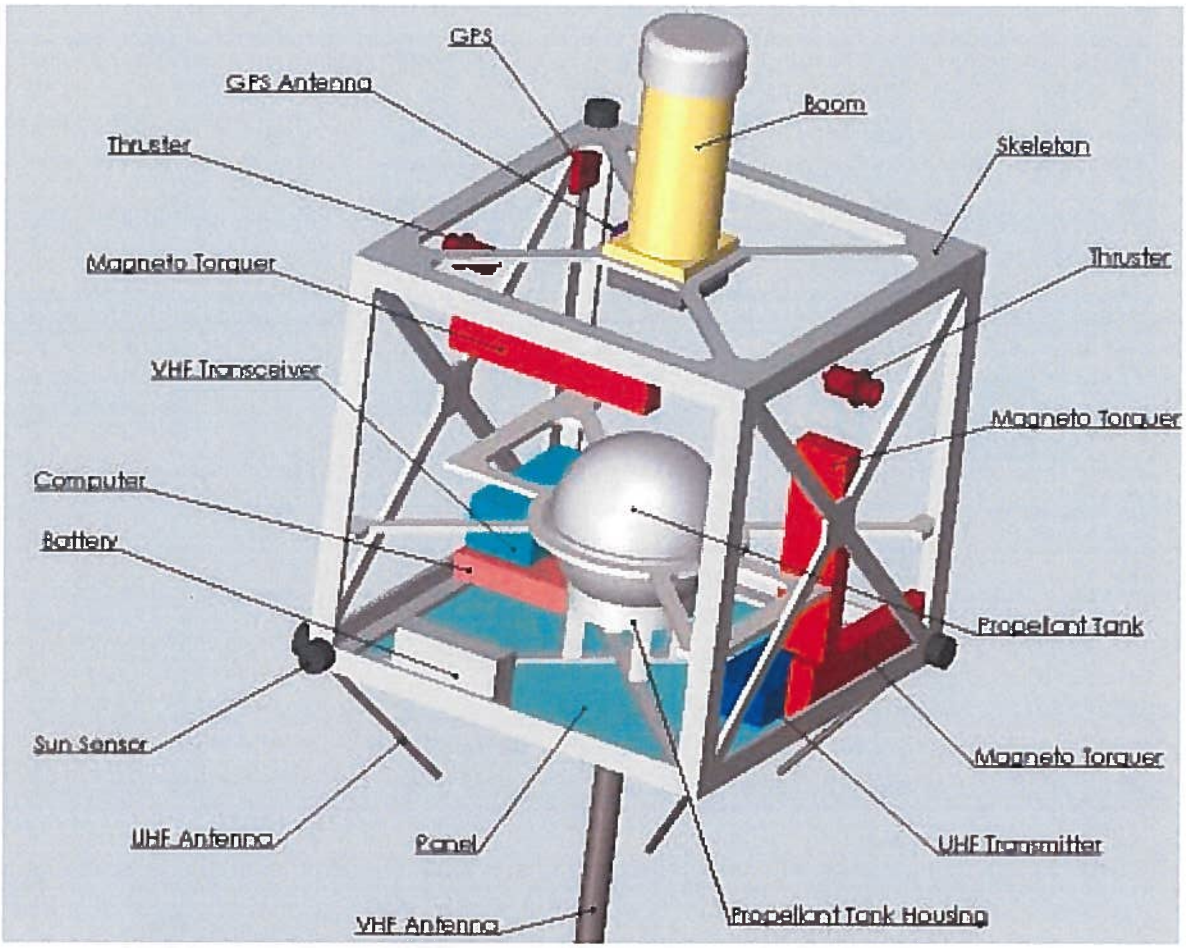
Attitude Measurement



Text 19.1: Attitu de Determination & Control

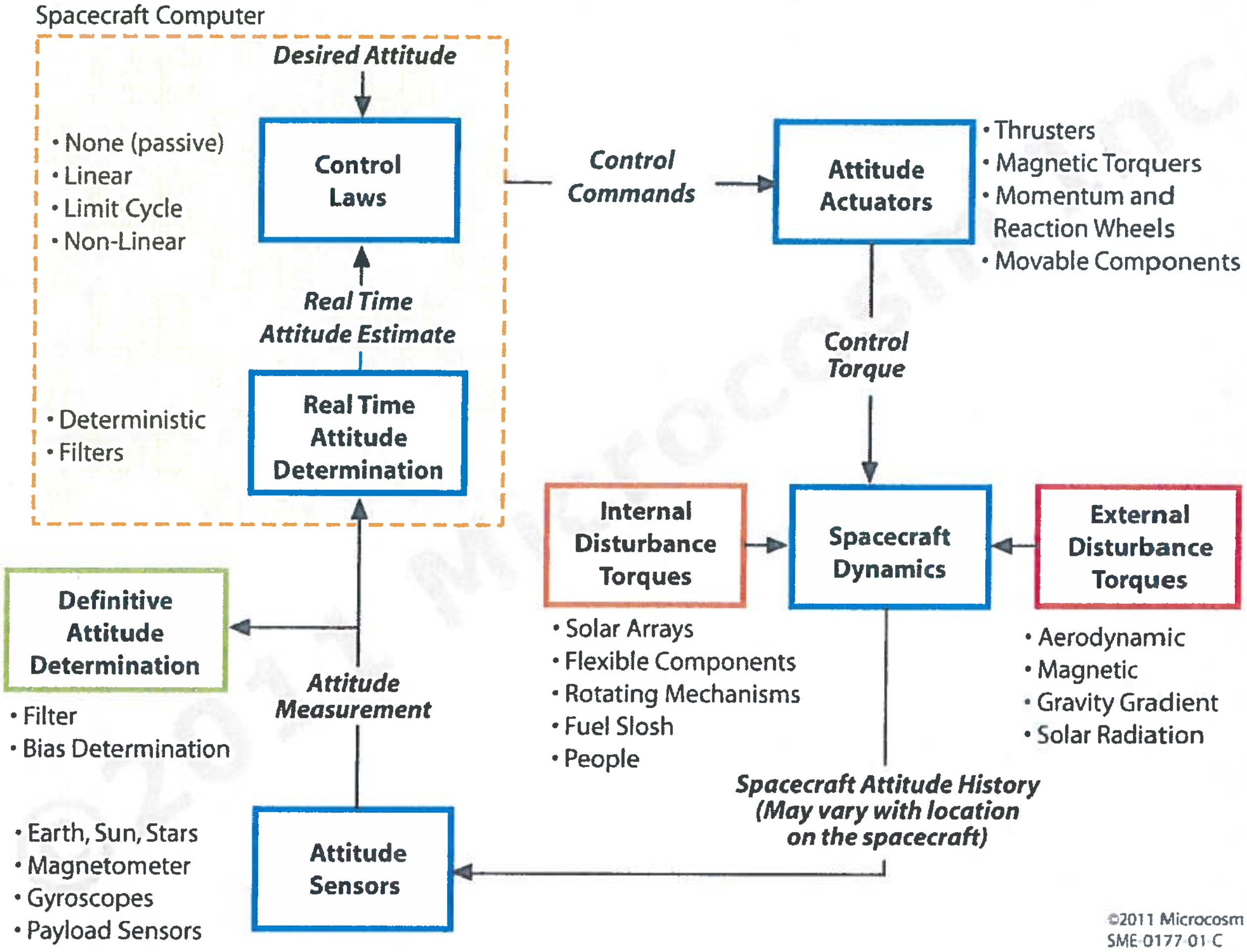


Fig. 19-2. Diagram of a Complete Attitude Determination and Control System. Definitive attitude determination usually occurs in ground processing of telemetry, whereas onboard, real-time determination design focuses on being extremely reliable and deterministic in its operation.

Attitude Sensing : generally >1 type of sensor

Attitude Control : Active and/or Passive methods

↑

let environment work for you

ADCS Motivation

- Motivation
  - In order to point and slew optical systems, spacecraft attitude control provides coarse pointing while optics control provides fine pointing
- Spacecraft Control
  - Spacecraft Stabilization
    - Spin Stabilization
    - Gravity Gradient
    - Three-Axis Control
    - Formation Flight
  - Actuators
    - Reaction Wheel Assemblies (RWAs)
    - Control Moment Gyros (CMGs)
    - Magnetic Torque Rods
    - Thrusters
  - Sensors: GPS, star trackers, limb sensors, rate gyros, inertial measurement units
  - Control Laws
- Spacecraft Slew Maneuvers
  - Euler Angles
  - Quaternions

Key Question:
What are the pointing requirements for satellite ?

NEED expendable propellant:

- **On-board fuel often determines life**
- **Failing gyros are critical (e.g. HST)**

Opening Remarks

- Nearly all ADCS Design and Performance can be viewed in terms of RIGID BODY dynamics
- Typically a Major spacecraft system
- For large, light-weight structures with low fundamental frequencies the flexibility needs to be taken into account
- ADCS requirements often drive overall S/C design
- Components are cumbersome, massive and power-consuming
- Field-of-View requirements and specific orientation are key
- Design, analysis and testing are typically the most challenging of all subsystems with the exception of payload design
- Need a true “systems orientation” to be successful at designing and implementing an ADCS

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Terminology

ATTITUDE : Orientation of a defined spacecraft body coordinate system with respect to a defined external frame (GCI,HCI)

ATTITUDE DETERMINATION: Real-Time or Post-Facto knowledge, within a given tolerance, of the spacecraft attitude

ATTITUDE CONTROL: Maintenance of a 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.

