MDC ADCS Primer

We've got attitude

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ADCS Overview

- An Attitude Determination Control System is designed to stabilize and orient a spacecraft toward a given direction
- This is often a critical system for mission success, for example:
  - **Survival of the spacecraft.** If the satellite needs to point its solar panels toward the Sun to charge the batteries.
  - **Completion of the mission.** If taking images of the Earth the payload must be pointed towards a desired target.

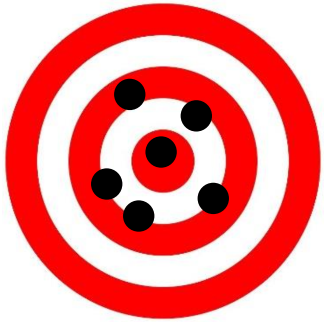
Terminology

- **Attitude**: Orientation of a defined spacecraft body coordinate system with respect to a defined external frame
- **Attitude Determination**: Real-time or post-facto knowledge, within a given tolerance, of the spacecraft attitude
- **Attitude Control**: Maintenance of desired, specified attitude within a given tolerance
- **Attitude Error**: ‘Low Frequency’ spacecraft misalignment; usually the intended topic of attitude control
- **Attitude Jitter**: ‘High Frequency’ spacecraft misalignment; usually ignored by ADCS; reduced by good design or fine pointing/optical control

Terminology

Low accuracy
Low precision

Low accuracy
High precision

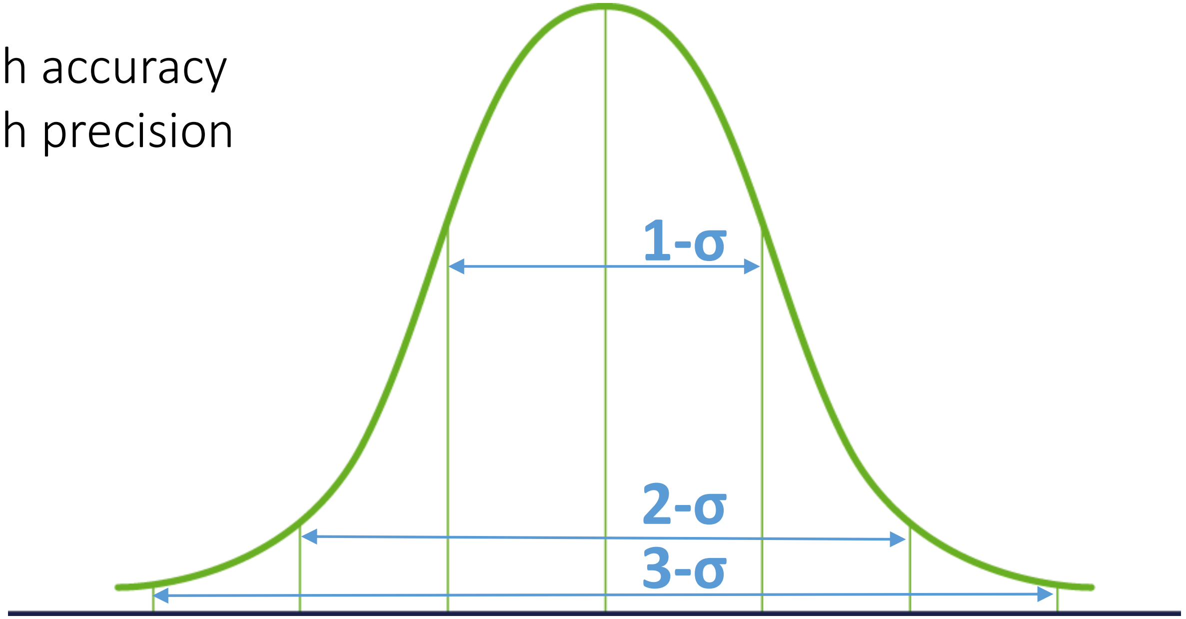


High accuracy
Low precision

High accuracy
High precision

- "Accuracy" is distance from truth
- "Precision" is repeatability
- **Be prepared for inconsistency between datasheets and manufacturers**

- Accuracy described with maximum bias
- Precision described with standard deviation
  - 1-σ bounds 68.3% (1 in 3.2)
  - 2-σ bounds 95.4% (1 in 22.0)
  - **3-σ bounds 99.7% (1 in 370.4)**
  - 6-σ bounds 99.99% (1 in 505 mil.)



Ask!

- Do I need an active ADCS system?
  - Some spacecraft systems do not have ADCS requirements
- Do I need to know where I am pointed?
- Why do I need to point there?
- What component on the spacecraft am I pointing there?
- How close is ‘close enough’?

What are the requirements?

ADCS Requirements

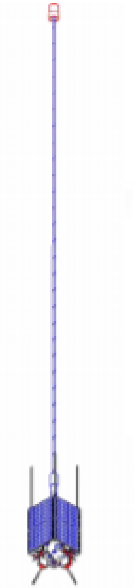
Criteria	Inputs	Outputs
Accuracy	Knowledge of and control over a vehicle's attitude relative to a target attitude as defined relative to an absolute reference	0.25 deg, 3-σ, often includes determination errors along with control errors, or there may be a separate requirement for determination & control, and even for different axes
	Determination/Knowledge Accuracy	+/-3.4 deg, 3-σ
	Pointing/Control Accuracy	+/-6.3 deg, 3-σ
Range	Range of angular motion over which determination and control performance must be met	Any attitude within 30˚ of nadir Whenever rotational rates are less than 2˚ deg/sec
Jitter	Specific bound on high-frequency angular motion	0.1˚over 60 sec, 1 deg/s, 1 to 20 Hz; prevents excessive blurring of sensor data
Drift	Limit on slow, low-frequency angular motion	0.01˚ over 20 min, 0.05 deg max; used when vehicle may drift off target with infrequent command inputs
Transient Response	Allowed settling time or max attitude overshoot when acquiring new targets or recover from upsets	10% max overshoot, decaying to <0.1 deg in 1 min; may also limit excursions from a set path between targets

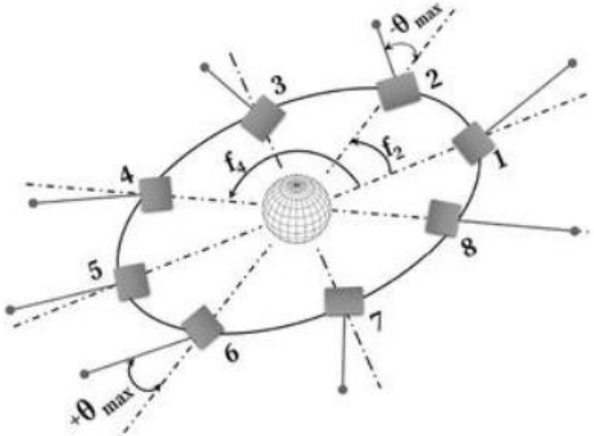
Steps In Attitude Design

Step	Inputs	Outputs
1a) Define control modes 1b) Define or derive system-level requirements by control mode	Mission requirements, mission profile, type of insertion for launch vehicle	List of different control modes during mission Requirements and constraints
2) Quantify disturbance environment	Spacecraft geometry, orbit, solar/mag models, mission profile	Values for torques from external and internal sources
3) Select type of spacecraft control by attitude control mode	Payload, thermal & power needs, orbit, pointing direction, disturbance environment, accuracy requirements	Method for stabilization and control: three-axis, spin, gravity gradient, etc.
4) Select and size ADCS hardware	Spacecraft geometry and mass properties, required accuracy, orbit geometry, mission lifetime, space environment, pointing direction, slew rates	**Sensor suite:** Earth, Sun, inertial, or other sensing devices. **Control actuators:** reaction wheels, thruster, magnetic torquers, etc. **Data processing avionics**, if any or processing requirements for other subsystems or ground computer
5) Define determination and control algorithms	Performance considerations (stabilization method(s), attitude knowledge & control accuracy, slew rates) balanced against system-level limitations (power and thermal needs, lifetime, jitter sensitivity, spacecraft processor capability)	Algorithms and parameters for each determination and control mode Logic for changing from one mode to another
6) Iterate and document	All of the above	Refined mission and subsystem requirements More detailed ADCS design Subsystem and component specifications

Attitude Control Types

Method	Typical Accuracy	Remarks
Spin Stabilized	0.1 deg	Passive, simple; single axis inertial, low cost, need slip rings – **good for deep space applications**
Gravity Gradient	1-10 deg	Passive, simple; central body oriented; low cost – **good for earth observing**
Thrusters	0.1 deg	Consumables required, fast; high cost
Magnetic	1 deg	Near Earth; slow; low weight, low cost – **good for LEO coarse pointing**
Reaction Wheels	0.01 deg	Internal torque; requires other momentum control; high power, cost – **good for 1 to 3-axis precision attitude control**



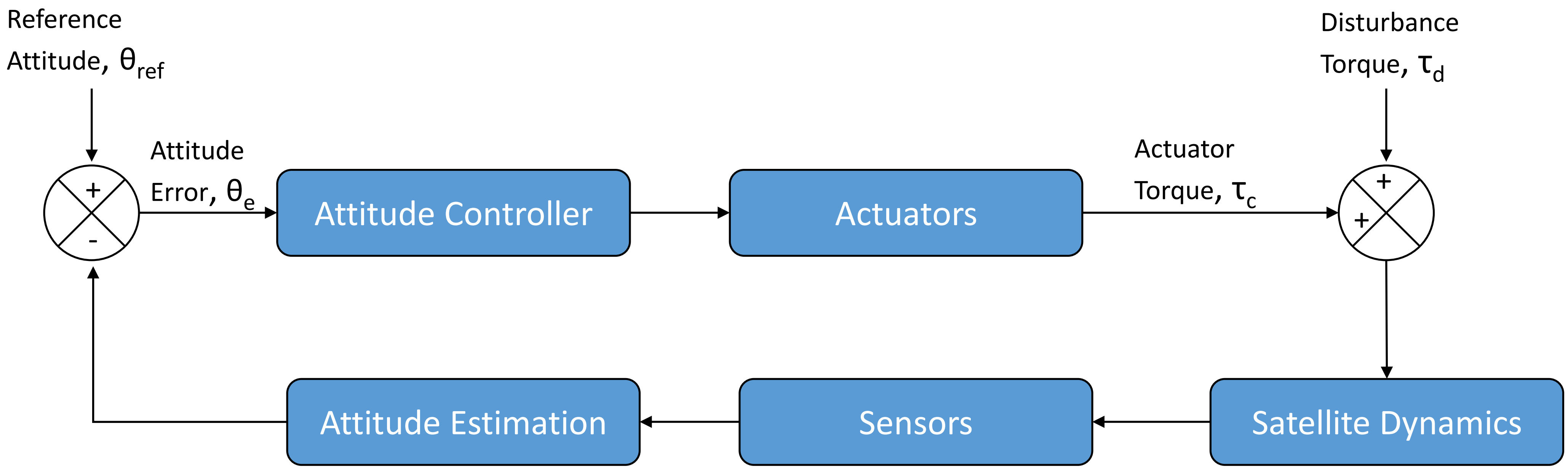


Gravity Gradient Stabilization

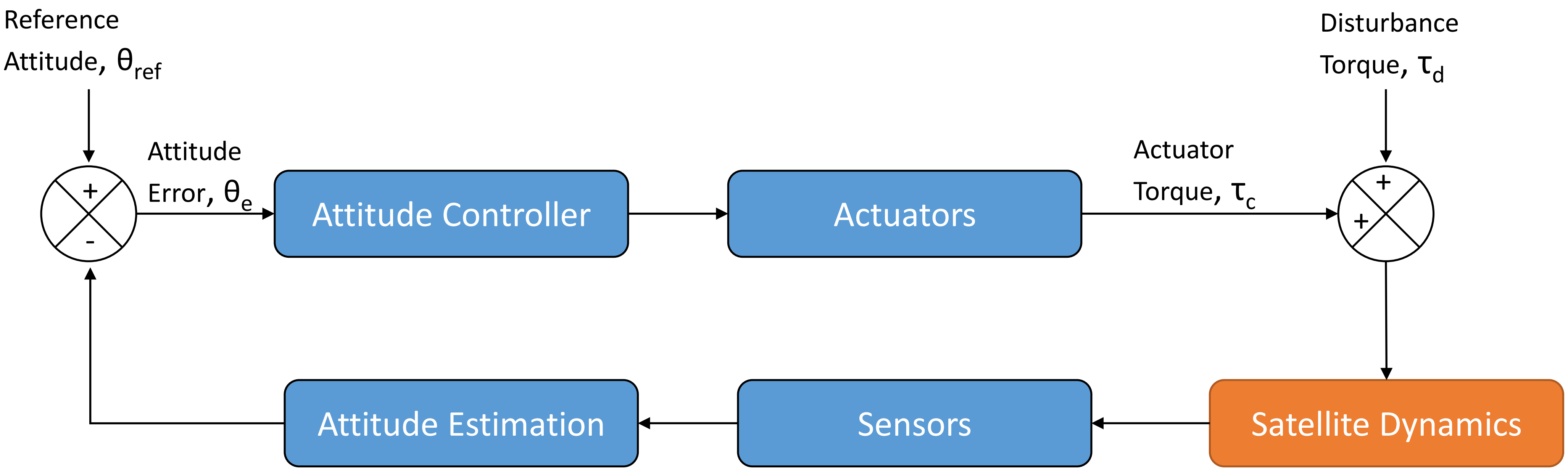
Spin-stabilized

Active Attitude Control

- Active Control Systems directly sense spacecraft attitude and supply a torque command to alter it as required. This is the basic concept of feedback control



