Spacecraft Attitude Control
Space System Design, MAE 342, Princeton University
Robert Stengel

- More on Rotation Matrices
  - Direction cosine matrix
  - Quaternions
- Yo-yo De-Spin
- Continuously Variable Torque Controllers
- On/Off-Torque Controllers

Attitude Control System

![Attitude Control System Diagram]

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http://www.princeton.edu/~stengel/MAE342.html
UARS Attitude Control System

Spacecraft Attitude Control Inputs

- On-Board Sensors
  - Inertial Measurements
    - Accelerometers
    - Angle sensors
    - Angular-rate sensors
  - Optical Sensors
    - Star sensors
    - Sun sensors
    - Horizon sensors
- Off-Board Observations
  - Ground-Based Tracking
    - Radar
    - Navigation beacons (VOR/DME, LORAN, ...)
  - Spaced-Based Tracking
    - GPS, GLONASS, ...
Potential Accuracies of External Attitude Measurements

<table>
<thead>
<tr>
<th>Reference object</th>
<th>Potential accuracy</th>
</tr>
</thead>
<tbody>
<tr>
<td>Stars</td>
<td>1 arc second</td>
</tr>
<tr>
<td>Sun</td>
<td>1 arc minute</td>
</tr>
<tr>
<td>Earth (horizon)</td>
<td>6 arc minutes</td>
</tr>
<tr>
<td>RF beacon</td>
<td>1 arc minute</td>
</tr>
<tr>
<td>Magnetometer</td>
<td>30 arc minutes</td>
</tr>
<tr>
<td>Narstar Global Positioning System (GPS)</td>
<td>6 arc minutes</td>
</tr>
</tbody>
</table>

Note: This table gives only a guideline. The GPS estimate depends upon the ‘baseline’ used (see text).

Spacecraft Attitude Control Outputs

• **Continuous Control Torques**
  – Control Moment/Reaction Wheel Gyros
  – Magnetic Torquers
  – Solar Panels

• **Pulsed Control Torques**
  – Reaction Control Thrusters (RCS)

• **One-Shot Devices**
  – RCS Spin-up
  – Yo-Yo De-Spin
Spacecraft Attitude Disturbances

• External Torques
  – Solar radiation pressure
  – Gravity gradient
  – Magnetic fields
  – Aerodynamics
  – Can be put to good use if related to attitude control objectives

• Vehicle-Based Torques
  – Mass movement
  – Elasticity
  – Out-gassing

More on Rotation
Matrices and Quaternions
**Direction Cosine Matrix**

- Cosines of angles between each $I$ axis and each $B$ axis
- Projections of vector components in one frame on the other

$$ H^B_I = \begin{bmatrix} \cos \delta_{11} & \cos \delta_{21} & \cos \delta_{31} \\ \cos \delta_{12} & \cos \delta_{22} & \cos \delta_{32} \\ \cos \delta_{13} & \cos \delta_{23} & \cos \delta_{33} \end{bmatrix} $$

$$ r_B = H^B_I r_I $$

**Euler’s Formula**

- Rotation from one axis system, $I$, to another, $B$, represented by
  - Orientation of axis vector about which the rotation occurs (3 parameters of a unit vector, $a_1$, $a_2$, and $a_3$)
  - Magnitude of the rotation angle, $\phi$, rad

$$ r_B = H^B_I r_I = \cos \phi r_I + (1 - \cos \phi)(a^T r_I) a - \sin \phi (a \times r_I) $$

$$ a = \begin{bmatrix} a_1 \\ a_2 \\ a_3 \end{bmatrix}^T $$

$$ (a^T r_I) a = (aa^T) r_I $$

$$ H^B_I = \cos \phi + (1 - \cos \phi)aa^T - \sin \phi \hat{a} $$

*Pisacane, 2005*
Quaternion Derived from Euler Rotation Angle and Orientation

\[ q = \begin{bmatrix} q_1 \\ q_2 \\ q_3 \\ q_4 \end{bmatrix} \overset{\Delta}{=} \begin{bmatrix} \mathbf{a}_\phi \\ q_4 \end{bmatrix} = \begin{bmatrix} \sin(\phi/2) \\ a_1 \\ a_2 \\ a_3 \end{bmatrix} \begin{bmatrix} \cos(\phi/2) \end{bmatrix} \]