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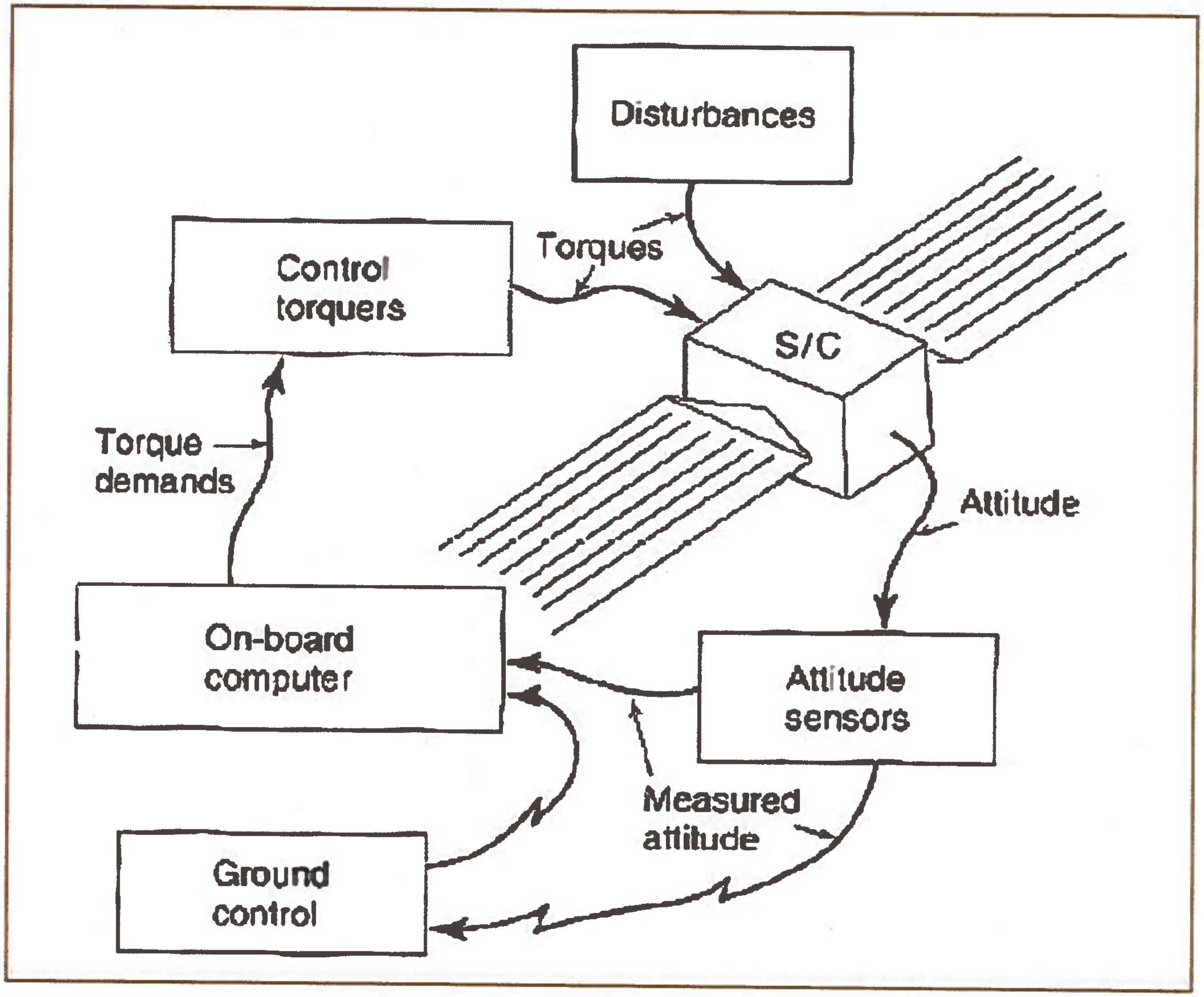
Attitude Control System



Introduction

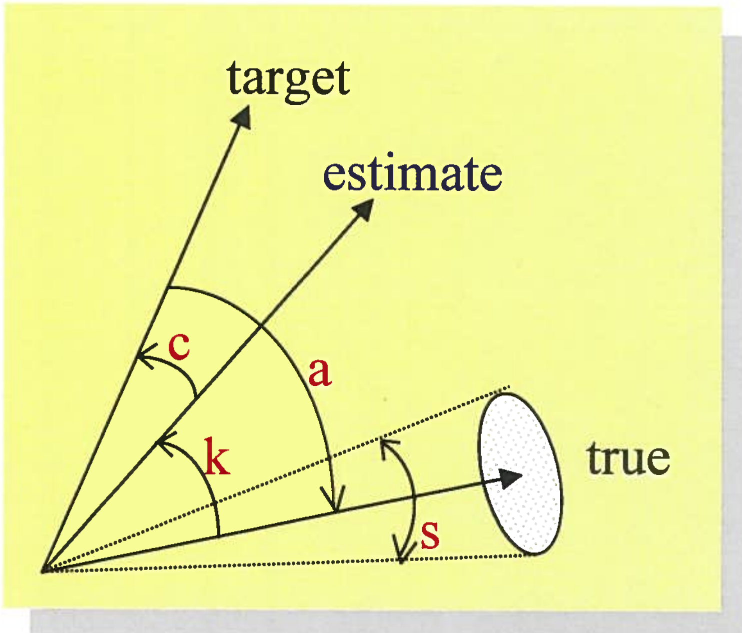
- ADCS Design Requirements and Constraints
  - Pointing Accuracy (Knowledge vs. Control)
    - Drives Sensor Accuracy Required
    - Drives Actuator Accuracy Required
  - Rate Requirements (e.g. Slew)
  - Stationkeeping Requirements
  - Disturbing Environment
  - Mass and Volume
  - Power
  - Reliability
  - Cost and Schedule

ADCS Design

TABLE 11-2. **Typical Attitude Control Modes.** Performance requirements are frequently tailored to these different control operating modes.

Mode	Description
Orbit Insertion	Period during and after boost while spacecraft is brought to final orbit. Options include no spacecraft control, simple spin stabilization of solid rocket motor, and full spacecraft control using liquid propulsion system.
Acquisition	Initial determination of attitude and stabilization of vehicle. Also may be used to recover from power upsets or emergencies.
Normal, On-Station	Used for the vast majority of the mission. Requirements for this mode should drive system design.
Slew	Reorienting the vehicle when required.
Contingency or Safe	Used in emergencies if regular mode fails or is disabled. May use less power or sacrifice normal operation to meet power or thermal constraints.
Special	Requirements may be different for special targets or time periods, such as eclipses.