Satellite Dynamics

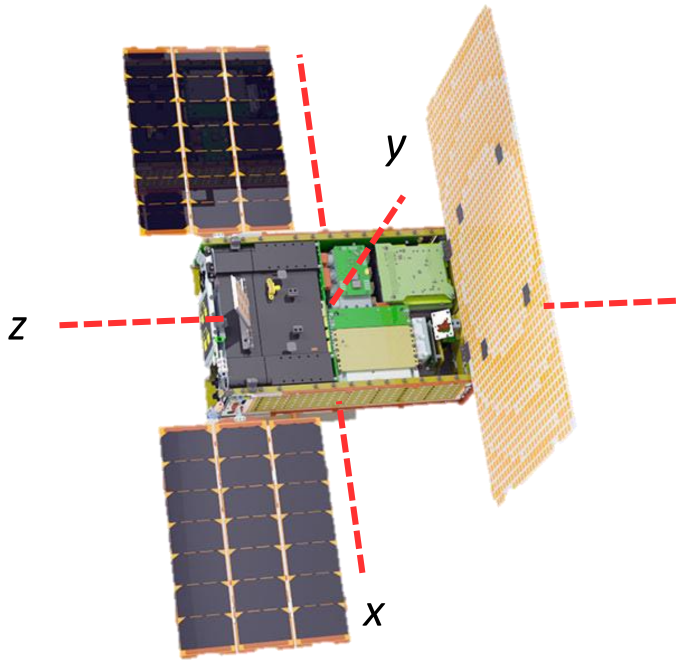


WARNING: MATH ON NEXT SLIDE

Satellite Dynamics - Theory

Degrees of Freedom (DOF): *the number of independent parameters that define its configuration or state*

Total: 6 DOF
Translational: x, y, z
Rotational: roll, pitch, yaw



5.7.1 Euler's 1st Law:

The resultant force applied to a rigid body $\mathcal{R}$ is equal to the product of the mass of the rigid body and the acceleration of the center of mass of the rigid body in an inertial reference frame $\mathcal{N}$, that is,

$$\boxed{\mathbf{F} = m\,{}^{\mathcal{N}}\bar{\mathbf{a}}} \qquad (5\text{-}225)$$

5.7.2 Euler's 2nd Law:

The resultant moment applied to a rigid body $\mathcal{R}$ relative to a point O fixed point in an inertial reference frame $\mathcal{N}$ is equal to the rate of change of angular momentum of the rigid body relative to point O in reference frame $\mathcal{N}$, that is,

$$\boxed{\mathbf{M}_O = \frac{{}^{\mathcal{N}}d}{dt}\left({}^{\mathcal{N}}\mathbf{H}_O\right)} \qquad (5\text{-}226)$$

The sum of the resultant moment applied to a rigid body relative to an arbitrary reference point Q and the inertial moment due to the acceleration of the reference point Q relative to the center of mass of a rigid body is equal to the rate of change of angular momentum of the rigid body relative to the reference point Q, that is,

$$\boxed{\mathbf{M}_Q - (\bar{\mathbf{r}} - \mathbf{r}_Q) \times m\,{}^{\mathcal{N}}\mathbf{a}_Q = \frac{{}^{\mathcal{N}}d}{dt}\left({}^{\mathcal{N}}\mathbf{H}_Q\right)} \qquad (5\text{-}235)$$

$$\mathbf{M}_Q - (\bar{\mathbf{r}} - \mathbf{r}_Q) \times m\,{}^{\mathcal{N}}\mathbf{a}_Q = \frac{{}^{\mathcal{R}}d}{dt}\left(\mathbf{I}_Q^{\mathcal{R}}\right) \cdot {}^{\mathcal{N}}\boldsymbol{\omega}^{\mathcal{R}} + \mathbf{I}_Q^{\mathcal{R}} \cdot {}^{\mathcal{N}}\boldsymbol{\alpha}^{\mathcal{R}} + {}^{\mathcal{N}}\boldsymbol{\omega}^{\mathcal{R}} \times \left(\mathbf{I}_Q^{\mathcal{R}} \cdot {}^{\mathcal{N}}\boldsymbol{\omega}^{\mathcal{R}}\right)$$

Governing equation of motion

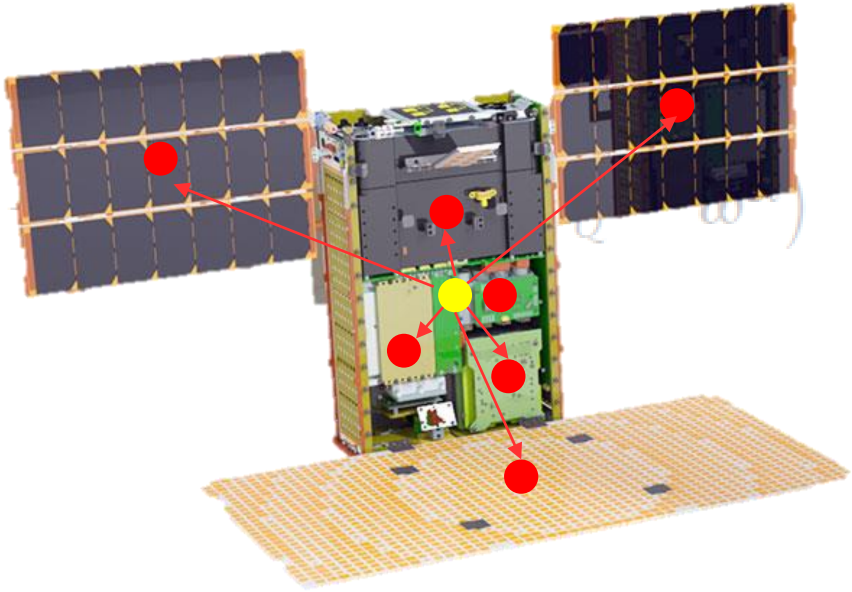
$$\mathbf{M}_Q - (\bar{\mathbf{r}} - \mathbf{r}_Q) \times m\, {}^{\mathcal{N}}\mathbf{a}_Q = \frac{{}^{\mathcal{R}}d}{dt}\left(\mathbf{I}_Q^{\mathcal{R}}\right) \cdot {}^{\mathcal{N}}\boldsymbol{\omega}^{\mathcal{R}} + \mathbf{I}_Q^{\mathcal{R}} \cdot {}^{\mathcal{N}}\boldsymbol{\alpha}^{\mathcal{R}} + {}^{\mathcal{N}}\boldsymbol{\omega}^{\mathcal{R}} \times \left(\mathbf{I}_Q^{\mathcal{R}} \cdot {}^{\mathcal{N}}\boldsymbol{\omega}^{\mathcal{R}}\right)$$

Euler's Second Law (1/3)

$$\mathbf{M}_Q - (\bar{\mathbf{r}} - \mathbf{r}_Q) \times m\, {}^{\mathcal{N}}\mathbf{a}_Q = \frac{{}^{\mathcal{R}}d}{dt}\left(\mathbf{I}_Q^{\mathcal{R}}\right) \cdot {}^{\mathcal{N}}\omega^{\mathcal{R}} \ldots$$

$$\mathbf{M}_Q = \sum_{i=1}^{n} (\mathbf{r}_i - \mathbf{r}_Q) \times \mathbf{F}_i + \boldsymbol{\tau}$$

- Red dot: Vector to each item mass ($\mathbf{r}_i$)
- Yellow dot: Vector to ref. point ($\mathbf{r}_Q$)
  - *Center of Geometry*
- Forces acting on the system ($\mathbf{F}_i$)
  - *Prop.*
  - *Drag*
- Torque acting on the system ($\boldsymbol{\tau}$)
  - *ADCS Actuators*
  - *Drag*



Euler's Second Law (2/3)

Center of Geometry → Vector to ref. point (yellow)

Mass of system

Orbit Dynamics → Acceleration of system relative to inertial ref. frame

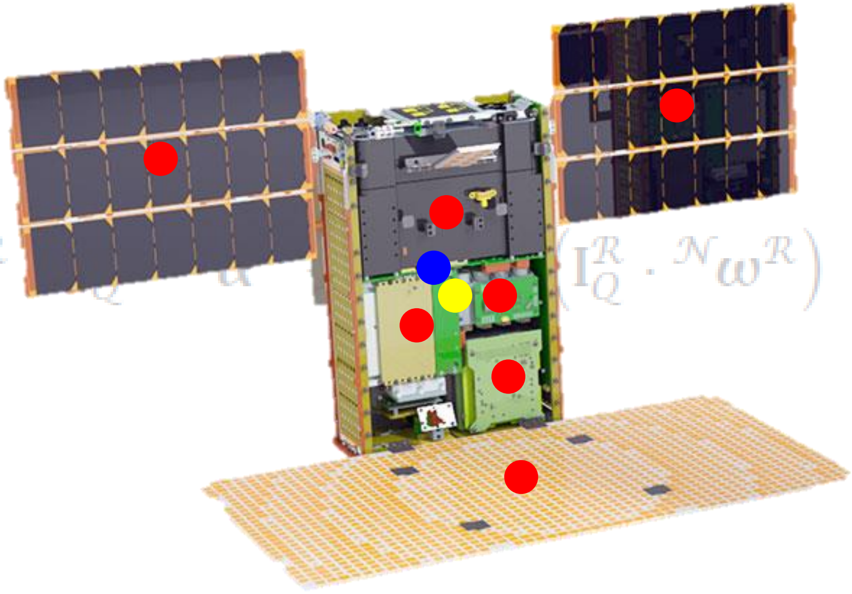
$$M_Q - (\bar{\mathbf{r}} - \mathbf{r}_Q) \times m\,{}^{\mathcal{N}}\mathbf{a}_Q = \frac{{}^{\mathcal{R}}d}{dt}\left(\mathbf{I}_Q^{\mathcal{R}}\right) \cdot {}^{\mathcal{N}}\omega^{\mathcal{R}} \cdots \left(\mathbf{I}_Q^{\mathcal{R}} \cdot {}^{\mathcal{N}}\omega^{\mathcal{R}}\right)$$

Vector to CoM of system (blue)

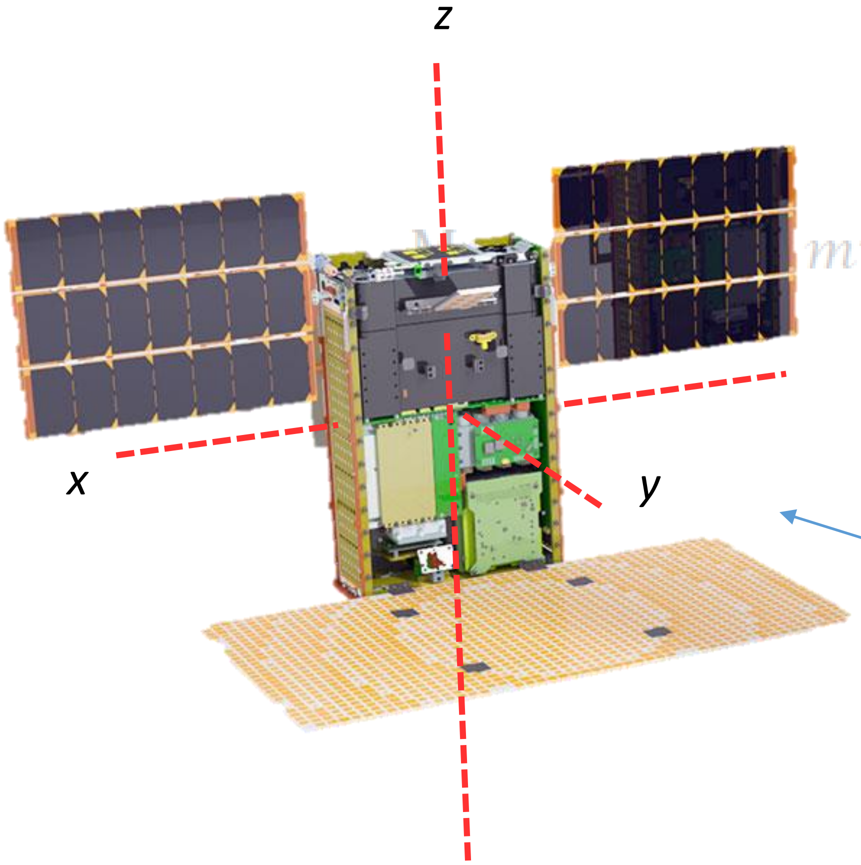
$$\bar{\mathbf{r}} = \frac{\sum_{i=1}^{n} m_i \mathbf{r}_i}{\sum_{i=1}^{n} m_i}$$

Item of mass (red)

Vector to mass item



Euler's Second Law (3/3)



System spin rate → Angular velocity of the system relative to the inertial ref. frame

Angular accel. of the system relative to the inertial ref. frame

$$m\,{}^{\mathcal{N}}\mathbf{a}_Q = \frac{{}^{\mathcal{R}}d}{dt}\left(\mathbf{I}_Q^{\mathcal{R}}\right)\cdot {}^{\mathcal{N}}\boldsymbol{\omega}^{\mathcal{R}} + \mathbf{I}_Q^{\mathcal{R}}\cdot {}^{\mathcal{N}}\boldsymbol{\alpha}^{\mathcal{R}} + {}^{\mathcal{N}}\boldsymbol{\omega}^{\mathcal{R}} \times \left(\mathbf{I}_Q^{\mathcal{R}}\cdot {}^{\mathcal{N}}\boldsymbol{\omega}^{\mathcal{R}}\right)$$

