2005
## ADCS Design Process

<table>
<thead>
<tr>
<th>Step</th>
<th>Inputs</th>
<th>Outputs</th>
</tr>
</thead>
</table>
| 1a) Define *control mode*  
1b) Define or derive system-level *requirements* by control mode | •Mission requirements  
•Mission profile  
•Type of insertion for launch vehicle | •List of different control mode during mission  
•Requirement and constraints |
| 2) Select *type of spacecraft control* by attitude control mode | •Payload, thermal & power needs  
•Orbit, pointing direction  
•Disturbance environment | •Method for stabilizing and control: three-axis control, spinning, or gravity gradient |
| 3) Quantify *disturbance environment* | •Spacecraft geometry  
•Orbit/Solar/Magnetic models  
•Mission profile | •Environmental disturbances: forces/torques: Gravity gradient, magnetic, aerodynamics, solar pressure  
•Internal disturbances and powered flight effects on control (cg offsets, slosh) |

Continued on next page
4) Select & size **ADCS hardware**

- Spacecraft geometry
- Pointing accuracy
- Orbit conditions
- Mission requirements
- Life time
- Slew rate

**Sensor** suite: Earth, Sun, inertial, or other sensing devices

**Actuators**: reaction wheels, thrusters, or magnetic torquer

- Data processing requirements

5) Define determination and control Algorithms (Design/Analysis/Simulation)

- All of above

- Algorithms, parameters, and logic for each determination and control mode

6) Iterate and document

- All of above

- Refines requirements and design
- Subsystem specification
### Typical ADCS Modes

<table>
<thead>
<tr>
<th>Mode</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>Orbit Insertion</td>
<td>Period during and after boost while spacecraft is brought to final orbit. Options include no spacecraft control, simple spin stabilization of solid state rocket, and full spacecraft attitude control.</td>
</tr>
<tr>
<td>Acquisition</td>
<td>Initial determination of attitude and stabilization of spacecraft. Also may be used to recover from power upsets or emergencies.</td>
</tr>
<tr>
<td>Normal, On-Station</td>
<td>Used for the vast majority of the mission. Requirements for this mode should derive system design.</td>
</tr>
<tr>
<td>Slew</td>
<td>Reorienting the spacecraft when required.</td>
</tr>
<tr>
<td>Contingency, or Safe</td>
<td>Used in emergencies if regular mode fails or is disabled. May use less power or sacrifice normal operation to meet power or thermal constraints.</td>
</tr>
<tr>
<td>Special</td>
<td>Requirement may be different for special targets or time period, such as eclipse.</td>
</tr>
</tbody>
</table>
### Typical ADCS Requirements

<table>
<thead>
<tr>
<th>Determination</th>
<th>Definition</th>
<th>Examples/Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>Accuracy</td>
<td>How well a vehicle’s orientation with respect to an absolute reference is known</td>
<td>0.25 deg, 3-sigma, all axes; may be real-time or post-processed on the ground</td>
</tr>
<tr>
<td>Range</td>
<td>Range of angular motion over which accuracy must be met</td>
<td>Any attitude within 30 deg of nadir</td>
</tr>
</tbody>
</table>