Pointing Control Definitions



target	desired pointing direction
true	actual pointing direction (mean)
estimate	estimate of true (instantaneous)
a	pointing accuracy (long-term)
s	stability (peak-peak motion)
k	knowledge error
c	control error

a = pointing accuracy = attitude error
s = stability = attitude jitter

Source:
G. Mosier
NASA GSFC

ADCS Design

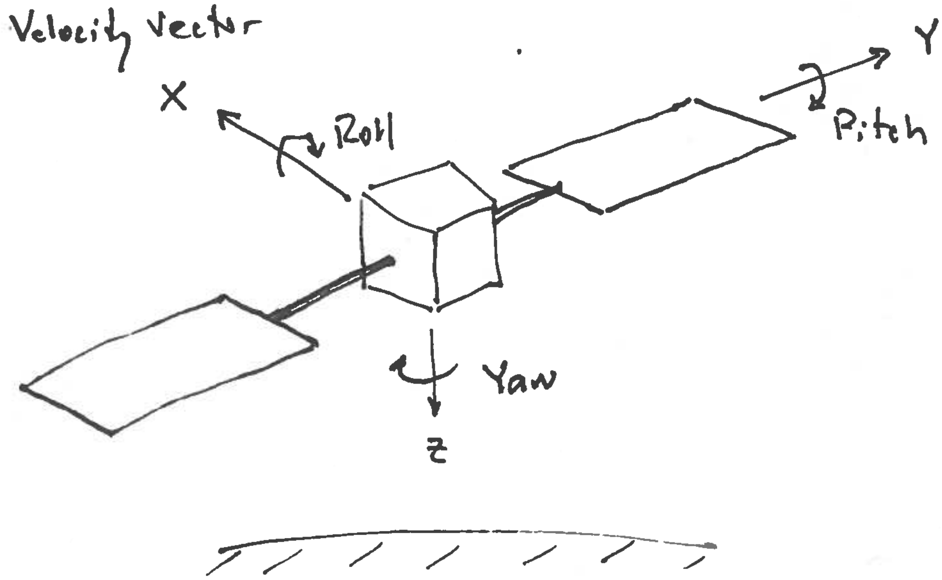
TABLE 11-3. Typical Attitude Determination and Control Performance Requirements. Requirements need to be specified for each mode. The following lists the areas of performance frequently specified.

Area	Definition*	Examples/Comments
	DETERMINATION	
Accuracy	How well a vehicle's orientation with respect to an absolute reference is known	0.25 deg, 3 σ, all axes; may be real-time or post-processed on the ground
Range	Range of angular motion over which accuracy must be met	Any attitude within 30 deg of nadir
	CONTROL	
Accuracy	How well the vehicle attitude can be controlled with respect to a commanded direction	0.25deg,3 σ;includesdeterminationand controlerrors,maybetakenwithrespect to an inertial or Earth-fixed reference
Range	Range of angular motion over which control performance must be met	All attitudes, within 50 deg of nadir, within 20 deg of Sun
Jitter	A specified angle bound or angular rate limit on short-term, high-frequency motion	0.1 deg over 1 min, 1 deg/s,1 to 20 Hz; usually specified to keep spacecraft motion from blurring sensor data
Drift	A limit on slow, low-frequency vehicle motion. Usually expressed as angle/time.	1 deg/hr, 5 deg max. Used when vehicle may drift off target with infrequent resets (especially if actual direction is known)
Settling Time	Specifies allowed time to recover from maneuvers or upsets.	2 deg max motion, decaying to < 0.1 deg in 1 min; may be used to limit overshoot, ringing, or nutation

* Definitions vary with procuring and designing agencies, especially in details (e.g., 1 or 3 σ, amount of averaging or filtering allowed). It is always best to define exactly what is required.

Text 19.1: Attitude Determination & Control

ACDS controls angular momentum of spacecraft



- Equation of motion (from Euler's eqn. for rotational dynamics)

$$\underbrace{[I]\,\dot{\vec{\omega}}} = \underbrace{\vec{T}} - \underbrace{\dot{\vec{h}}} - \underbrace{[\dot{I}]\,\vec{\omega}} - \underbrace{\vec{\omega} \times \vec{H}}$$

rate of change of angular momentum

attitude change rate

external torque, including thrusters, etc

change rate of ang momentum of onboard rotating masses; flywheels, gyros, spinning sections (internal torques)

changes in mass distribution: propellant, deployed mechanisms, etc

"gyroscopic torque"
Change of direction of angular momentum due to spacecraft ref frame rotation rate ω in inertial reference frame

Comparison of Attitude Descriptions

Method	Euler Angles	Direction Cosines	Angular Velocity $\underline{\omega}$	Quaternions
Pluses	If given ϕ, ψ, θ then a unique orientation is defined	Orientation defines a unique dir-cos matrix **R**	Vector properties, commutes w.r.t addition	Computationally robust Ideal for digital control implement
Minuses	Given orient then Euler non-unique Singularity	6 constraints must be met, non-intuitive	Integration w.r.t time does not give orientation Needs transform	Not Intuitive Need transforms