Mom. of Inertia of system relative to body ref. frame

$$\begin{array}{c} \\ x \\ y \\ z \end{array}\begin{array}{c} \begin{array}{ccc} x & y & z \end{array} \\ \left\{\begin{array}{ccc} I_{11} & I_{12} & I_{13} \\ I_{12} & I_{22} & I_{23} \\ I_{13} & I_{23} & I_{33} \end{array}\right\} \end{array}$$

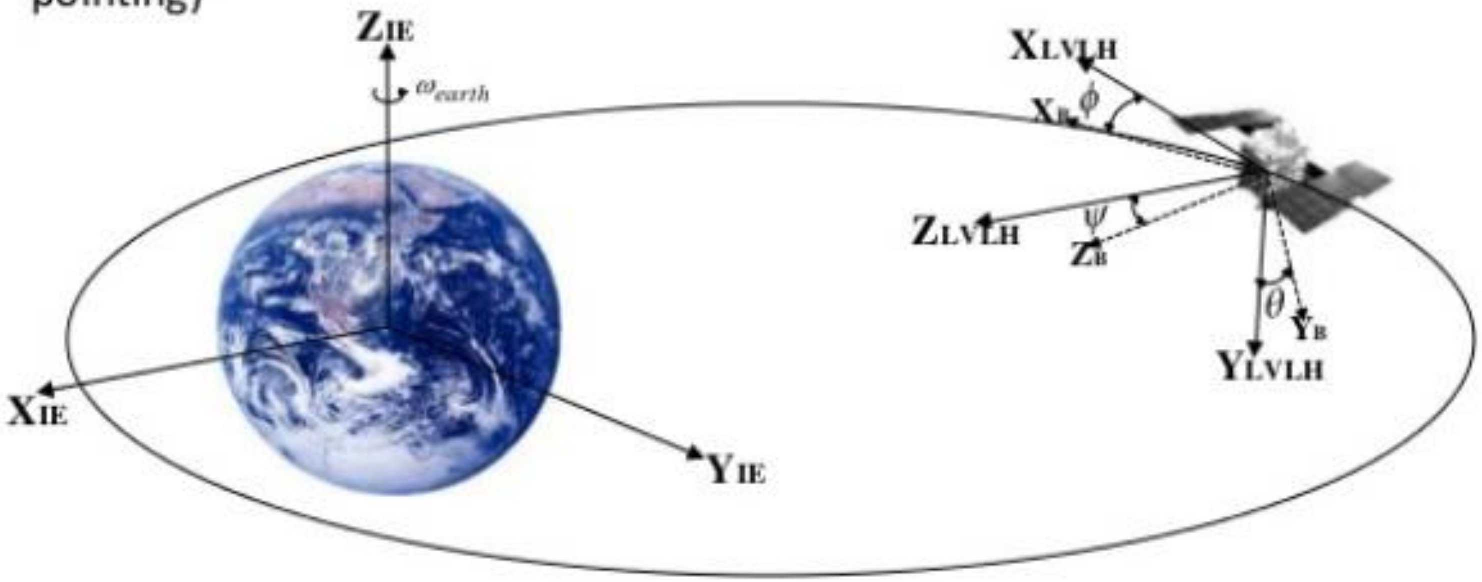
Reference Frames

$$\mathbf{M}_Q - (\bar{\mathbf{r}} - \mathbf{r}_Q) \times m\,{}^{\mathcal{N}}\mathbf{a}_Q = \frac{{}^{\mathcal{R}}d}{dt}\left(\mathrm{I}^{\mathcal{R}}_{Q}\right) \cdot {}^{\mathcal{N}}\boldsymbol{\omega}^{\mathcal{R}} + \mathrm{I}^{\mathcal{R}}_{Q} \cdot {}^{\mathcal{N}}\boldsymbol{\alpha}^{\mathcal{R}} + {}^{\mathcal{N}}\boldsymbol{\omega}^{\mathcal{R}} \times \left(\mathrm{I}^{\mathcal{R}}_{Q} \cdot {}^{\mathcal{N}}\boldsymbol{\omega}^{\mathcal{R}}\right)$$

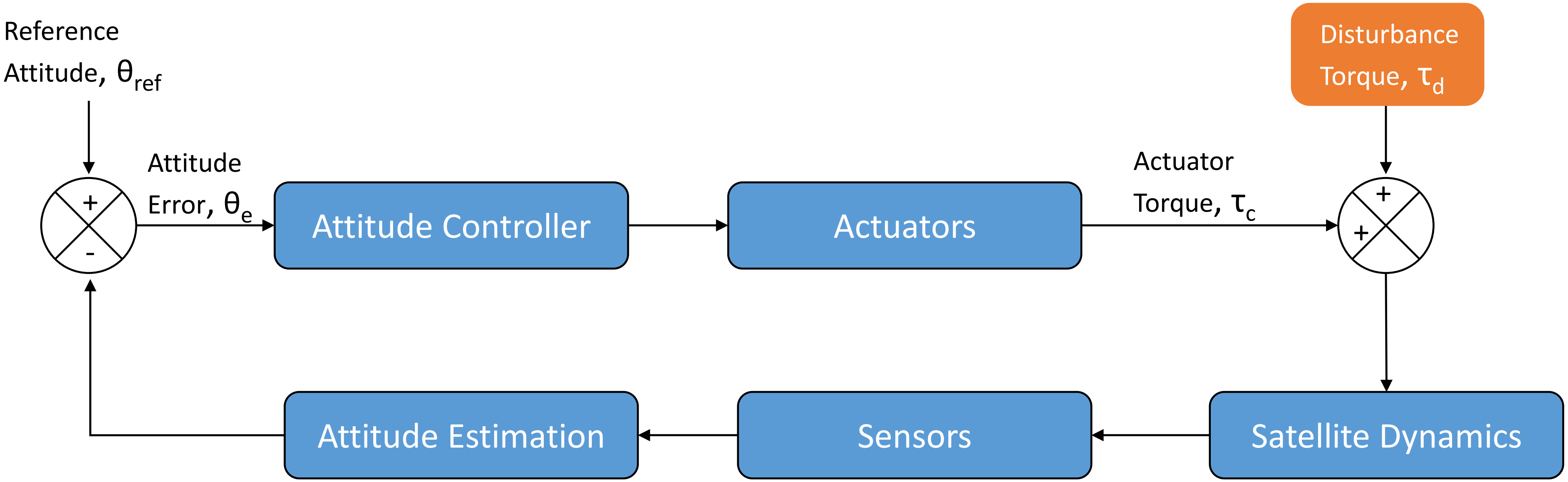
"Relative to inertial ref. frame"

Require to describe the motion of a satellite

- Inertial Earth (IE) coordinate system,
- Satellite's Body (B) coordinate system
- Local-Vertical-Local-Horizontal (LVLH) coordinate system (assigned for nadir pointing)

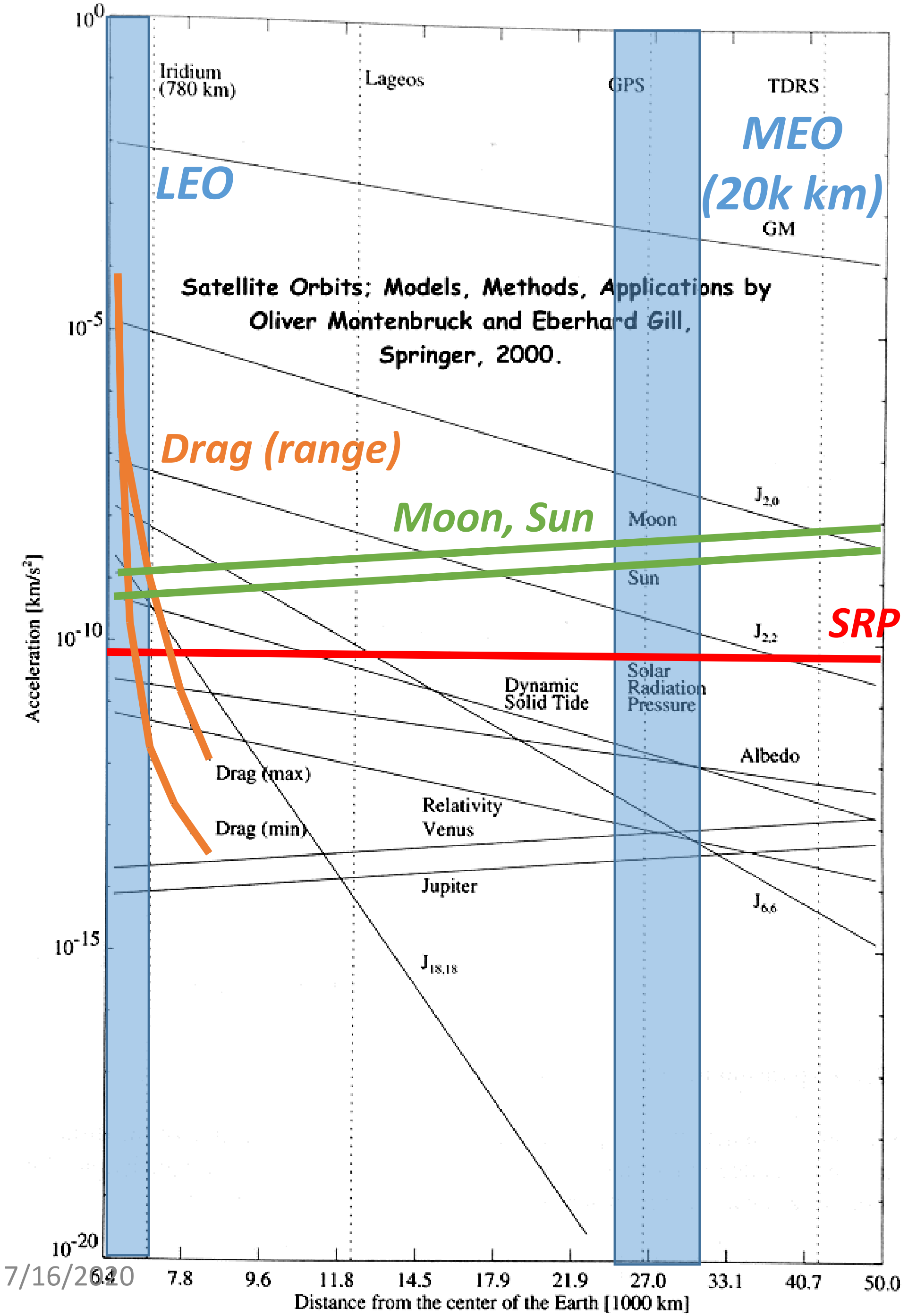


Disturbance Torques

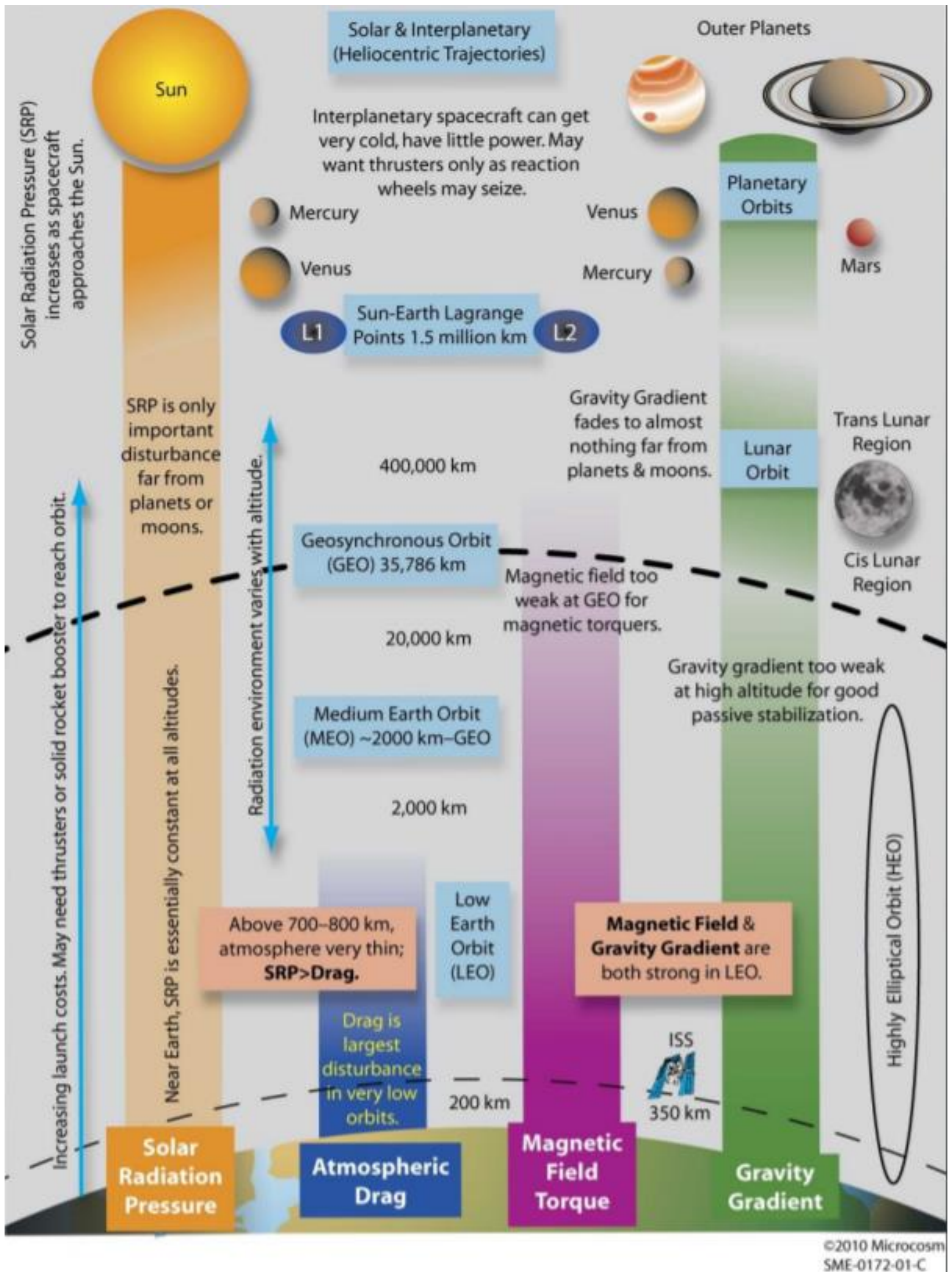


Typical Disturbances

- **Gravity Gradient**: "Tidal" Force due to $1/r^2$ gravitational field variation for long, extended bodies (e.g. Space Shuttle, Tethered vehicles)
- **Aerodynamic Drag**: "Weathervane" Effect due to an offset between the CM and the drag center of Pressure (CP). Only a factor in LEO.
- **Magnetic Torques**: Induced by residual magnetic moment. Model the spacecraft as a magnetic dipole. Only within magnetosphere.
- **Solar Radiation**: Torques induced by CM and solar CP offset. Can compensate with differential reflectivity or reaction wheels.
- **Mass Expulsion**: Torques induced by leaks or jettisoned objects
- **Internal**: On-board Equipment (machinery, wheels, cryocoolers, pumps etc…). No net effect, but internal momentum exchange affects attitude.