Continued on next page
<table>
<thead>
<tr>
<th>Control</th>
<th>Definition</th>
<th>Examples/Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>Accuracy</td>
<td>How well the vehicle attitude can be controlled with respect to a commanded direction</td>
<td>0.25 deg, 3-sigma, including determination and control errors, may be taken with respect to an inertial or Earth-fixed reference</td>
</tr>
<tr>
<td>Range</td>
<td>Range of angular motion over which control performance must be met</td>
<td>All attitude, within 50 deg of nadir, within 20 deg of Sun</td>
</tr>
<tr>
<td>Jitter</td>
<td>A specified angle bound or angular rate limit on short-term, high-frequency motion</td>
<td>0.1 deg /min, 1deg/s, 1 to 20 Hz; usually specified to keep spacecraft motion from blurring sensor data</td>
</tr>
<tr>
<td>Drift</td>
<td>A limit on slow, low-frequency vehicle motion. Usually expressed as angle/time</td>
<td>1 deg/hr, 5deg max. Used when vehicle may drift off target with infrequent resets</td>
</tr>
<tr>
<td>Setting Time</td>
<td>Specifies allowed time to recover from maneuvers or upsets</td>
<td>2 deg max motion; may be used to limit overshoot or nutation</td>
</tr>
</tbody>
</table>
• **Attitude Control Type**
  – Passive Control Techniques:
    • **Gravity-gradient control** uses the inertial properties of a spacecraft to keep it pointed toward the Earth. This relies on the fact that an elongated object in the gravity field tends to align its longitudinal axis through the Earth’s center.
    • **Passive magnetic control** uses permanent magnets on board the spacecraft to force alignment along the Earth’s magnetic field. This is most effective in near-equatorial orbits where the field orientation stays almost constant.
    • **Spin stabilization** is a passive control technique in which the entire spacecraft rotates so that its angular momentum vector remains approximately fixed in inertial space. Spin-stabilized spacecraft employ the gyroscopic stability to passively resist disturbance torques about two axes.
- Three-axis Active Control Technique:
  - Spacecraft stabilized in three axes are more common today than those using passive control.
  - It can be stable and accurate but also more expensive, complex, and potentially less reliable.
  - Broadly, these system take two forms: one uses momentum bias by placing a momentum wheel along the pitch axis; the other is called zero momentum with a reaction wheel on each axis. Either option usually need thrusters or magnetic torquers for wheel momentum unloading.
### ADCS Control Methods and their Capabilities

<table>
<thead>
<tr>
<th>Type</th>
<th>Pointing Options</th>
<th>Attitude Maneuverability</th>
<th>Typical Accuracy</th>
<th>Lifetime Limit</th>
</tr>
</thead>
<tbody>
<tr>
<td>Gravity Gradient</td>
<td>Earth local vertical only</td>
<td>Very limited</td>
<td>± 5° (2 axes)</td>
<td>No limit</td>
</tr>
<tr>
<td>Passive Magnetic</td>
<td>North/South only</td>
<td>Very limited</td>
<td>± 5° (2 axes)</td>
<td>No limit</td>
</tr>
<tr>
<td>Spin Stabilization</td>
<td>Inertially fixed (any direction)</td>
<td>High propellant usage to move stiff momentum vector</td>
<td>± 0.1° ~ ± 1.0° (proportional to spin rate)</td>
<td>Propellant</td>
</tr>
<tr>
<td>Bias Momentum (1 wheel)</td>
<td>Best suited for Earth local vertical pointing</td>
<td>Momentum vector of the bias wheel prefers to stay normal to orbit plane</td>
<td>± 0.1° ~ ± 1.0°</td>
<td>Propellant (if applies) Life of sensor and wheel bearing</td>
</tr>
</tbody>
</table>

Continued on next page
<table>
<thead>
<tr>
<th>Zero Momentum Bias (Thruster)</th>
<th>No constraints</th>
<th>No constraints</th>
<th>± 0.1° ~ ± 5.0°</th>
<th>Propellant</th>
</tr>
</thead>
<tbody>
<tr>
<td>Zero Momentum Bias (Three wheels)</td>
<td>No constraints</td>
<td>No constraints</td>
<td>± 0.001° ~ ± 1.0°</td>
<td>Life of sensor and wheel bearing</td>
</tr>
</tbody>
</table>
Control Example: Active Attitude Control System