Best for analytical and ACS design work

Must store initial condition

Best for digital control implementation

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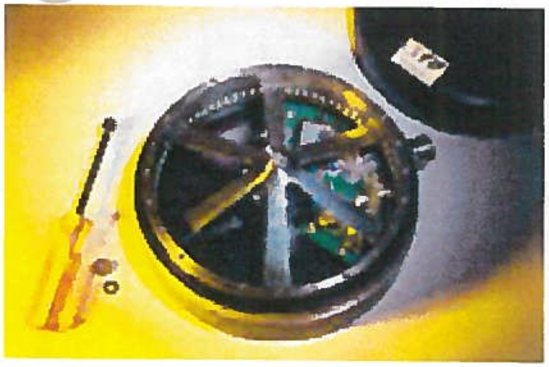
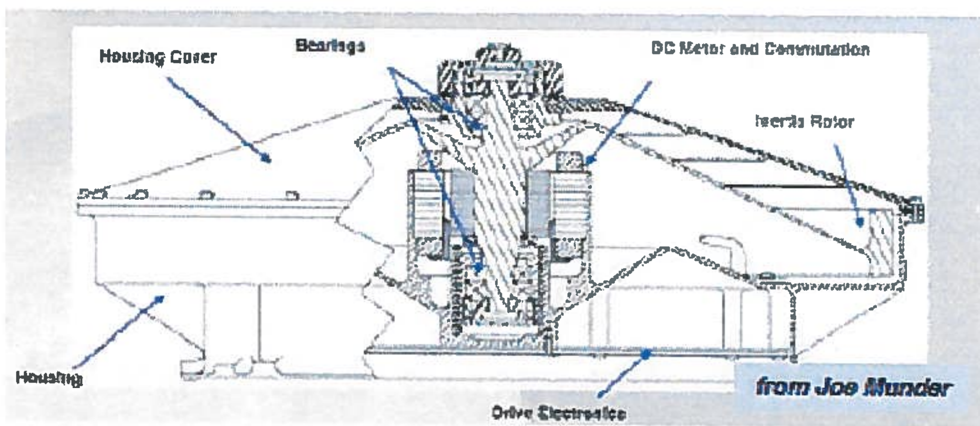


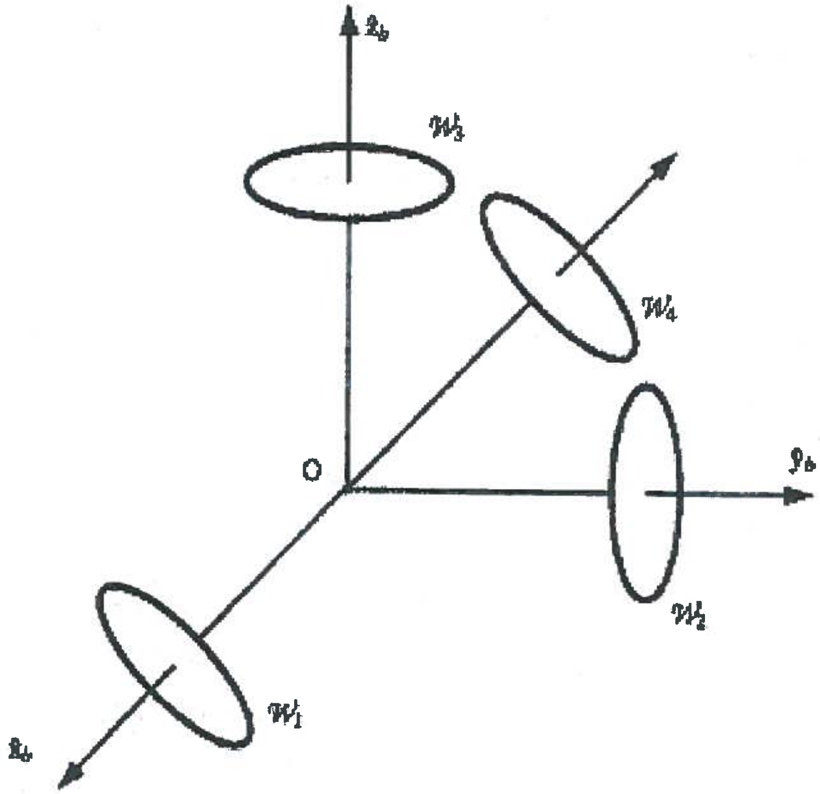
Momentum/ Reaction Wheels

- Flywheel on a motor shaft
- **Momentum wheel** operates at high rpm and provides spin stability
- **Reaction wheel** rpm is varied to trade angular momentum with the spacecraft for control
  - Three orthogonal wheels vary all components of angular momentum
  - Fourth wheel at oblique angle provides redundancy