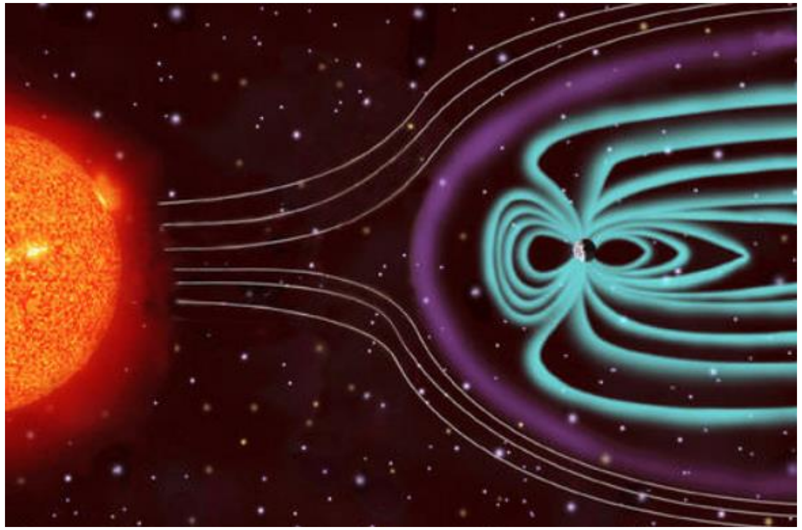
Iridium (780 km)
Lageos
GPS
TDRS
LEO
MEO (20k km)
GM
Satellite Orbits; Models, Methods, Applications by Oliver Montenbruck and Eberhard Gill, Springer, 2000.
Drag (range)
Moon, Sun
Moon
Sun
SRP
Dynamic Solid Tide
Solar Radiation Pressure
Albedo
Drag (max)
Drag (min)
Relativity
Venus
Jupiter
Acceleration [km/s²]
Distance from the center of the Earth [1000 km]
6.4
7.8
9.6
11.8
14.5
17.9
21.9
27.0
33.1
40.7
50.0



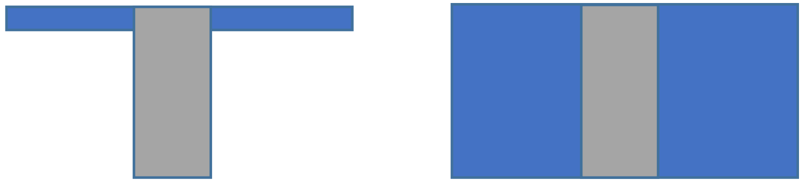
Solar & Interplanetary (Heliocentric Trajectories)
Outer Planets
Sun
Interplanetary spacecraft can get very cold, have little power. May want thrusters only as reaction wheels may seize.
Mercury
Venus
Mars
Planetary Orbits
Sun-Earth Lagrange Points 1.5 million km
L1
L2
Solar Radiation Pressure (SRP) increases as spacecraft approaches the Sun.
SRP is only important disturbance far from planets or moons.
Gravity Gradient fades to almost nothing far from planets & moons.
Lunar Orbit
Trans Lunar Region
400,000 km
Radiation environment varies with altitude.
Geosynchronous Orbit (GEO) 35,786 km
Cis Lunar Region
Magnetic field too weak at GEO for magnetic torquers.
20,000 km
Gravity gradient too weak at high altitude for good passive stabilization.
Medium Earth Orbit (MEO) ~2000 km–GEO
Increasing launch costs. May need thrusters or solid rocket booster to reach orbit.
Near Earth, SRP is essentially constant at all altitudes.
2,000 km
Above 700–800 km, atmosphere very thin; SRP>Drag.
Low Earth Orbit (LEO)
Magnetic Field & Gravity Gradient are both strong in LEO.
Highly Elliptical Orbit (HEO)
Drag is largest disturbance in very low orbits.
ISS
200 km
350 km
Solar Radiation Pressure
Atmospheric Drag
Magnetic Field Torque
Gravity Gradient
©2010 Microcosm
SME-0172-01-C

Typical Disturbance Torques

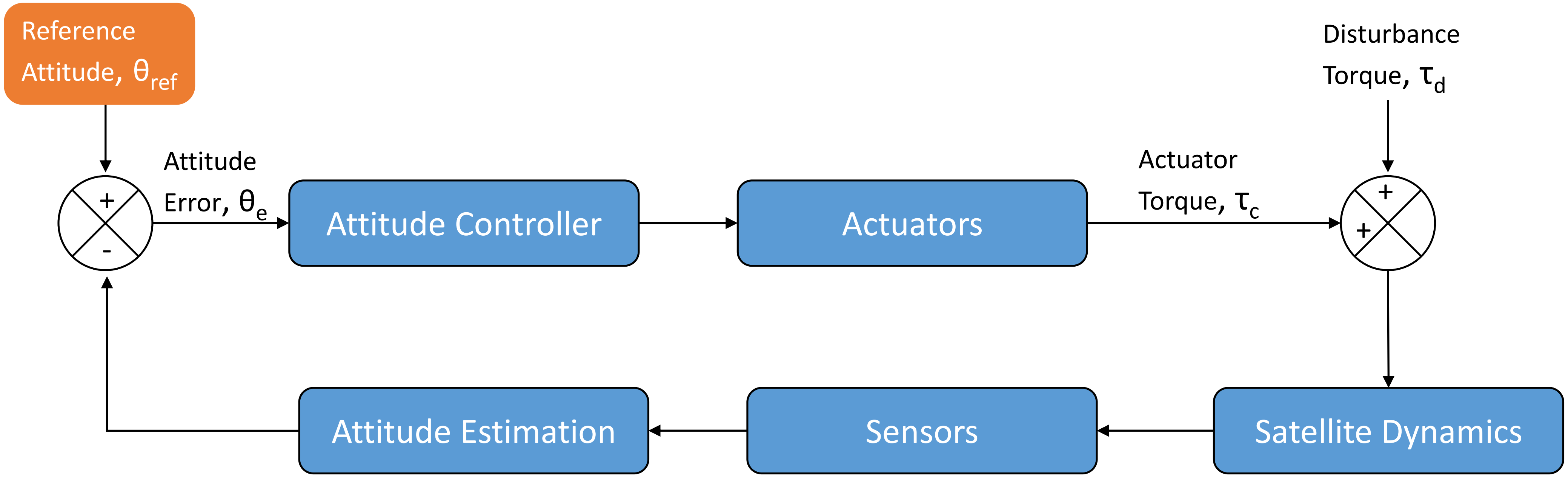
- Generally, gravity gradient and third-body are largest disturbances
  - In LEO, third largest is aerodynamic drag
  - In MEO+, third largest is solar radiation pressure
- Aerodynamic Drag and Solar Radiation Pressure
  - Both depend on mass and geometric distribution
    - Center of Pressure vs Center of Mass
    - Drag Area/Flight configuration



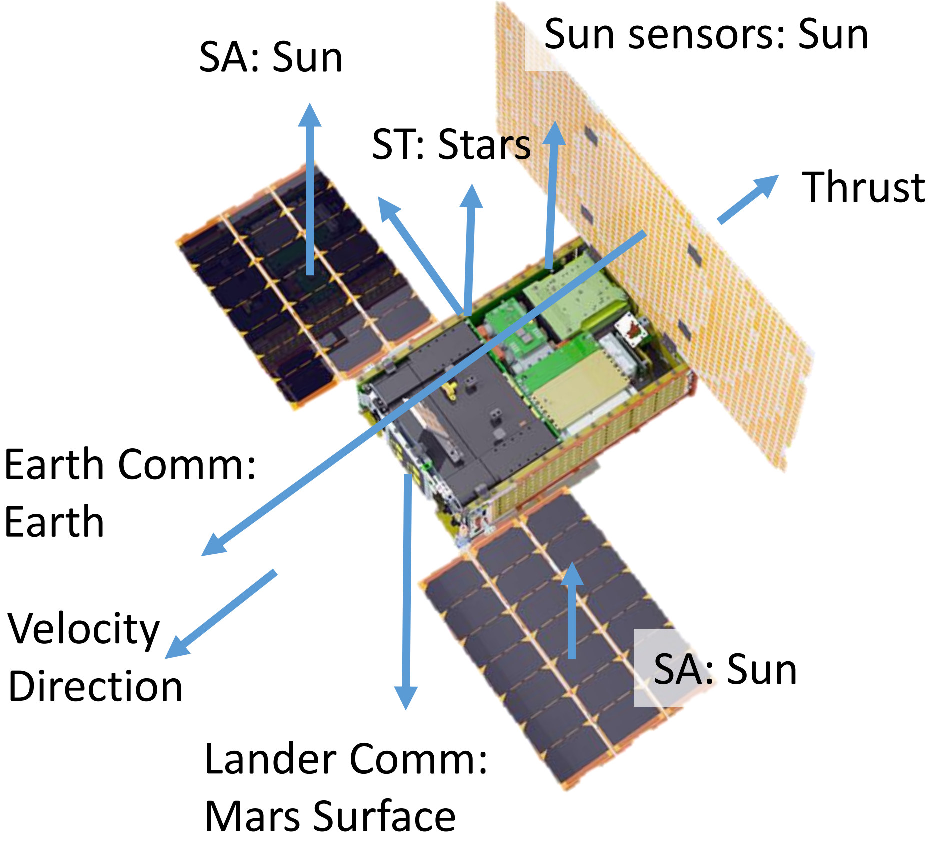
Low/High Drag based on area in the velocity direction