- Quaternion vector
  - 4 parameters based on Euler’s formula
- Is not singular at \( \theta = \pm 90^\circ \)
- 4-parameter representation of 3 parameters; hence, it requires a constraint

\[
q^Tq = q_1^2 + q_2^2 + q_3^2 + q_4^2 = \sin^2(\phi/2) + \cos^2(\phi/2) = 1
\]

Rotation Matrix Expressed with Quaternion

From Euler’s formula

\[
H^B_I = \left[q_4^2 + (q^Tq)I_3 + 2qq^T - 2q_4\tilde{q}\right]
\]

\[
H^B_I = \begin{bmatrix}
q_1^2 - q_2^2 - q_3^2 + q_4^2 & 2(q_1q_2 + q_3q_4) & 2(q_1q_3 - q_2q_4) \\
2(q_1q_2 - q_3q_4) & -q_1^2 + q_2^2q_3^2 + q_4^2 & 2(q_2q_3 + q_1q_4) \\
2(q_1q_3 + q_2q_4) & 2(q_2q_3 - q_1q_4) & -q_1^2 - q_2^2 + q_3^2 + q_4^2
\end{bmatrix}
\]

Pisacane, 2005
Quaternion Expressed from Elements of Rotation Matrix

\[ q_4 = \frac{1}{2} \sqrt{1 + h_{11} + h_{22} + h_{33}} \]

Assuming that \( q_4 \neq 0 \)

\[
\mathbf{a}_\phi = \begin{bmatrix} q_1 \\ q_2 \\ q_3 \\ q_4 \end{bmatrix} = \frac{1}{4q_4} \begin{bmatrix} (h_{23} - h_{32}) \\ (h_{31} - h_{13}) \\ (h_{12} - h_{21}) \end{bmatrix}
\]

Successive Rotations Expressed by Products of Quaternions and Rotation Matrices

Rotation from Frame A to Frame C through Intermediate Frame B

\[
\mathbf{q}_A \mathbf{q}_B \mathbf{q}_C = \mathbf{H}_B^C \left( \mathbf{q}_C^B \right) \mathbf{H}_A^B \left( \mathbf{q}_A^B \right)
\]

Matrix Multiplication Rule

\[
\mathbf{q}_A = \begin{bmatrix} q_1 \\ q_2 \\ q_3 \\ q_4 \end{bmatrix}_A = \mathbf{q}_C^B \mathbf{q}_A^B \triangleq \begin{bmatrix} (q_4)_B^C (q_4)_A^B + (q_4)_A^B q_4^C - q_4^C q_4^B \\ (q_4)_B^C (q_4)_A^B - (q_4)_B^C q_4^B \end{bmatrix}
\]

Quaternion Multiplication Rule

\[
\mathbf{H}_A \left( \mathbf{q}_A^B \right) = \mathbf{H}_B \left( \mathbf{q}_B^C \right) \mathbf{H}_A \left( \mathbf{q}_A^B \right)
\]

Pisacane, 2005
Quatetion Vector Kinematics

ODE is linear in both $q$ and $\omega_B$

\[
\dot{q} = \frac{d}{dt} \begin{bmatrix} a_4 \\ q_4 \end{bmatrix} = \frac{1}{2} \begin{bmatrix} q_4 \omega_B - \tilde{\omega}_B a_4 \\ -\omega_B^T a_4 \end{bmatrix}
\]

Propagate Quaternion Vector

Digital integration to compute $q(t_k)$

\[
q_{\text{int}}(t_k) = q(t_{k-1}) + \int_{t_{k-1}}^{t_k} \frac{dq(\tau)}{dt} d\tau
\]

Normalize $q(t_k)$ to enforce constraint

\[
q(t_k) = q_{\text{int}}(t_k) / \sqrt{q_{\text{int}}^T(t_k) q_{\text{int}}(t_k)}
\]
Quaternion Interface with Euler Angles

- Quaternion and its kinematics unaffected by Euler angle convention
- Definition of $H_I^B$ makes the connection
- Specify Euler angle convention (e.g., 1-2-3 or 3-1-3); for (1-2-3),

$$H_I^B = \begin{bmatrix} h_{11} & h_{12} & h_{13} \\ h_{21} & h_{22} & h_{23} \\ h_{31} & h_{32} & h_{33} \end{bmatrix}$$