General Structure of a Satellite Attitude Determination and Control Subsystem
• Typical Environmental Disturbance
  • Gravity-Gradient Torque:
    – Any nonsymmetrical object of finite dimensions in orbit is subject to a gravitational torque because of the variation in the Earth’s gravitational force over the object
  • Solar Radiation Torque:
    – Radiation incident on a spacecraft’s surface produces a force which results in a torque about the spacecraft’s center of mass
  • Aerodynamic Torque
    – The interaction of the upper atmosphere with a spacecraft’s surface produces a torque about the center of mass
• Magnetic Disturbance Torque
  – Magnetic disturbance torques result from the interaction between the spacecraft’s residual magnetic field and the geomagnetic field
Gravity Gradient Torque

\[ \tau_{gg} = \frac{3\mu}{\gamma^3} \left[ \gamma_b \times (I \cdot \gamma_b) \right] \]

where
- \( \mu \): Earth's gravitational constant
- \( \gamma_b \): Geocentric position to the spacecraft's center of mass
- \( \gamma \): The size of \( \gamma_b \)
- \( I \): The moment of inertia
Solar Radiation Torque

\[ \tau_{solar} = \sum r_s \times f_{solar} \]

\[ f_{solar} = PA(\overrightarrow{N} \cdot \overrightarrow{S}) \left[ (Ca + Cd)\overrightarrow{S} + \frac{2}{3}Cd + 2Cs(\overrightarrow{N} \cdot \overrightarrow{S})\overrightarrow{N} \right] \]

\[ Ca = \sigma_A \quad 0 \leq \sigma_A \leq 1 \]
\[ Cd = \sigma_D(1-\sigma_A) \quad 0 \leq \sigma_D \leq 1 \]
\[ Cs = (1-\sigma_D)(1-\sigma_A) \]

where
- \( P \): Mean momentum flux
- \( \overrightarrow{S} \): Unit vector from the sun to the spacecraft
- \( Ca \): Absorption coefficient
- \( Cd \): The coefficient of diffuse reflection
- \( Cs \): The coefficient of specular reflection
- \( \sigma_A \): Absorption coefficient
- \( \sigma_D \): Diffuse reflectivity coefficient
Aerodynamic Torque

\[ \tau_{aero} = \sum r_s \times f_{aero} \]

\[ f_{aero} = \frac{1}{2} C_D \rho V^2 A (\overrightarrow{N} \cdot \overrightarrow{V}) f_T V + (2 - f_T - f_N) (\overrightarrow{N} \cdot \overrightarrow{V}) \overrightarrow{N} \]

where

- \( C_D \): Drag coefficient
- \( \rho \): Atmospheric density
- \( V \): Speed of the impinging particle flow
- \( \overrightarrow{V} \): Unit vector in the direction of the impinging particle flow
- \( \overrightarrow{N} \): Normal unit vector
- \( A \): Surface areas
- \( f_T \): Tangential accommodation coefficient
- \( f_N \): Normal accommodation coefficient
Magnetic Torque

\[ \tau_{mag} = m \times B \]

\[ B = -\nabla V \]

\[ V(r, \theta, \phi) = a \sum_{n=1}^{k} \left( \frac{a}{r} \right)^{n+1} \sum_{m=1}^{n} \left( g^m_n \cos m\phi + h^m_n \sin m\phi \right) P^m_n(\theta) \]

where

- \( m \): The residual spacecraft magnetic dipole moment
- \( B \): The magnetic field vector
- \( g^m_n, h^m_n \): Gaussian coefficients
- \( a \): The equatorial radius of the Earth
- \( r, \theta, \phi \): The geocentric distance, coelevation, and East longitude from Greenwich
- \( P^m_n \): Legendre function
### Typical ADCS Sensors

<table>
<thead>
<tr>
<th>Sensors</th>
<th>Typical Performance</th>
<th>Weight (kg)</th>
<th>Power (Watt)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Gyro</td>
<td>Drift rate: 0.003°/hr ~ 1°/hr</td>
<td>3 ~ 25</td>
<td>10 ~ 200</td>
</tr>
<tr>
<td>Sun sensor</td>
<td>Accuracy: 0.005° ~ 3°</td>
<td>0.5 ~ 2</td>
<td>0 ~ 3</td>
</tr>
<tr>
<td>Star sensor</td>
<td>Accuracy: 0.0003° ~ 0.01°</td>
<td>3 ~ 7</td>
<td>5 ~ 20</td>
</tr>
<tr>
<td>Horizon sensor</td>
<td>Accuracy: 0.1° ~ 1°</td>
<td>2 ~ 5</td>
<td>0.3 ~ 10</td>
</tr>
<tr>
<td>Magnetometer</td>
<td>Accuracy: 0.5° ~ 3°</td>
<td>0.6 ~ 1.2</td>
<td>0 ~ 1</td>
</tr>
</tbody>
</table>
### Typical ADCS Actuators