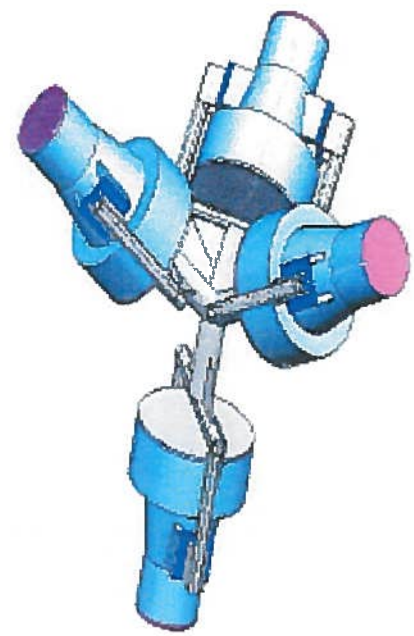


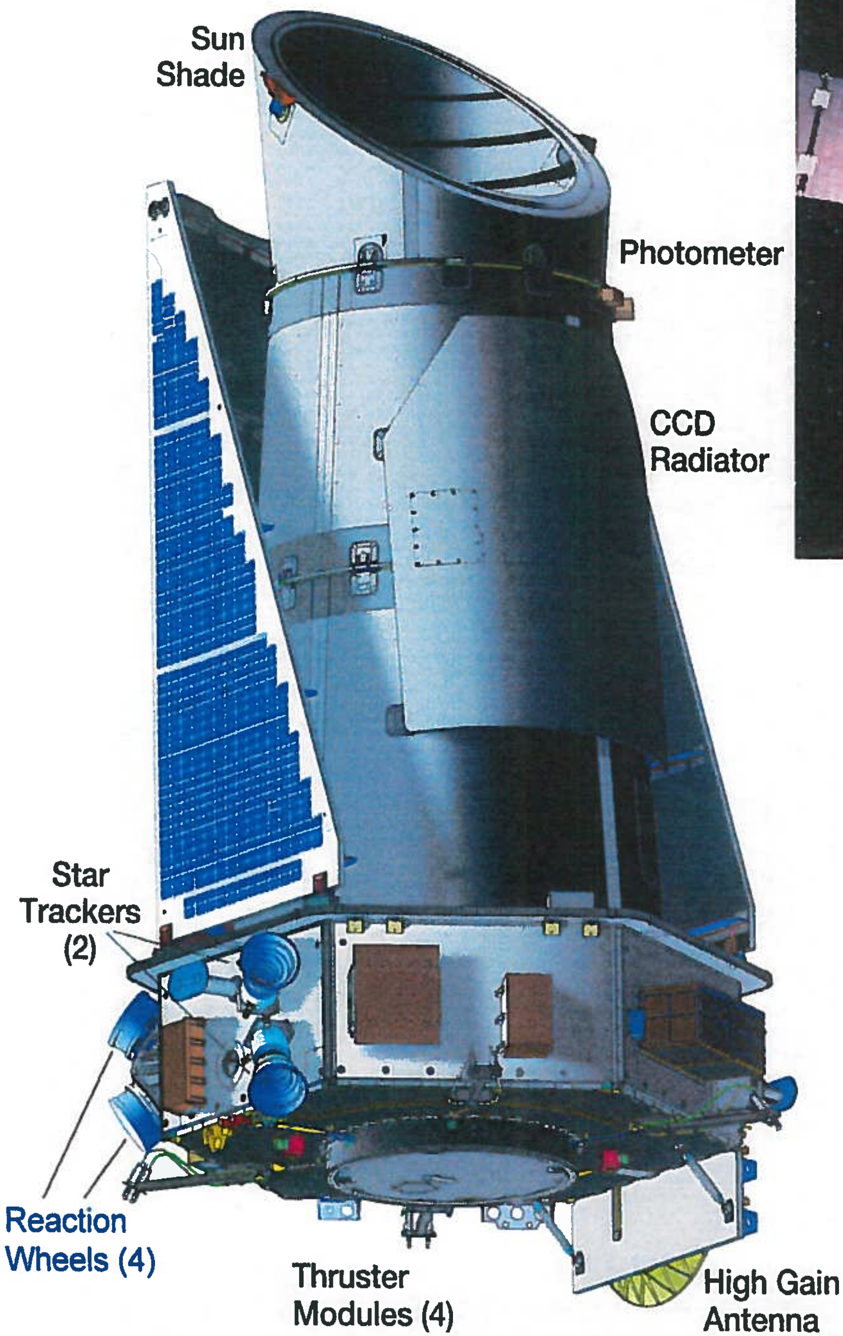
from Joe Munder

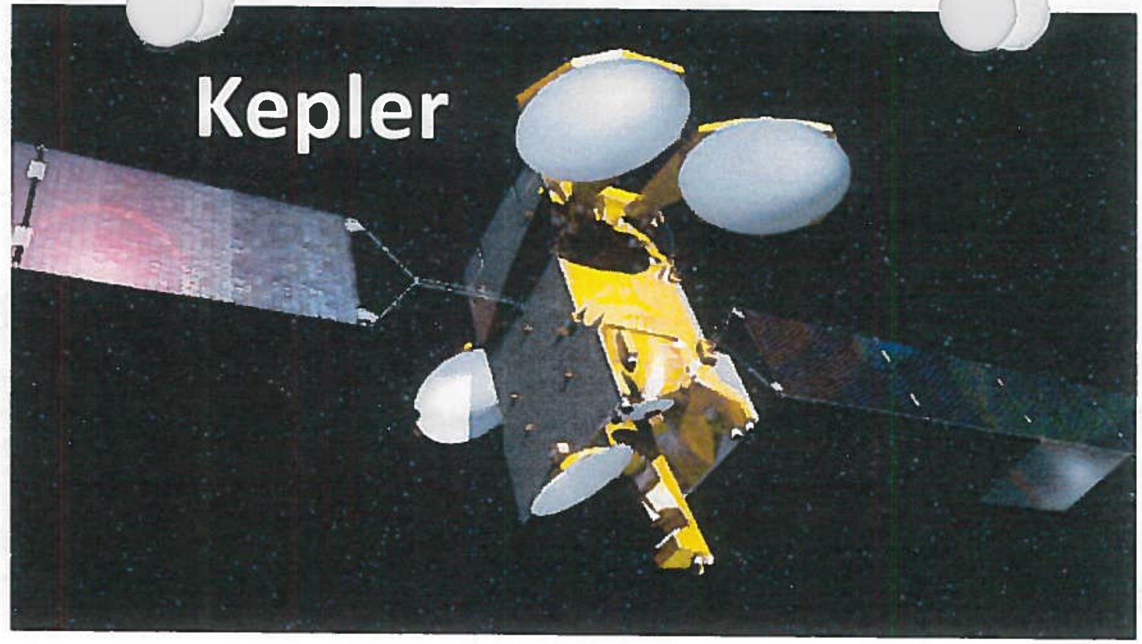
Tetrahedron configuration of Reaction wheels