- Apply equations on earlier slide to find $q(0)$
- Perform trigonometric inversions as indicated to generate $[\phi(t_o), \theta(t_o), \psi(t_o)]$ from $q(t_o)$

Yo-Yo De-Spin
Mars Odyssey Launch Phases

- Booster Separation
- Stage 2 Separation
- Stage 2 Ignition
- Heat Shield Separation
- Stage 3 Spinup
- Yo-Yo De-Spin

Yo-Yo De-spin

- Satellite is initially spinning at $\omega_z$ rad/s
- Angular momentum and rotational energy of satellite plus expendable masses are conserved
- Masses are released, moment of inertia increases, and angular velocity of satellite decreases
- With proper cord length (independent of initial spin rate), satellite is de-spun to zero angular velocity
Angular momentum
\[ h_z = I_{zz} \omega_z + m R^2 \left( \omega_z + \phi \left( \omega_z + \phi \right) \right) \]

Rotational energy
\[ T = \frac{1}{2} I_{zz} \omega_z^2 + \frac{1}{2} m R^2 \left( \omega_z^2 + \phi^2 \left( \omega_z + \phi \right)^2 \right) \]

Simultaneous solution for final angular rate
\[ \omega_{final} = \omega_{initial} \left( \frac{c R^2 - l^2}{c R^2 + l^2} \right) = 0 \quad \text{if} \quad l = R \sqrt{c} \]

Spaceloft 7 Sounding Rocket De-Spin
https://www.youtube.com/watch?v=5ZqbjQ9ASc8

Continuously Variable Torque Controllers
Overview of Control

Single- or multi-axis interpretation

Single-Axis “Classical” Control of Non-Spinning Spacecraft

Pitching motion (about the y axis) is to be controlled

\[
\begin{bmatrix}
\dot{p}(t) \\
\dot{q}(t) \\
\dot{r}(t)
\end{bmatrix} =
\begin{bmatrix}
M_x(t)/I_{xx} \\
M_y(t)/I_{xy} \\
M_z(t)/I_{zz}
\end{bmatrix} -
\begin{bmatrix}
(I_{zz} - I_{yy})q(t)r(t)/I_{xx} \\
(I_{xx} - I_{zz})p(t)r(t)/I_{yy} \\
(I_{yy} - I_{xx})p(t)q(t)/I_{zz}
\end{bmatrix}
\]

- For motion about the y axis only, this reduces to:

\[\dot{q}(t) = M_y(t)/I_{yy}\]

- Pitch angle equation:

\[\dot{\theta}(t) = q(t)\]
Single-Axis Angular Rate Control of Non-Spinning Spacecraft

- Small angle and angular rate perturbations
- Linear actuator, e.g.,
  - Momentum wheel
- Linear measurement, e.g.,
  - Angular rate gyro

Simplified Control Law \((C = \text{Control Gain})\)

\[
e(t) = q_c(t) - q(t)
\]
\[
u(t) = C e(t)
\]

Angular Rate Control

\[
q(t) = \frac{g_A}{I_{yy}} \int_0^t u(t) dt = \frac{C g_A}{I_{yy}} \int_0^t e(t) dt = \frac{C g_A}{I_{yy}} \int_0^t [q_c - q(t)] dt
\]

- \(I_{yy}\): moment of inertia
- \(q(t)\): angular rate
- \(q_c(t)\): desired angular rate
- \(g_A\): actuator gain
- \(g_A u(t)\): control torque
Step Response of Angular Rate Controller

Step input:
\[ q_c(t) = \begin{cases} 
0, & t < 0 \\
1, & t \geq 0 
\end{cases} \]

\[ q(t) = q_c \left[ 1 - e^{-\frac{C_g A}{I_{yy}} t} \right] = q_c \left[ 1 - e^{\lambda t} \right] = q_c \left[ 1 - e^{-\frac{t}{\tau}} \right] \]

- where
  - \( \lambda = -\frac{C_g A}{I_{yy}} \) = eigenvalue or root of the system (rad/s)
  - \( \tau = \frac{I_{yy}}{C_g A} \) = time constant of the response (s)

Angle Control of the Spacecraft

- Small angle and angular rate perturbations
- Linear actuator, e.g.,
  - Momentum wheel
- Linear measurement, e.g.,
  - Earth horizon sensor