<table>
<thead>
<tr>
<th>Actuators</th>
<th>Typical Performance</th>
<th>Weight (kg)</th>
<th>Power (Watt)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thrusters</td>
<td></td>
<td>Variable</td>
<td>N/A</td>
</tr>
<tr>
<td>• Hydrazine</td>
<td>0.5 ~ 9000 N*</td>
<td>Variable</td>
<td>N/A</td>
</tr>
<tr>
<td>• Cold gas</td>
<td>&lt; 5</td>
<td>Variable</td>
<td>N/A</td>
</tr>
<tr>
<td>Reaction Wheel</td>
<td>0.4 ~ 400 Nms (momentum)</td>
<td>2 ~ 20</td>
<td>10 ~ 110</td>
</tr>
<tr>
<td></td>
<td>0.01 ~ 1 Nm (torque)</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Control moment gyro</td>
<td>25 ~ 500 Nm (torque)</td>
<td>&gt; 40</td>
<td>90 ~ 150</td>
</tr>
<tr>
<td>Magnetic torquer</td>
<td>1 ~ 4000 Am²</td>
<td>0.4 ~ 50</td>
<td>0.6 ~ 16</td>
</tr>
</tbody>
</table>

* Multiply by moment arm (typical 1 or 2 m) to get torque
Spacecraft Coordinate Systems

1. Spacecraft (ROCSAT-2) Coordinate System

2. Euler Angle Definition
Earth Centered Inertial (ECI) Coordinate System

$X_{ECI}$: the rotation axis of the Earth

ECI is a inertial fixed coordinate system
Local Vertical Local Horizontal (LVLH) Coordinate System

LVLH is not a inertial fixed coordinate system
**Spacecraft Attitude Definition**

– Spacecraft Attitude: the orientation of the body coordinate w.r.t. the ECI (or LVLH) coordinate system

– Euler angle representation:

  • \[ \phi \theta \phi \] : rotate \( \phi \) angle around \( X \)-axis, then rotate \( \theta \) angle around \( Y \)-axis, then rotate \( \phi \) angle around \( Z \)-axis

  • The transition matrix:

\[
R_{123}(\phi, \theta, \phi) = \begin{bmatrix}
\cos \phi & \sin \phi & 0 \\
-\sin \phi & \cos \phi & 0 \\
0 & 0 & 1
\end{bmatrix} \begin{bmatrix}
\cos \theta & 0 & -\sin \theta \\
0 & 1 & 0 \\
\sin \theta & 0 & \cos \theta
\end{bmatrix} \begin{bmatrix}
1 & 0 & 0 \\
0 & \cos \phi & \sin \phi \\
-\sin \phi & \cos \phi & 0
\end{bmatrix}
\]
 Quaternion: rotation an angle ($\theta$) around arbitrary axis $(N)$ of ECI coordinate system:

- $Q = [ n_1 \times \sin(\theta/2) \ n_2 \times \sin(\theta/2) \ n_3 \times \sin(\theta/2) \ \cos(\theta/2) ]$
- $N \equiv [ n_1 \ n_2 \ n_3 ]$
Spacecraft Dynamic & Kinematical Equations