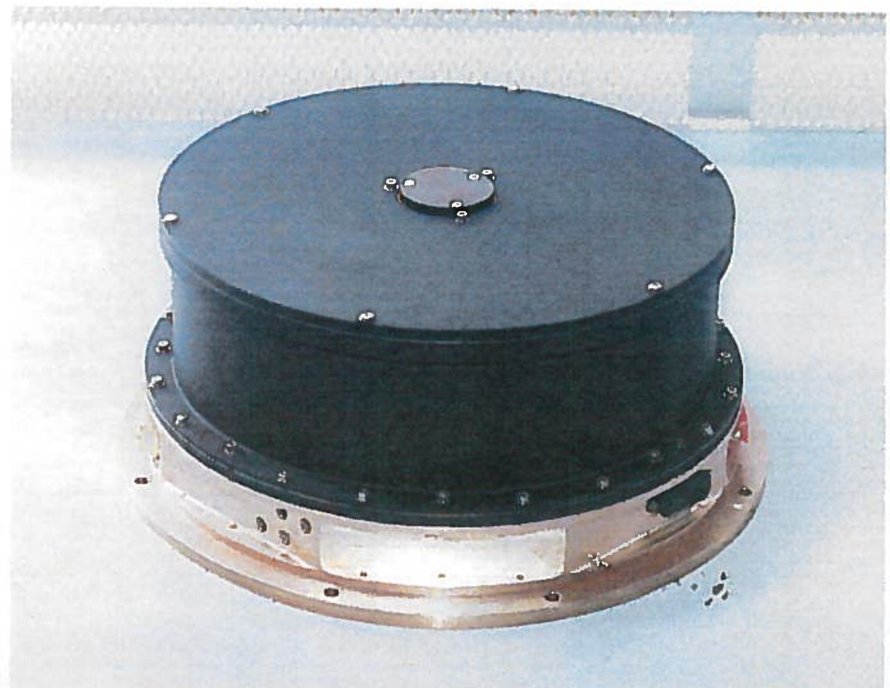


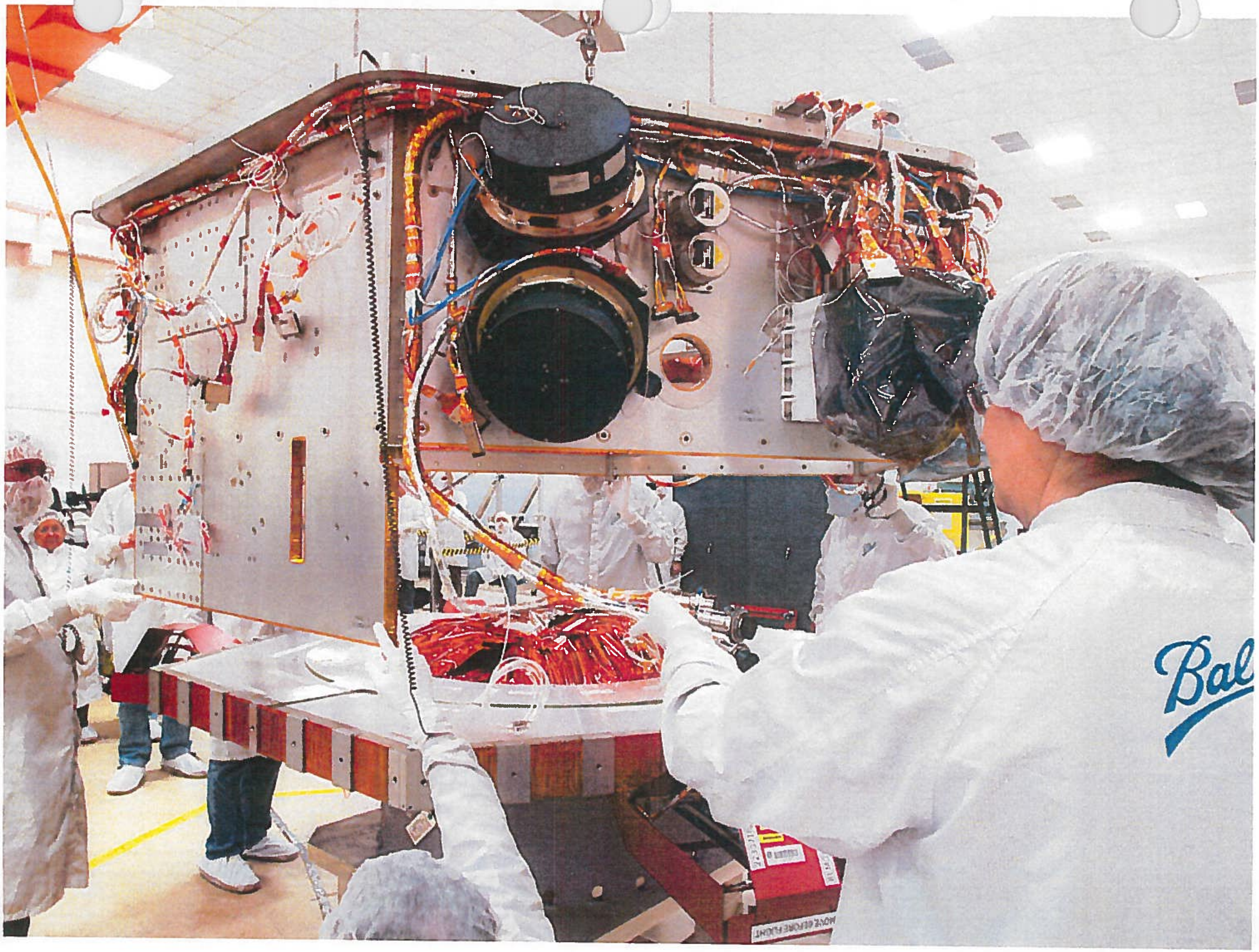
Configuration of energy/momentum wheels.