Angle Control Law (\( C = \text{Control Gain} \))

\[ e(t) = \theta_c(t) - \theta(t) \]
\[ u(t) = C e(t) \]
Model of Dynamics and Angle Control

- 2\textsuperscript{nd}-order ordinary differential equation

\[
d\frac{d^2\theta(t)}{dt^2} = \ddot{\theta}(t) = \frac{C g_A}{I_{yy}} \left[ \theta_c - \theta(t) \right]
\]

- Output angle, $\theta(t)$, as a function of time

\[
\theta(t) = \frac{g_A}{I_{yy}} \int_{0}^{t} \int_{0}^{t} u(t) dt \, dt = \frac{C g_A}{I_{yy}} \int_{0}^{t} \int_{0}^{t} e(t) dt \, dt = \frac{C g_A}{I_{yy}} \int_{0}^{t} \int_{0}^{t} \left[ \theta_c - \theta(t) \right] dt \, dt
\]

Rewrite 2\textsuperscript{nd}-Order Model as Two 1\textsuperscript{st}-Order Equations

\[
\dot{\theta}(t) = q(t)
\]

\[
\dot{q}(t) = \frac{C g_A}{I_{yy}} \left[ \theta_c - \theta(t) \right]
\]

\[
\begin{bmatrix}
\dot{\theta}(t) \\
\dot{q}(t)
\end{bmatrix} =
\begin{bmatrix}
0 & 1 \\
0 & 0
\end{bmatrix}
\begin{bmatrix}
\theta(t) \\
q(t)
\end{bmatrix} +
\begin{bmatrix}
0 \\
\frac{g_A}{I_{yy}}
\end{bmatrix}
C \left[ \theta_c(t) - \theta(t) \right]
\]

\[
\begin{bmatrix}
\dot{\theta}(t) \\
\dot{q}(t)
\end{bmatrix} =
\begin{bmatrix}
0 & 1 \\
-C \frac{g_A}{I_{yy}} & 0
\end{bmatrix}
\begin{bmatrix}
\theta(t) \\
q(t)
\end{bmatrix} +
\begin{bmatrix}
0 \\
C g_A / I_{yy}
\end{bmatrix}
\theta_c
\]
Simulation of **Step Response** with Angle Feedback

Objective is to control angle to 1 rad, but solution oscillates about the target

![Graph showing step response with angle feedback](image)

\[ \frac{Cg_A}{I_{yy}} = 1, 0.5, \text{ and } 0.25 \]

**What Went Wrong?**

- No damping!
- Solution: *Add* rate feedback
- Control law with rate feedback

\[
\begin{align*}
    u(t) &= c_1 \left[ \theta_c(t) - \theta(t) \right] - c_2 q(t)
\end{align*}
\]

**Closed-loop dynamic equation**

\[
\begin{bmatrix}
    \dot{\theta}(t) \\
    \dot{q}(t)
\end{bmatrix}
= \begin{bmatrix}
    0 & 1 \\
    -c_1 g_A / I_{yy} & -c_2 g_A / I_{yy}
\end{bmatrix}
\begin{bmatrix}
    \theta(t) \\
    q(t)
\end{bmatrix}
+ \begin{bmatrix}
    0 \\
    c_1 g_A / I_{yy}
\end{bmatrix} \theta_c
\]
Step Response with Angle and Rate Feedback

2\textsuperscript{nd}-Order Dynamics

Oscillation and damping are induced by linear feedback control

\[
\begin{bmatrix}
\dot{\theta}(t) \\
\dot{q}(t)
\end{bmatrix} = \begin{bmatrix}
0 & 1 \\
-c_1g_A/I_{yy} & -c_2g_A/I_{yy}
\end{bmatrix} \begin{bmatrix}
\theta(t) \\
q(t)
\end{bmatrix} + \begin{bmatrix}
0 \\
c_1g_A/I_{yy}
\end{bmatrix} \theta_c
\]

\[
\begin{bmatrix}
\dot{\theta}(t) \\
\dot{q}(t)
\end{bmatrix} = \begin{bmatrix}
0 & 1 \\
-\omega_n^2 & -2\zeta\omega_n
\end{bmatrix} \begin{bmatrix}
\theta(t) \\
q(t)
\end{bmatrix} + \begin{bmatrix}
0 \\
\omega_n^2
\end{bmatrix} \theta_c
\]