- Dynamic Equation (3x1) to be stabilized:
  - $I$: Spacecraft moment of inertia
  - $N$: External torque (Thruster & Environment)
  - $H_w$: Wheel angular momentum
  - $w$: Spacecraft angular velocity

\[
I \frac{dw}{dt} = N - \frac{dH_w}{dt} - w \times (Iw + H_w)
\]
• Kinematical Equation (4x1):
  – Q: Spacecraft quaternion

\[
\dot{\mathbf{Q}} = \frac{1}{2} \mathbf{w} \otimes \mathbf{Q}
\]

\[
\mathbf{Q} = \begin{bmatrix}
q_1 & q_2 & q_3 & q_4
\end{bmatrix}^T
\]

\[
\dot{\mathbf{Q}} = \frac{1}{2} \begin{bmatrix}
0 & w_z & -w_y & w_x \\
-w_z & 0 & w_x & w_y \\
w_y & -w_x & 0 & w_z \\
-w_x & -w_y & -w_z & 0
\end{bmatrix}
\begin{bmatrix}
q_1 \\
q_2 \\
q_3 \\
q_4
\end{bmatrix}
\]
• Position/Velocity Propagation Equation (6x1):

\[
\begin{bmatrix}
\dot{P}_x \\
\dot{P}_y \\
\dot{P}_z \\
\dot{V}_x \\
\dot{V}_y \\
\dot{V}_z
\end{bmatrix} =
\begin{bmatrix}
0 & 0 & 0 & 1 & 0 & 0 \\
0 & 0 & 0 & 0 & 1 & 0 \\
0 & 0 & 0 & 0 & 0 & 1 \\
0 & 0 & 0 & 0 & 0 & 0 \\
0 & 0 & 0 & 0 & 0 & 0 \\
0 & 0 & 0 & 0 & 0 & 0
\end{bmatrix}
\begin{bmatrix}
P_x \\
P_y \\
P_z \\
V_x \\
V_y \\
V_z
\end{bmatrix} +
\begin{bmatrix}
0 \\
0 \\
0 \\
0 \\
0 \\
a_x
\end{bmatrix}
\]

• Earth Gravity Model:

\[
V(r, \theta, \phi) = \frac{GM}{a} \left\{ \sum_{n=0}^{\infty} \sum_{m=0}^{\infty} \left( \frac{R_e}{r} \right)^{n+1} P^{\mu\nu}(\sin \theta) [C^{nm} \cos m \phi + S^{nm} \sin m \phi] \right\}
\]

\[
a = -\nabla V_e
\]
Attitude Determination (AD) System

\[ x_k = \Phi_{k-1} x_{k-1} + w_{k-1} \]
\[ w_k = N(0, Q_k) \]

Propagation Equation

\[ z_k = H_k x_k + v_k \]
\[ v_k = N(0, R_k) \]

Measurement Equation

\[ \hat{x}_k (-) = \Phi_{k-1} \hat{x}_{k-1} (+) \]
\[ P_k (-) = \Phi_{k-1} P_{k-1} (+) \Phi_{k-1}^T + Q_{k-1} \]

State propagate:

Error covariance propagate:

State update:
\[ \hat{x}_k (+) = \hat{x}_k (-) + K_k [z_k - H_k \hat{x}_k (-)] \]
\[ P_k (+) = [I - K_k H_k] P_{k-1} \]

Error covariance update:

Kalman gain matrix:
\[ K_k = P_k (-) H_k^T [H_k P_k (-) H_k^T + R_k]^{-1} \]

\( x_k \): state  
\( w_k \): propagation noise  
\( v_k \): measurement noise

For a navigation system (not AD):
State is spacecraft position/velocity  
Earth gravity model is the input of the propagation equation  
GPS receiver’s output (spacecraft position/velocity) is the measurement equation
## Key Design Parameters

### Control Accuracy on Hardware Selection and ADCS Design

<table>
<thead>
<tr>
<th>Required Accuracy (3σ)</th>
<th>ADCS Design</th>
<th>Hardware Selection</th>
</tr>
</thead>
</table>
| > 5°                   | •Permits major cost saving  
|                        | •Permit gravity gradient (GG) stabilization | •No sensors required for GG stabilization |
| 1° ~ 5°                | •Gravity gradient not feasible  
|                        | •Spin stabilization feasible  
|                        | •Three-axis stabilization will work | •Sensor: sun sensor & horizon sensor  
|                        |                                      | •Actuator: Thruster dead band control or wheel control  
|                        |                                      | •Magnetic torquer (and magnetometer) useful |
| 0.1° ~ 1°              | •Three-axis stabilization an momentum-bias stabilization feasible | •Sensor: star tracker or horizon sensor and gyros  
|                        |                                      | •Actuator: reaction wheel  
|                        |                                      | •Magnetic torquers feasible on light spacecraft  
|                        |                                      | (magnetometer also required) |
| < 0.1°                 | •Three-axis stabilization is necessary | •Need star tracker and better class of gyros  
|                        |                                      | •Control laws and computational needs are more complex  
|                        |                                      | •Flexible body performance is very importance |
### Slewing Requirements that Affect Actuator Selection