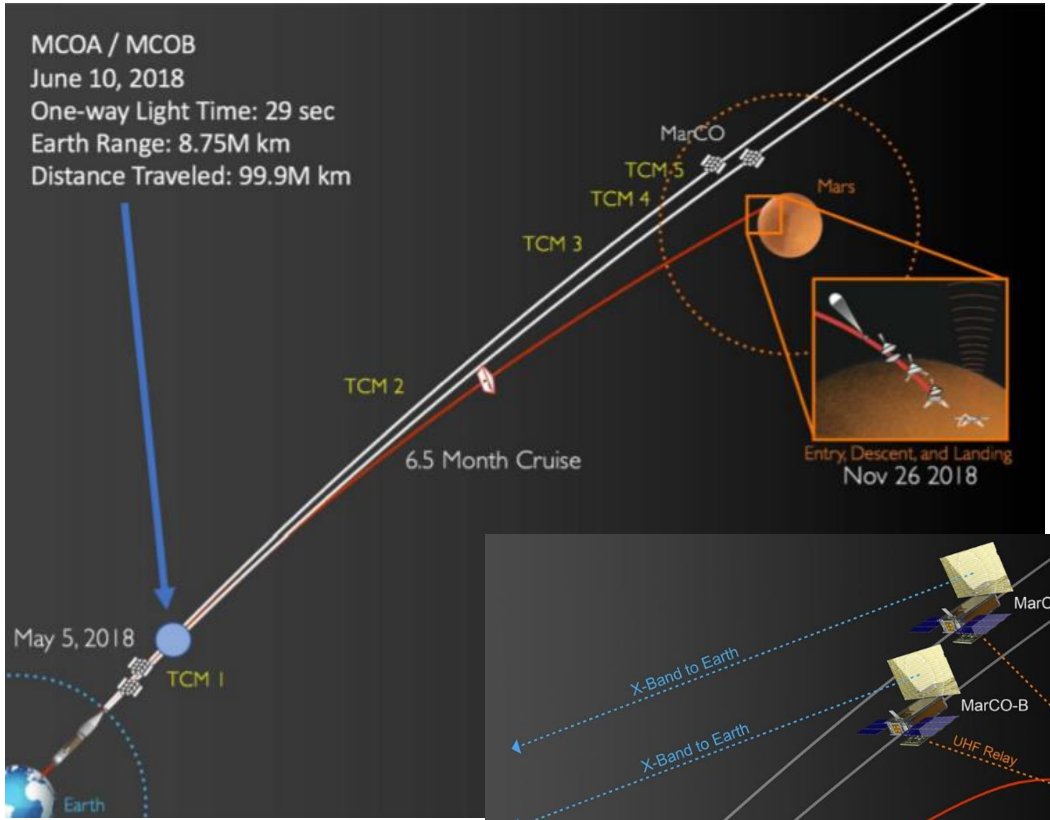


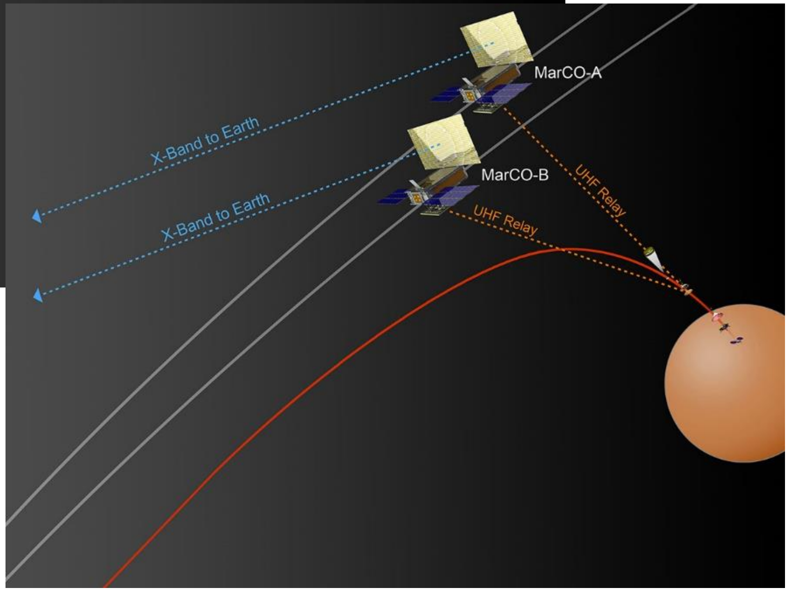
Reference Attitude



S/C Configuration, ConOps



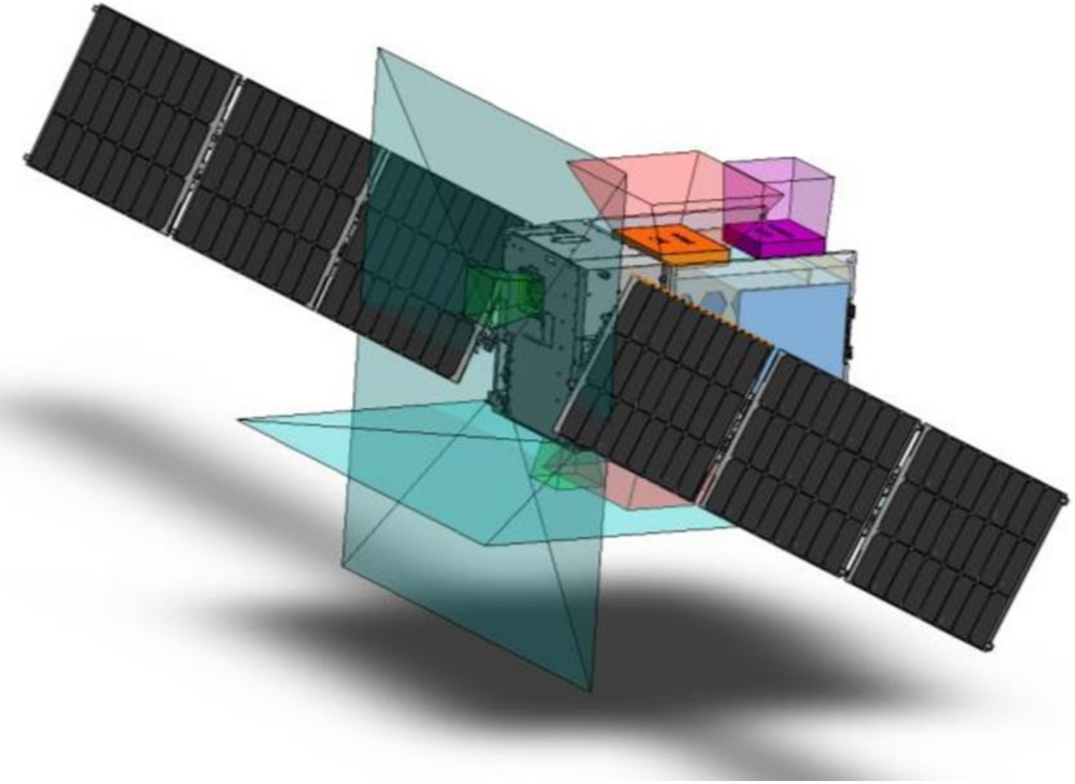




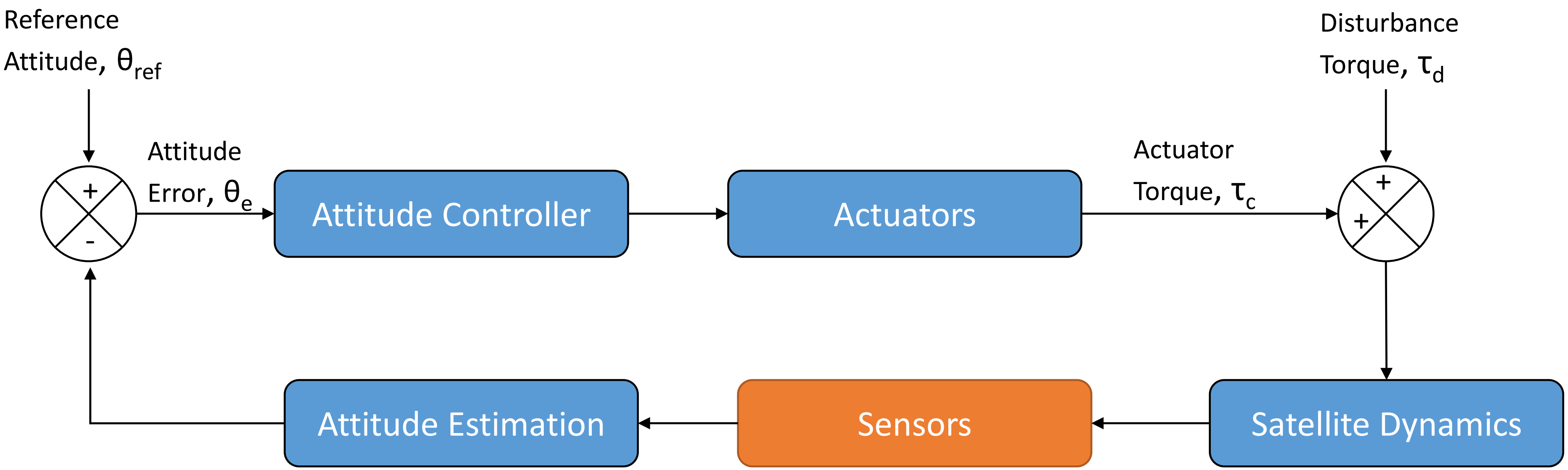
- Coordinate system definition and consistency important
- ADCS can be used to mitigate Thermal, Power, Comm risks

Field of Views (FOV)

- ConOps impacted by FOV of sensors and other equipment
- Typical Sensors FOV
  - Sun sensors 170-180°
  - Star Trackers 10-20°
    - With moon and sun keepout 30-60°
- Communications narrow at higher frequencies, highly variable on antenna design and link budget
  - UHF typically omnidirectional
  - S-band 20-60°
  - K-band 1-10°
  - Optical < 1°
- Payloads are mission dependent
  - For earth-pointing, FOV and orbital altitude determines swath width



Sensors



Reference
Attitude, θref
Disturbance
Torque, τd
Attitude
Error, θe
+
-
Attitude Controller
Actuators
Actuator
Torque, τc
+
+
Attitude Estimation
Sensors
Satellite Dynamics

Traditional Sensor Types

Criteria	Inputs	Mass (kg)	Power (W)
Gyroscopes	Drift Rate = 0.003 deg/hr to 1 deg/hr Drift rate stability varies wildly	<0.1 to 15	<1 to 200
Sun Sensors	Accuracy = 0.005 deg to 3 deg	0.1 to 2	0 to 3
Star Trackers	Accuracy = 1 arcsecond to 1 arcminute = 0.0003 deg to 0.01 deg	2 to 5	5 to 20
Horizon Sensors	Scanner Accuracy: 0.05 deg to 1 deg (0.1 deg best for LEO) Fixed Head Accuracy: <0.1 deg to 0.25 deg	1 to 4 0.5 to 3.5	5 to 10 0.3 to 5
Magnetometer	Accuracy = 0.5 deg to 3 deg	0.3 to 1.2	<1

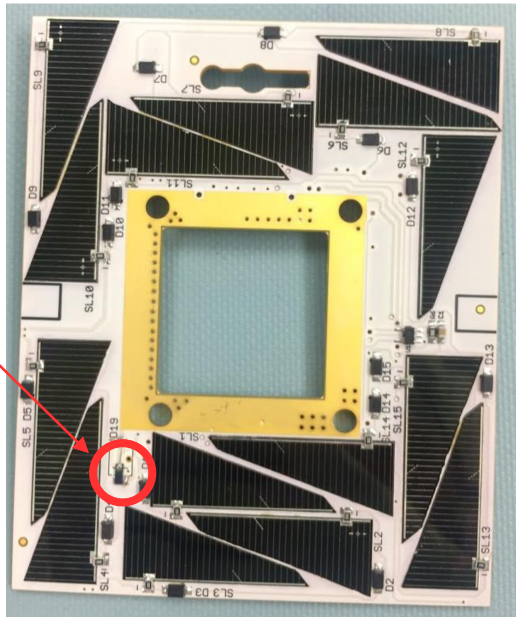
Not typical of CubeSat class

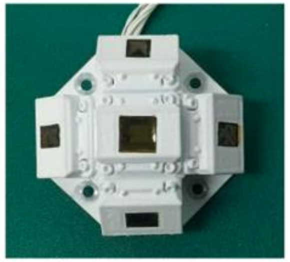
CubeSat Sensor Overview

- **Sun sensors** ($-$$): measures solar intensity hitting detector
- **Magnetometer** ($): measures magnetic field strength and direction with respect to body axis
- Inertial Measurement Unit (**IMU**) ($)
  - **Gyro**: measures angular velocity with respect to the body axis
  - **Accelerometer**: measures acceleration with respect to body axis
- **Star tracker** ($$$$): determines attitude through comparing visible stars to star catalogue
- **Horizon sensor** ($$$): orients through viewing edge of earth's atmosphere

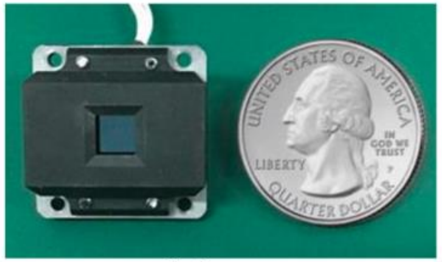
Sun Sensors

- **Sun sensors**: measures solar intensity hitting detector
- Typical accuracy 5°, down to 1° for high performance
  - Sun angular size in LEO is 0.5° accuracy (3-σ), 0.1° precision
- Variants: array (pyramidal/strip), slot, diode
- Placement and operational concerns
  - Must know spacecraft and the sun position for use as attitude reference (typically)
  - Antennae, solar arrays, and other deployables can interfere
  - Solar panel currents can be used as a ‘coarse sun sensor’
  - Inoperable in eclipse, must handle case in FSW to avoid attitude determination solution issue





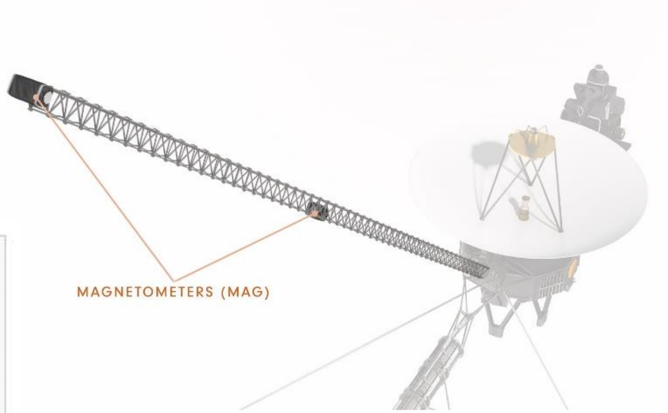
Sun sensor array



Single sun sensor

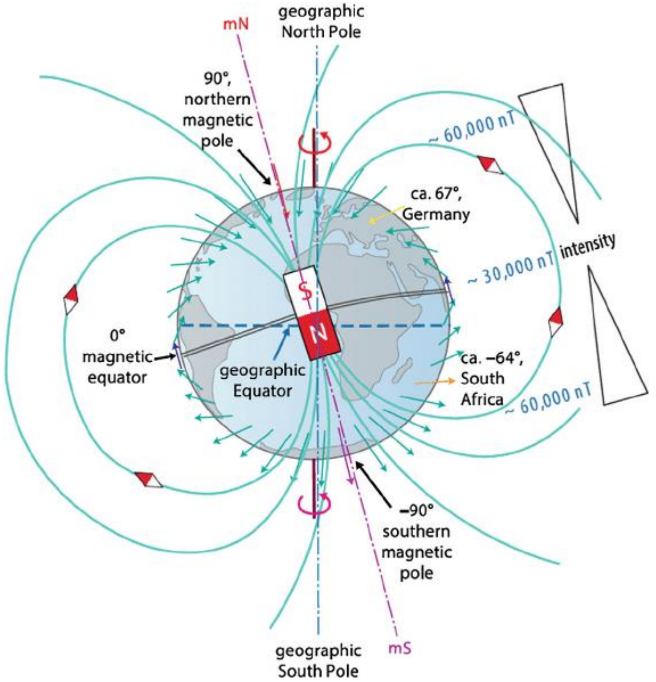
Magnetometer

- **Magnetometer**: measures magnetic field strength and direction with respect to body axis
- Typical accuracy 1-15°, varies by quality of sensor and implementation
- Can be used as science payloads or attitude determination
- Consumer COTS ($10) to precision instruments ($1000s)
- Placement and operational concerns
  - Spacecraft induced dipole (esp. magnetorquers, RWs) can ruin measurements (20°+)
  - Place far from high current components and ferromagnetic materials (stainless steel, regulators, etc.)
  - Must use up-to-date Earth magnetic field models (~1° error)
  - Tilted dipole model of earth's field
  - Not useful on bodies/environments with unknown magnetic models