Natural frequency and damping ratio

\[
\omega_n = \sqrt{c_1g_A/I_{yy}}
\]

\[
\zeta = \left( \frac{c_2g_A}{I_{yy}} \right) / 2\omega_n = c_2 / 2\sqrt{c_1g_A/I_{yy}}
\]
Effect of Damping on Eigenvalues, Damping Ratio, and Natural Frequency

Eigenvalues

\[ \lambda_1, \lambda_2 = \]
\[ 0 + 1.0000i \]
\[ 0 - 1.0000i \]
\[ -0.7070 + 0.7072i \]
\[ -0.7070 - 0.7072i \]
\[ -0.4143 \]
\[ -2.4137 \]

Damping Ratio, Natural Frequency

\[ \zeta = \omega_n = ( \text{rad/s}) \]
\[ 0 \]
\[ 1 \]
\[ 0.707 \]
\[ 1 \]

Overdamped

Control System Design to Adjust Roots

Choose control gains to satisfy desirable eigenvalue range
Control System Design to Adjust Transient Response

Choose control gains to satisfy step response criteria

![Graph showing step response criteria with Maximum Overshoot, Rise Time, and Setting Time.

Control System Design to Adjust Frequency Response

Choose control gains to satisfy frequency response criteria

![Graph showing frequency response criteria with Amplitude Ratio and Phase Angle.

Input Frequency, rad/s
Laplace Transform of the State Vector

Neglecting the initial condition

\[ x(s) = \frac{Adj(sI - F)}{\Delta(s)} Gu(s) \]

Applied to the closed-loop system

\[
\begin{bmatrix}
\Delta\theta(s) \\
\Delta q(s)
\end{bmatrix} =
\begin{bmatrix}
c_1 g_{A}/I_{yy} \\
sc_1 g_{A}/I_{yy}
\end{bmatrix}
\begin{bmatrix}
\Delta(s) \\
\Delta(s)
\end{bmatrix}
\Delta u(s) =
\begin{bmatrix}
c_1 g_{A}/I_{yy} \\
sc_1 g_{A}/I_{yy}
\end{bmatrix}
\begin{bmatrix}
(s)^2 \Delta u(s)
\end{bmatrix}
\]

Frequency Response of the System

\[ \sigma = j\omega \]

Angle Frequency Response

\[
\frac{\Delta\theta(j\omega)}{\Delta u(j\omega)} = \frac{\omega_n^2}{(j\omega)^2 + 2\zeta\omega_n(j\omega) + \omega_n^2}
\]

Rate Frequency Response

\[
\frac{\Delta q(j\omega)}{\Delta u(j\omega)} = \frac{(j\omega)\omega_n^2}{(j\omega)^2 + 2\zeta\omega_n(j\omega) + \omega_n^2}
\]

- Bode plot
  - 20 log(Amplitude Ratio) [dB] vs. log \( \omega \)
  - Phase angle (deg) vs. log \( \omega \)
Proportional-Integral-Derivative (PID) Controller

PID Control Law (or compensator):

\[ e(t) = \theta_c(t) - \theta(t) \]
\[ u(t) = c_I \int e(t) \, dt + c_P e(t) + c_D \frac{de(t)}{dt} \]

Control Law Transfer Function:

\[ e(s) = \Theta_C(s) - \Theta(s) \]
\[ u(s) = c_P e(s) + c_I \frac{e(s)}{s} + c_D se(s) \]
\[ \frac{u(s)}{e(s)} = \frac{c_I + c_P s + c_D s^2}{s} \]

Differentiator produces rate term for damping
Integrator compensates for persistent (bias) disturbance
Proportional-Integral-Derivative (PID) Controller

Forward-Loop Angle Transfer Function:

\[
\frac{\theta(s)}{e(s)} = \left[ \frac{c_I + c_p s + c_D s^2}{s} \right] \left[ \frac{g_A}{I_{yy} s^2} \right]
\]

Closed-Loop Spacecraft Control Transfer Function w/PID Control

Closed-Loop Angle Transfer Function:

\[
\frac{\theta_c(s)}{\theta(s)} = \frac{\theta(s)}{e(s)} \frac{1 + \theta(s)}{e(s)} = \left[ \frac{c_I + c_p s + c_D s^2}{s} \right] \left[ \frac{g_A}{I_{yy} s^3} \right]
\]

\[
= \frac{c_I + c_p s + c_D s^2}{c_I + c_p s + c_D s^2 + g_A / I_{yy} s^3}
\]
Closed-Loop Frequency Response w/PID Control

\[
\frac{\Theta(s)}{\Theta_c(s)} = \frac{c_i + c_ps + c_ds^2}{c_i + c_ps + c_ds^2 + g_A / I_{yy}s^3}
\]