<table>
<thead>
<tr>
<th>Slewing</th>
<th>Effect on Spacecraft</th>
<th>Effect on ADCS Design</th>
</tr>
</thead>
</table>
| None             | • Spacecraft constrained to one attitude --- highly improbable | • Reaction wheel can be smaller  
• Magnetic torquer used for dumping wheel momentum |
| Nominal rate     | • Minimal                                                 | • Thruster or reaction wheel are adequate                                           |
| (0.05°/s ~ 0.5°/s) |                                                           |                                                                                      |
| High rate        | • Weight and cost increase                               | • Control moment gyros or,  
• Two thruster force levels --- one for stationkeeping and one for high rate maneuvers |
| (> 0.5°/s)       |                                                           |                                                                                      |
ROCSAT-2 Design Example

- **Mission Goals**
  - The primary goals of this mission are remote sensing applications for natural disaster evaluation, agriculture application, urban planning, environmental monitoring, and ocean surveillance over Taiwan area and its surrounding oceans.
  - In addition, the spacecraft will carry a scientific instrument, the ISUAL, for upper atmospheric lightning researches.
• Mission Requirements:
  – \( \Delta V \) is needed to bring spacecraft from the parking orbit to the mission orbit
  – Inertial-pointing during daylight for charging solar array
  – Earth-pointing during eclipse for ISUAL operation
  – Remote sensing instrument operation over Taiwan
  – A safe mode shall be engaged automatically when ever loss of spacecraft attitude is detected
  – Pointing Accuracy: the spacecraft shall provide pointing accuracy within 2 km in nadir direction
– **Pointing Knowledge**: the spacecraft shall provide pointing knowledge with geo-location error less than 450 m

– **Position knowledge**: the spacecraft on-board position knowledge shall be better than 100 m

– **Slewing Requirement**: 10° roll-axis maneuver in 25 s, 30° roll-axis maneuver in 60 s, and 45° pitch-axis maneuver in 60 s
• Other systems requirements to ADCS
  – Power subsystem:
    • Energy balance
  – Communication subsystem:
    • None: the communication subsystem own two hemispherical antennas in opposite direction
  – Thermal subsystem:
    • No special maneuver is needed for thermal
• ADCS Modes

– The Acquisition and Safe Hold (ASH) Mode:
  • It ensures the reduction of the initial angular rates, impacted to the spacecraft at launcher separation or by a severe failure (e.g. of propulsion), by magnetic damping
  • The steady state phase of ASH is a combination of Sun pointing, on the illuminated part of the orbit, and of geomagnetic pointing during the eclipse
  • This mode is robust: only few pieces of equipment with high reliability are used (magnetometer, coarse Sun sensor, reaction wheels and magneto-torquers
– The Normal Mode (NM):
  • A three-axis attitude control is performed with four reaction wheels, which are off-loaded with three magneto-torquers
  • Attitude estimation is performed by mitigation of star tracker measurements and gyroscopes measurements
  • Position and velocity are estimated by the Navigation function, which relies on GPS measurements
– The Orbit Control Mode (OCM):
  • The OCM is for transfer between parking orbit and operational one, and for orbit maintenance
ASH Spacecraft Attitude

Rotation at two times the orbital rate during eclipse

Sun-Pointing Phase

Eclipse limit

Sun Direction
**NM Spacecraft Attitude**

- Sun-Pointing during Daylight
- Earth-Pointing during Eclipse
- Eclipse limit
- Sun Direction