Helium vector magnetometer of Pioneer 10 and 11 spacecraft

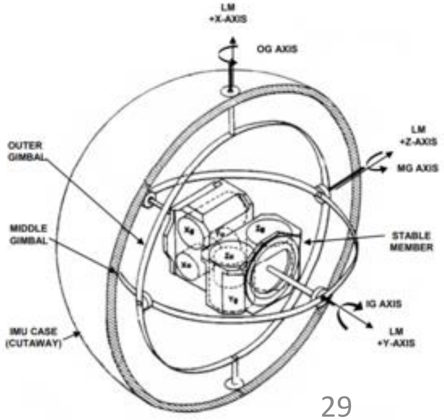
COTS sensor-on-a-chip



IMU

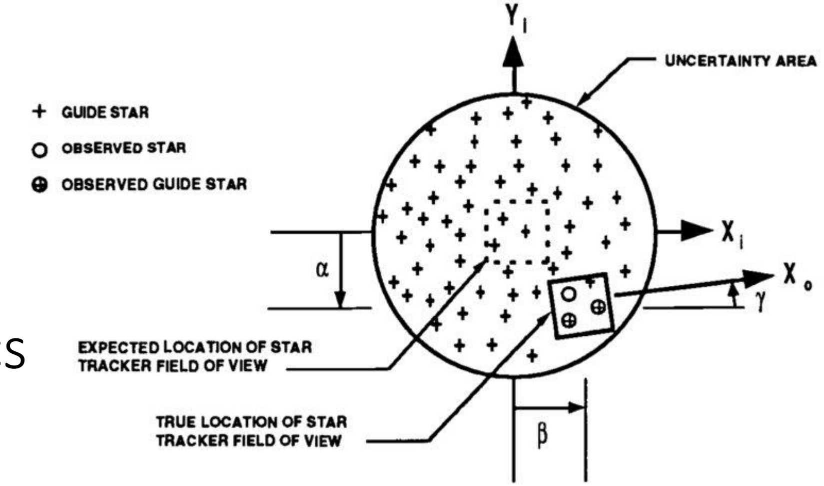
- Inertial Measurement Unit (**IMU**)
  - **Gyro**: measures angular velocity with respect to the body axis
  - **Accelerometer**: measures acceleration with respect to body axis
- Consumer COTS ($10) to precision instruments ($10k)
  - Mechanical systems (Apollo Capsule IMU)
  - Micro Electro Mechanical System (MEMS), "sensor on a chip", < 10 deg/hr
  - Fiber Optic Gyros (FOGs), "military grade", < 0.01 deg/hr
- Placement and operational concerns
  - Only useful when using Kalman filter; does not provide attitude determination solution alone
  - Notorious for bias, must calibrate or characterize
  - Electrical noise can interfere, place away from high current/dipole areas
  - Integrating rates without other reference sensor measurements will diverge

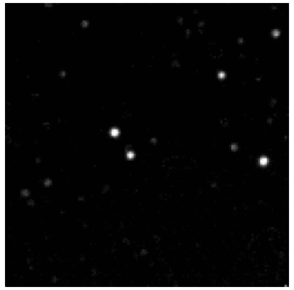

Apollo Capsule IMU



Star Tracker

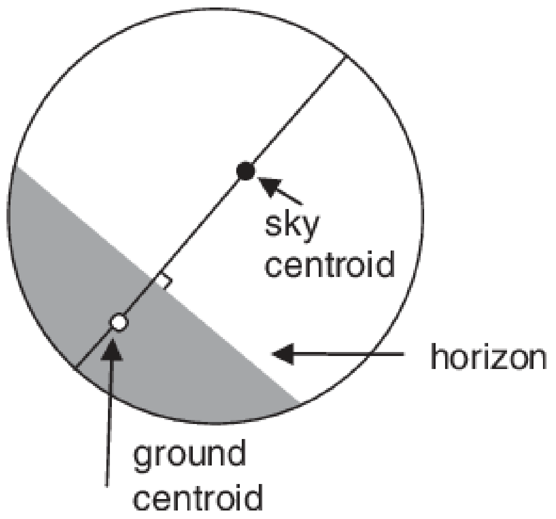
- **Star tracker**: determines attitude through comparing visible stars to star catalogue
- Accuracy down to < 10 arc-seconds (0.003°)
  - Less accurate (< 100 arc-seconds) about boresight
- Minimum ¼ U size, depending on baffle and quality of optics
- Placement and operational concerns
  - Baffle for sun and earth keep-out (30°+)
    - Not uncommon to use 2 orthogonal trackers to improve probability of continuous coverage, Sensitive optics can be damaged if active when pointing at the sun
  - Must keep slew rate down (< 2°/s typical)
  - 5-25° FOV typical
  - Thermal and mechanical misalignment must be controlled or calibrated out
  - Cleanliness of the optics

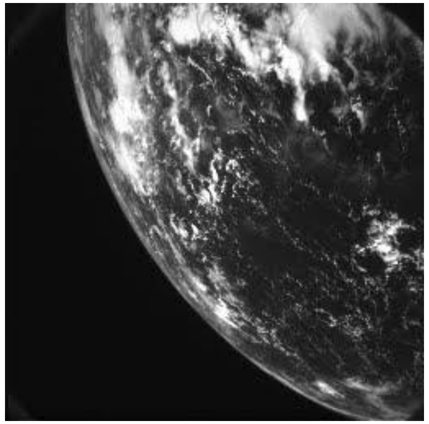


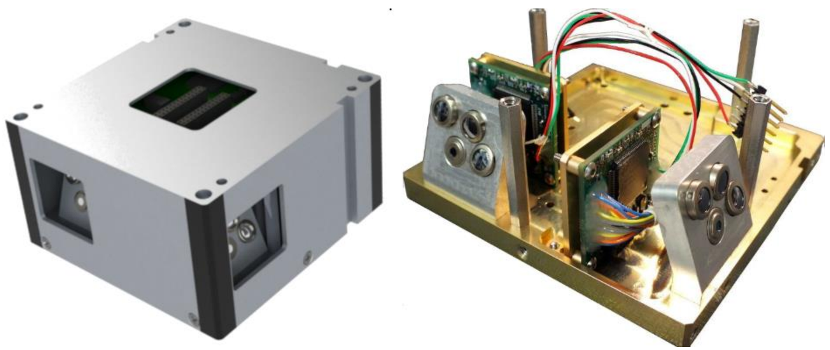


Horizon/IR Sensor

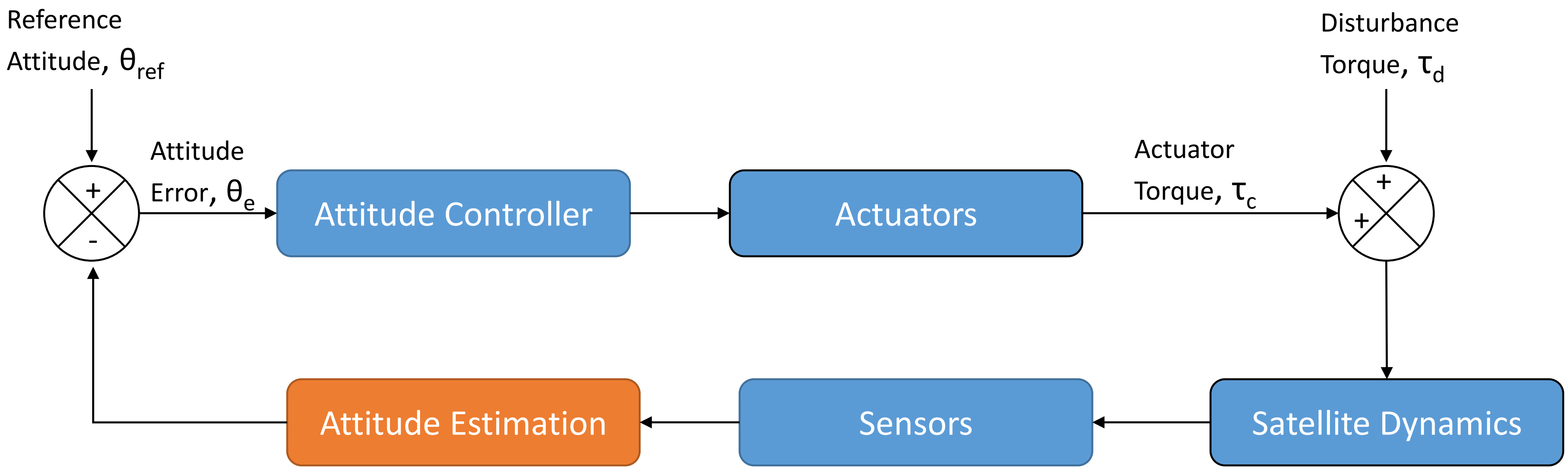
- **Horizon sensor**: orients through viewing edge of earth's atmosphere
- Fixed telescopes for nadir pointing, accuracy $< 0.1°$
- Cheaper electronics for scanning, accuracy $< 1°$
  - Earth limb is about 2° wide depending on orbit
- Most useful for mission-specific pointing
  - Can achieve full 'nadir' attitude solution with 3 simultaneous measurements







Attitude Estimation



Attitude Estimation

- Deterministic Methods, e.g. “TRIAD” or “Q-method”
  - Minimum of 2 measured vectors whose directions in reference frame are known
    - Most commonly sun and magnetic field direction
  - In TRIAD specifically, one vector given priority and attitude solution is rotated such that other vector is aligned as closely as possible
- Probabilistic Estimator, aka some flavor of Kalman Filter
  - Wide variety of potential measurements and estimated elements (“state”)
    - State is typically position, attitude, angular rates, but also can include sensor bias estimate
    - Can incorporate all sensors (and their performance) to achieve probabilistically optimal estimate
  - At each time step, propagate state with model of dynamics, then update estimated state based on how well sensor readings match expected measurements
  - Must initialize with a beginning estimated state, typically use deterministic method

Typical Attitude Determination Systems

0-axis

- Nothing

1-axis

- Sun sensor (on all faces) only when sunlit, no positional knowledge
- Magnetometer, no positional knowledge

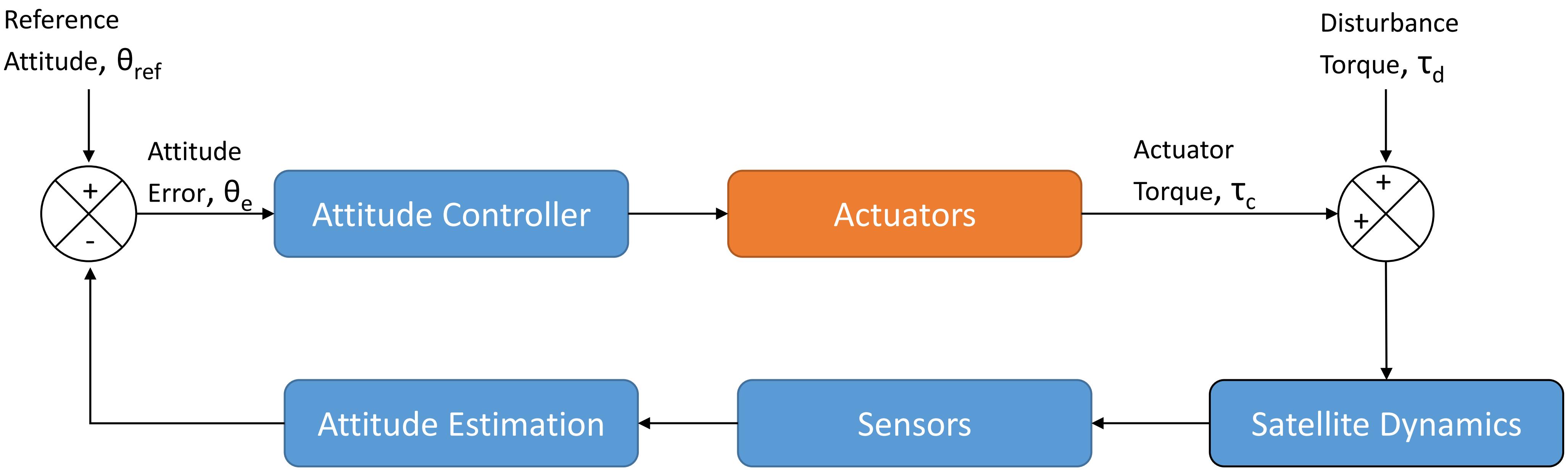
2-axis

- Sun sensor (on all faces) only when sunlit, positional knowledge
- Magnetometer, positional knowledge
- 3x Horizon Sensors