Let \( s = j\omega \). As \( \omega \to 0 \)

\[
\frac{\Theta(j\omega)}{\Theta_c(j\omega)} \to \frac{c_i}{c_i} = 1
\]

Steady-state output = desired steady-state input

As \( \omega \to \infty \)

\[
\frac{\Theta(j\omega)}{\Theta_c(j\omega)} \to \frac{-c_D\omega^2}{-jI_{yy}\omega^2 g_A} = \frac{c_D}{jI_{yy}\omega} g_A = \frac{-jc_D}{I_{yy}\omega} g_A
\]

AR \to \frac{c_D}{I_{yy}\omega} g_A; \quad \phi \to -90 \text{ deg}

High-frequency response "rolls off" and lags input

State ("Phase")-Plane Plots

Cross-plot of angle (or displacement) against rate
Time not shown explicitly in phase-plane plot
Effect of Damping Ratio on State-Plane Plots

Damping ratio = 0.1

Damping ratio = 0.3

Damping ratio = −0.1 (Unstable)

On/Off-Torque Controllers
Single-Axis State History with Constant Thrust

What if the control torque can only be turned ON or OFF?

\[
\begin{bmatrix}
\dot{\theta}(t) \\
\dot{q}(t)
\end{bmatrix} = \begin{bmatrix}
0 & 1 \\
0 & 0
\end{bmatrix} \begin{bmatrix}
\theta(t) \\
q(t)
\end{bmatrix} + \begin{bmatrix}
0 \\
g_A/I_{yy}
\end{bmatrix} u(t)
\]

\[ u(t) = +1, \ 0, \ \text{or} \ -1 \]

What is the time evolution of the state while a thruster is on \([u(t) = 1]\)?

\[
q(t) = \left(\frac{g_A}{I_{yy}}\right)t + q(0)
\]

\[
\dot{\theta}(t) = \left(\frac{g_A}{I_{yy}}\right)t^2 / 2 + q(0)t + \theta(0)
\]

Neglecting initial conditions, what does the phase-plane plot look like?

---

Constant-Thrust (Acceleration) Trajectories

*For* \(u = 1\),
*Acceleration* = \(g_A/I_{yy}\)

*For* \(u = -1\),
*Acceleration* = \(-g_A/I_{yy}\)

With zero thrust, what does the phase-plane plot look like?
Phase Plane Plot with Zero Thrust

How can you use this information to design an on-off control law?

Switching-Curve Control Law for On-Off Thrusters

- Origin (i.e., zero rate and attitude error) can be reached from any point in the state space
- Control logic:
  - Thrust in one direction until switching curve is reached
  - Then reverse thrust
  - Switch thrust off when errors are zero
Switching-Curve Control with Coasting Zone

Apollo Lunar Module Control

- 16 reaction control thrusters
  - Control about 3 axes
  - Redundancy of thrusters
- LM Digital Autopilot
Apollo Lunar Module Phase-Plane Control Logic

- Coast zones conserve RCS propellant by limiting angular rate
- With no coast zone, thrusters would chatter on and off at origin, wasting propellant
- State limit cycles about target attitude
- Switching curve shapes modified to provide robustness against modeling errors
  - RCS thrust level
  - Moment of inertia

Apollo Lunar Module Phase-Plane Control Law

Switching logic implemented in the Apollo Guidance & Control Computer
More efficient than a linear control law for on-off actuators
Typical Phase-Plane Trajectory

• With angle error, RCS turned on until reaching OFF switching curve
• Phase point drifts until reaching ON switching curve
• RCS turned off when rate is 0-
• Limit cycle maintained with minimum-impulse RCS firings
  – Amplitude = ±1 deg (coarse), ±0.1 deg (fine)

Multi-Axis Spacecraft Control

Asymmetry Introduces Dynamic Coupling, Complicating Control
Next Time:
Sensors and Actuators

Supplemental Material
GOES Attitude Control Sub-System