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OCM Spacecraft Attitude

- Earth-pointing (orbit altitude is adjusting only)
- Eclipse limit

Sun Direction

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Spacecraft Control Method

- Acquisition and Safe Hold Mode:
  - The ASH is based on the use of a very simple magnetic control law (non-linear control law), known as the “B-dot law”.
  - ASH performs rate reduction and Sun acquisition using only simple equipment: magnetometer, coarse sun sensor, magnetic-torquers, and wheels. Neither thrusters nor gyroscopes are used.
  - ASH is only one mode to perform initial attitude acquisition, after launcher separation, and attitude recovery, following an Attitude Reconfiguration Order (ARO)
– Normal Mode:

• Three-axis active control, zero momentum bias, and PD controller.
• Magnetic torquer for wheel momentum unloading
• Orbit Determination: GPS receiver combined with a Kalman filter
• Attitude Determination: IRU, Star Tracker combined with a Kalman filter

– Orbit Control Mode:

• Same as the NM
• PID controller for the thruster controlled axes
### ADCS Hardware

<table>
<thead>
<tr>
<th>ADCS Mode</th>
<th>Magneto-meter</th>
<th>Sun Sensor</th>
<th>Magnetic Torquer</th>
<th>Reaction Wheel</th>
<th>IRU</th>
<th>Star Tracker</th>
<th>GPS Receiver</th>
<th>Thruster</th>
</tr>
</thead>
<tbody>
<tr>
<td>ASH</td>
<td>X</td>
<td>X</td>
<td>X</td>
<td>X</td>
<td>(X)</td>
<td>(X)</td>
<td>(X)</td>
<td></td>
</tr>
<tr>
<td>NM</td>
<td>X</td>
<td>X</td>
<td>X</td>
<td>X</td>
<td>X</td>
<td>X</td>
<td>X</td>
<td></td>
</tr>
<tr>
<td>OCM</td>
<td>X</td>
<td>X</td>
<td>X</td>
<td>X</td>
<td>X</td>
<td>X</td>
<td>X</td>
<td>X</td>
</tr>
</tbody>
</table>

X: Hardware used  
(X): Hardware used only for transition to NM
ASH Architecture

ADCS Hardware Models (Wheel & Torquers)

- Pitch Momentum Bias
- Magnetic Coil
  - H/W Model
- Wheel
  - H/W Model
- Environmental Disturbance
  - Td: Disturbance Torque
  - 1. Q: Attitude
  - 2. Ω: Angular Rate
- S/C Dynamics & Earth Gravity Model
  - P: S/C position
  - T: Julian Date

ADCS Hardware Models (Sun Sensor & Magnetometer)

- Magnetometer
  - H/W Model
- Earth Magnetic Field Model
- Sun Sensor
  - H/W Model
- Sun Vector Model

Flight Software

- Low-pass Filter
- B-dot Control Law
- PD Control Law

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NM Architecture

```
IRU

Star Tracker

GPS Receiver

Attitude estimation

Q_{body/eci}

Sub-mode manager

Commanded quaternion

NORMAL Mode Control:
- pointing error \( \Theta_x, \Theta_y, \Theta_z \)
- reaction Wheel Command
- magnet-torquer Command

Wheels
(Attitude Control)

Magnetic Torquers
(Wheel Momentum Off-loading)

Orbit estimation

Q_{ivlh/eci}

Telecommand
```

---

**PICOSAT ENGINEERING SYSTEM : ATTITUDE CONTROL**

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**OCM Architecture**

- **Telecommand**
- **Commanded quaternion**
- **Wheels** (Attitude Control)
- **Thrusters** (Attitude Control & Orbit Maneuver)
- **Magnetic Torquers** (Wheel Momentum Off-loading)

**Attitude estimation**

- $Q_{body/eci}$
- OCM Control:
  - pointing error $\Theta_x, \Theta_y, \Theta_z$
  - reaction Wheel Command
  - magnet-torquer Command
  - Thruster On/Off Command

**Orbit estimation**

- $Q_{lvlh/eci}$

**Inputs**

- IRU
- Star Tracker
- GPS Receiver
References