3-axis

- Sun sensor (on all faces), magnetometer [only when sunlit]
- 1x Star tracker

Actuators



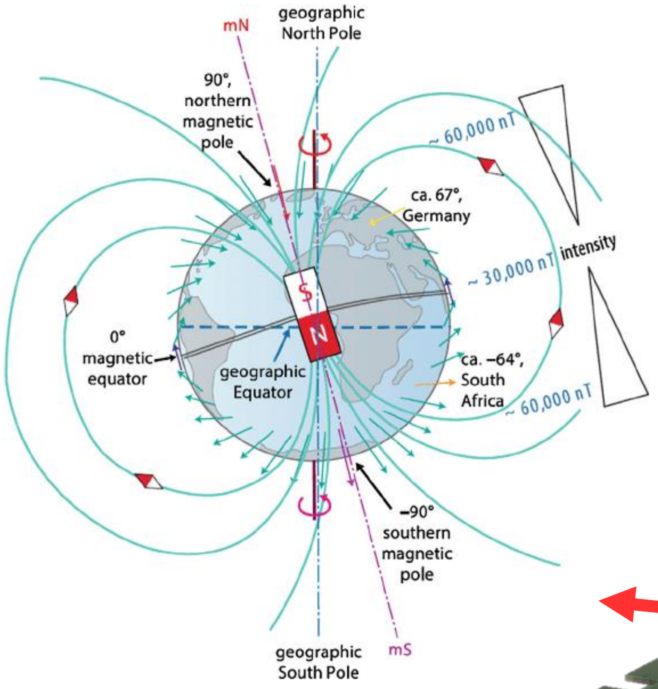
Reference
Attitude, θref
Attitude
Error, θe
+
-
Attitude Controller
Actuators
Actuator
Torque, τc
Disturbance
Torque, τd
+
+
Satellite Dynamics
Sensors
Attitude Estimation

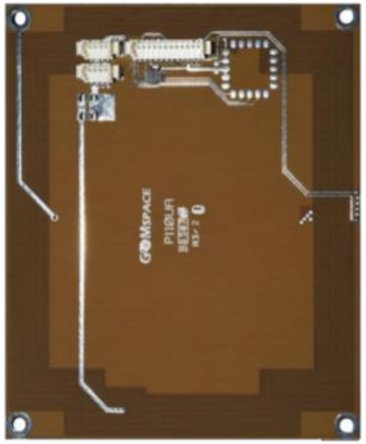
CubeSat Actuator Overview

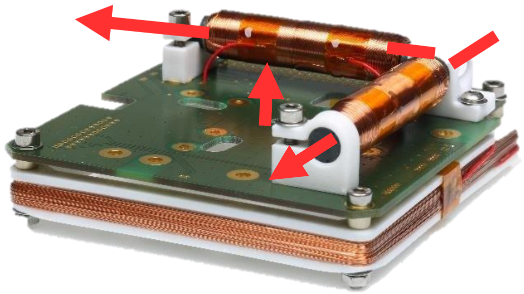
- Magnetorquers
- Reaction Wheels
- Propulsion
- Others not covered here
  - Control Moment Gyros
  - Drag devices: Sails/deployables

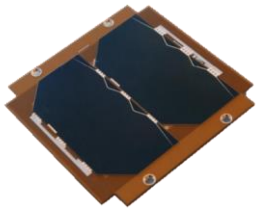
Magnetorquers

- Requires magnetic field
  - Often used in low-earth orbit
- Commonly used for:
  - Initial acquisition maneuvers (i.e. detumble)
  - Coarse attitude control
  - Momentum dumping in reaction wheel systems



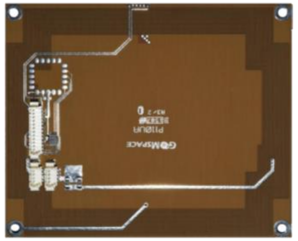






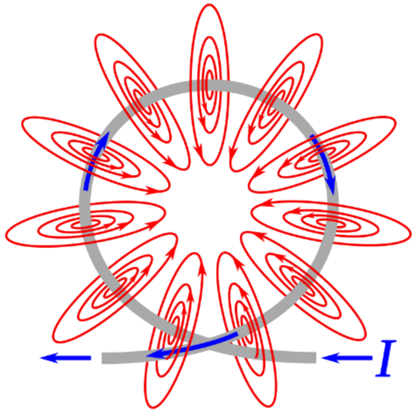
How Magnetorquers work

- Torque rods are long helical coils
- Use current to generate a magnetic field
- This field will try to align with the Earth's magnetic field, thereby creating a torque on the spacecraft
- Size of dipole varies based on application
  - 6U CubeSat 0.2 A•m^2 dipole = 0.01 mNm torque

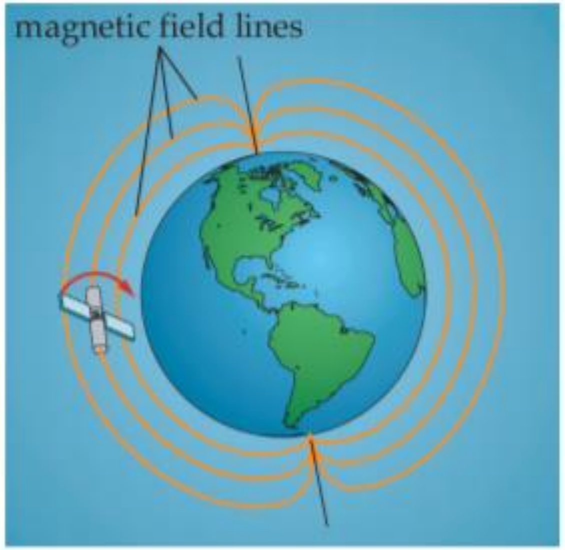


Solar Panel Embedded Coils

Torque Rods



Magnetic Field generated here is INTO the page

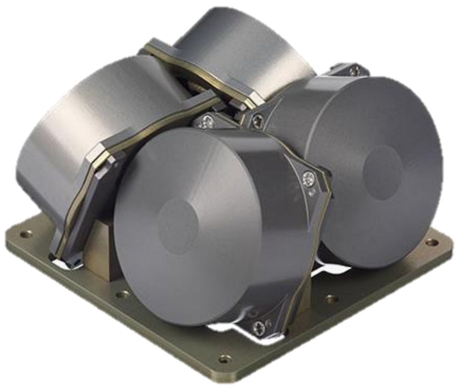


Magnetorquer Pros & Cons

Advantages	Disadvantages
• Doesn’t consume fuel • Easily duty cycled to meet power needs • No moving parts • Can detumble before complete attitude solution is determined • Control logic simple for independent axes • “External” torque allows angular momentum to leave the spacecraft	• **Can only generate torque in plane perpendicular to external magnetic field** • Residual dipole can ruin magnetometer measurements • Residual dipole limits how quickly direction of generated torque can change • Relies on presence of external magnetic field • Generates small amount of torque • Earth magnetic field direction changes throughout orbit

Reaction Wheels

- Flywheel with fixed rotation axis, variable rotation speed
- Provide fast; continuous feedback control
- Number of Axis-control = number of wheels each axis
- Sizing varies widely based on application
  - CubeSat: 1 mNm torque, 10 mNms storage, < 1 W
  - Flagship: 100 mNm torque, 10k+ mNms storage, 20 W

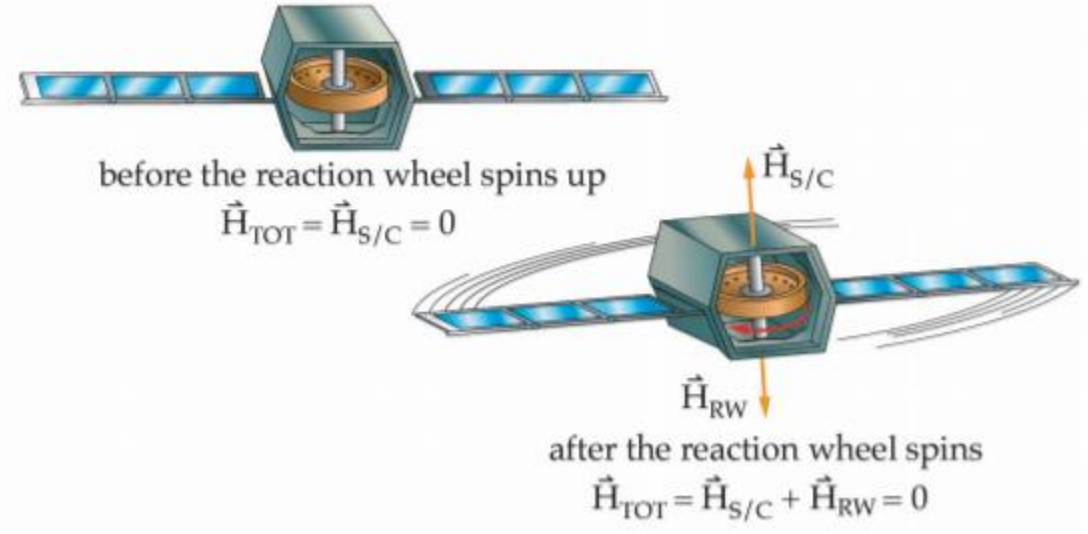


Four wheels allows for three-axis control in the event of single wheel failure

How Reaction Wheels Work

- Torques are created on a spacecraft by creating equal but opposite torques on Reaction Wheels (flywheels on motors)
- For three-axes of torque, three wheels are necessary. Use four wheels for redundancy
- If external torques exist, wheels will angularly accelerate to counteract these torques
- They will eventually reach a max RPM limit at which time they must be desaturated

Conservation of angular momentum



Ang. Momentum = MOI x Ang. Velocity

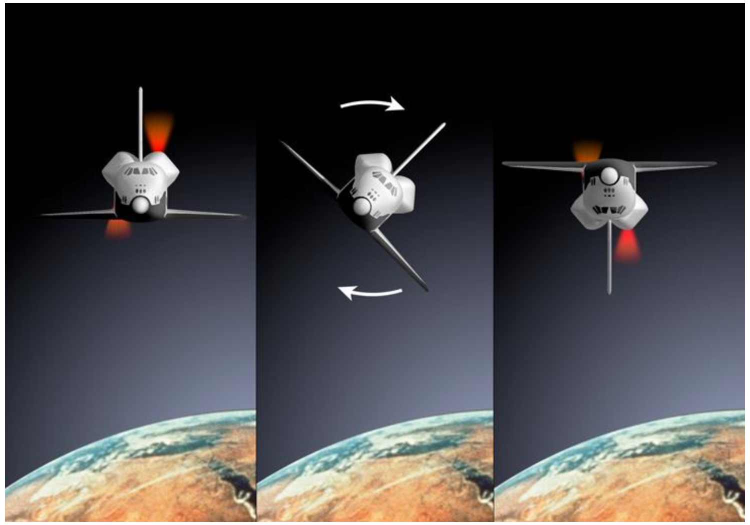
$$\vec{H} = I * \vec{\omega}$$

Reaction Wheel Pros and Cons

Advantages	Disadvantages
• Doesn't consume fuel • Relatively power efficient • Doesn’t rely on a mag. field • Generate torque in any direction • Stores angular momentum • Achieve precision pointing • Fast; continuous feedback control • Control logic simple for independent axes (gets complicated with redundancy)	• Mechanical moving part • Requires lubrication • Usually operate around some nominal spin rate to avoid stiction effects. • Saturation • Internal torque only; external still required for ‘momentum dumping’ • Relatively high power, weight, cost • Jitter and mechanical misalignment • Static & dynamic imbalances can induce vibrations (mount on isolators)

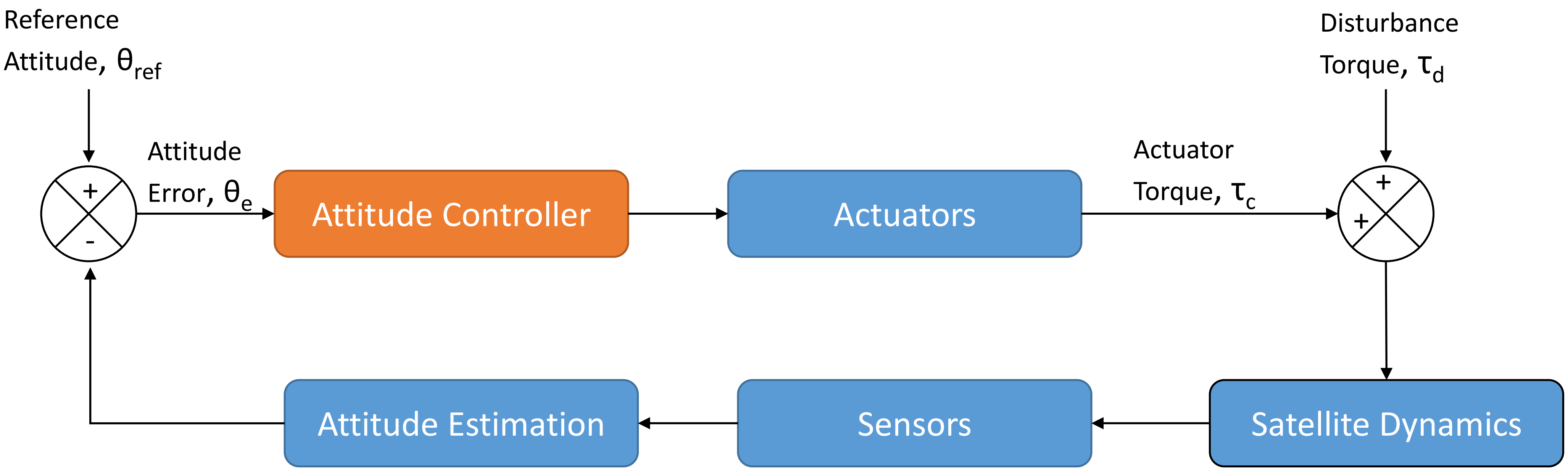
Reaction Control System (RCS)

- Typically cold gas propulsion
- Quick, effective way to dump momentum
- 1 mNm – 10 Nm torque, 10-10k Nms storage
  - Varies based on propellant type, thruster size, length of moment arm, fuel storage
  - CubeSat typically 1-100 mNm, 10-100 Nms range
- Operational concerns
  - Generally cannot generate pure torque, creates some undesirable translational force
  - Minimum bit results in uncontrollable deadband
  - Special controls to reduce fuel usage
  - Amount of fuel directly determines mission lifetime
    - Unmodeled disturbances increase fuel usage, but too much fuel wastes mass and volume



Double canted thrusters give control in all three axis

Attitude Controller



Attitude Controller

- Receive data from vehicle sensors and derive the appropriate commands to the actuators to rotate the vehicle to the desired attitude.
- Algorithms range from very simple, e.g. proportional control, to complex nonlinear or adaptive functions, depending on mission requirements.
  - Remember to KISS!
- The design of the control algorithm depends on the actuator to be used for the specific attitude maneuver although using a simple proportional–integral–derivative controller (PID controller) satisfies most control needs.
  - Prop systems most common cause to use non-linear control

is the control torque, is the attitude deviation signal, and are the PID controller parameters.

PID Control Example for a Reaction Wheel

Proportional | Integral | Derivative

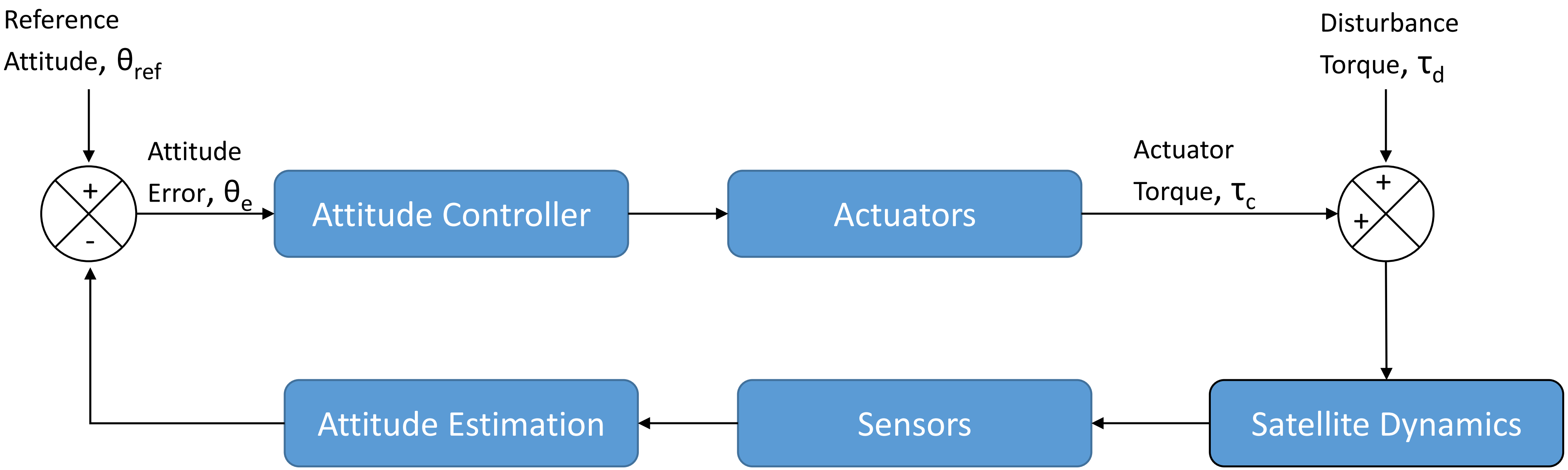
$$T_c(t) = K_\mathrm{p} e(t) + K_\mathrm{i} \int_0^t e(\tau)\, d\tau + K_\mathrm{d} \dot{e}(t),$$

Control Torque | Attitude Error | History of Error | Error Rate of Change

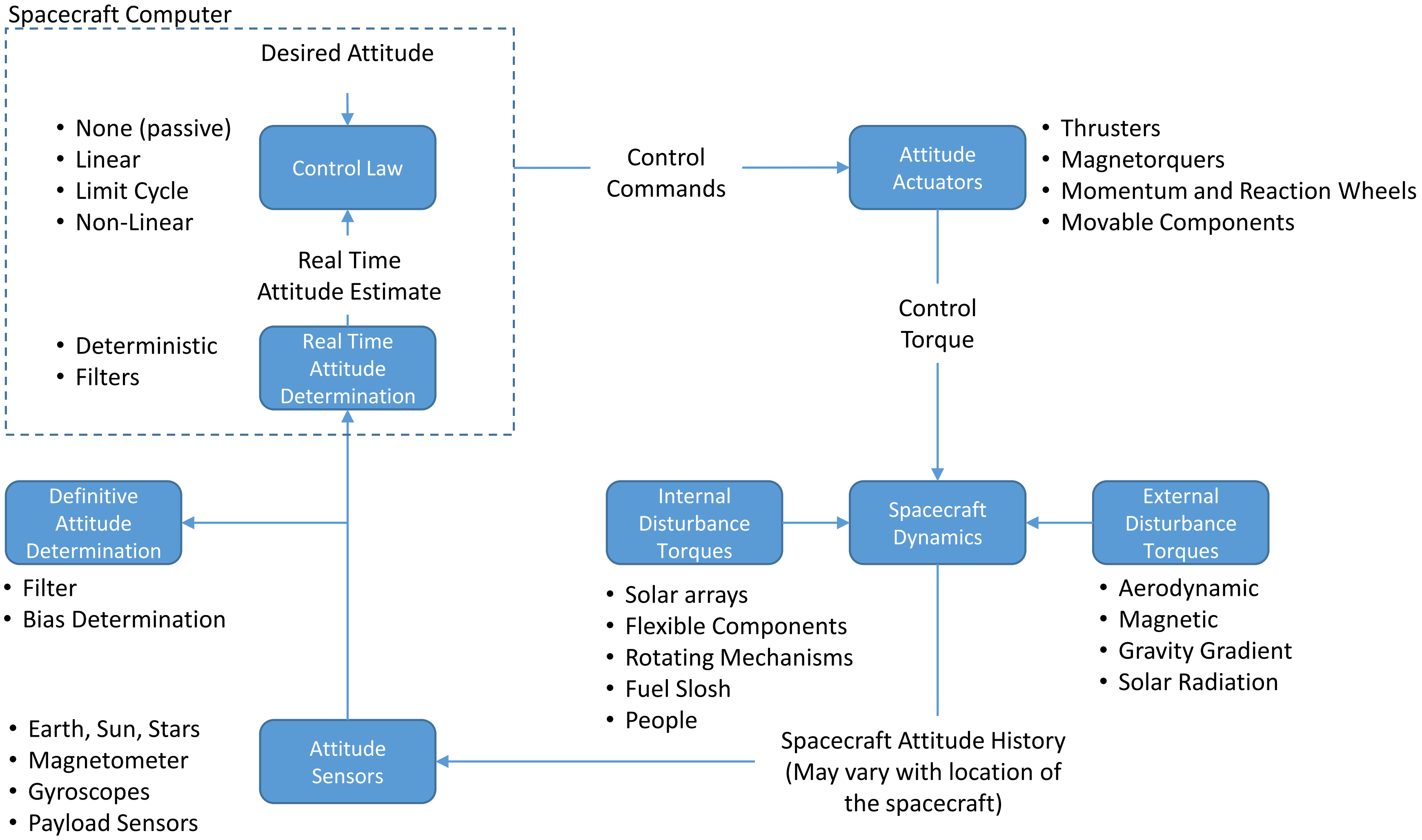
Corresponds to a Pulse Width Modulation value applied on a motor controller to change the Reaction Wheel direction or accelerate/decelerate

Values from Sensors

ADCS Summary



ADCS Summary

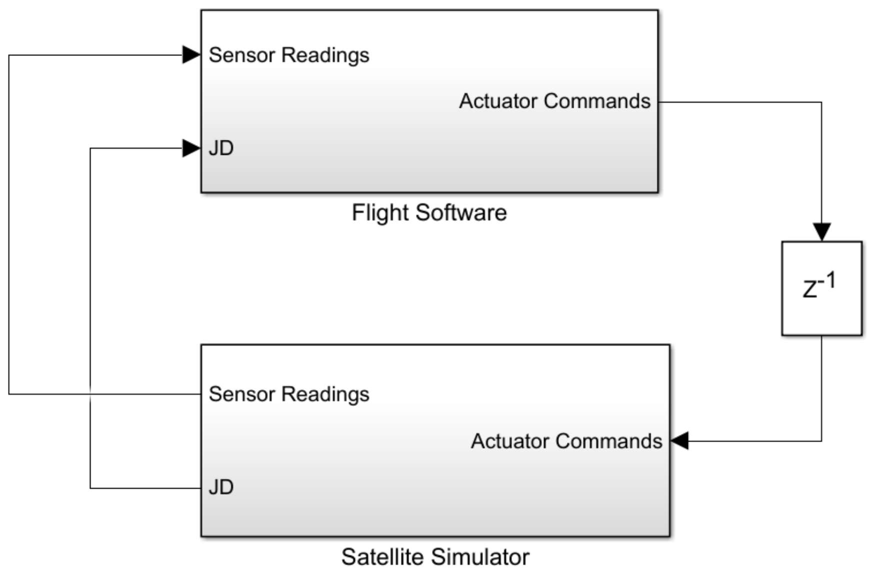


Other System Considerations

- On-board computer, processing power
- Ground processing
- **Simulation**
- **Testing**
- Maneuvers
  - Attitude/Translation Coupling

ADCS Modeling and Simulation

- Flight Software half
  - Running on the spacecraft (more or less)
- Spacecraft + Environmental Model half
  - Dynamics and disturbances
  - Model fidelity vs time and effort
- Helps guide and validate mission design
- Answer questions to reduce mission risk at lower cost and effort than physical testing. For example:
  - "What is the impact on mission success if we suffer a reaction wheel failure?"
  - "How long will detumble last?"
  - "What is the expected pointing error if our sun sensors are not calibrated correctly?"
  - "How goes the center of mass offset from center of geometry affect fuel use over mission lifetime?
- Helps guide analysis of downlinked health data



ADCS Testing

- "Test-as-you-fly" mentality
  - Risk reduction efforts
  - FSW development
  - End-to-end tests
- Procedures
  - Test plan, "as-run"
  - Documentation, signatures
- ADCS-specific challenges
  - Environmental disturbances
  - Measuring actuator response
  - Complicated end-to-end behavior

Fig. 1: CHOMPTT working on its tan

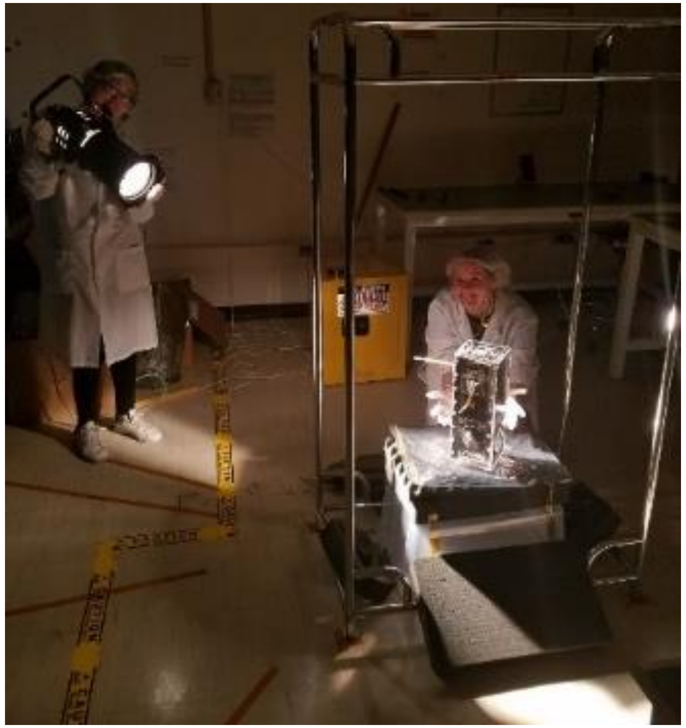


Fig. 2: Million dollar knots

QUESTIONS?

Adam Zufall (adam.b.zufall@nasa.gov)

More resources here:

- https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/20110007876.pdf
- https://www.faa.gov/about/office_org/headquarters_offices/avs/offices/aam/cami/library/online_libraries/aerospace_medicine/tutorial/media/III.4.3.1_Space_Vehicle_Control_Systems.pdf

Questions