Sun
Shade
Photometer
CCD
Radiator
Star
Trackers
(2)
Reaction
Wheels (4)
Thruster
Modules (4)
High Gain
Antenna

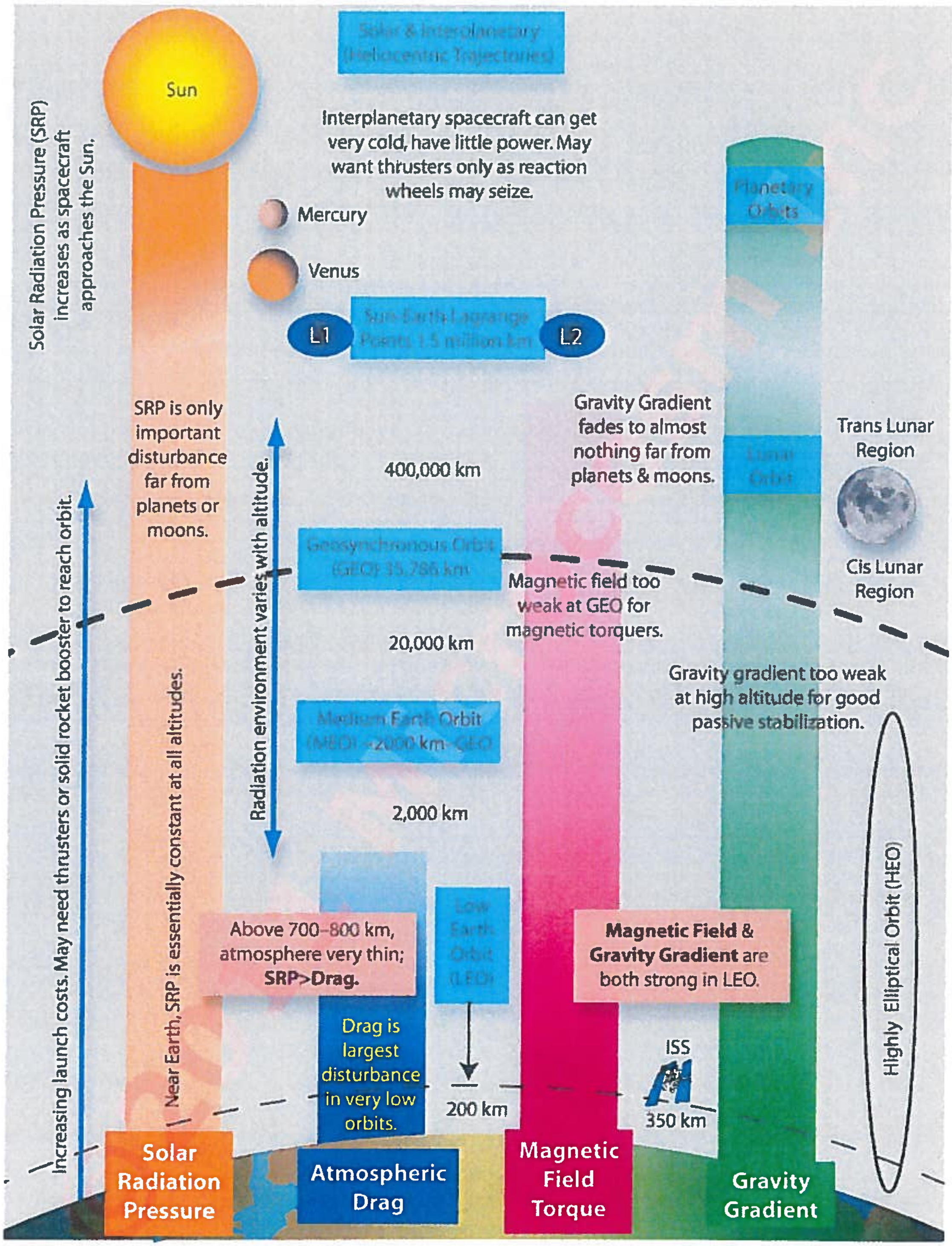

Kepler



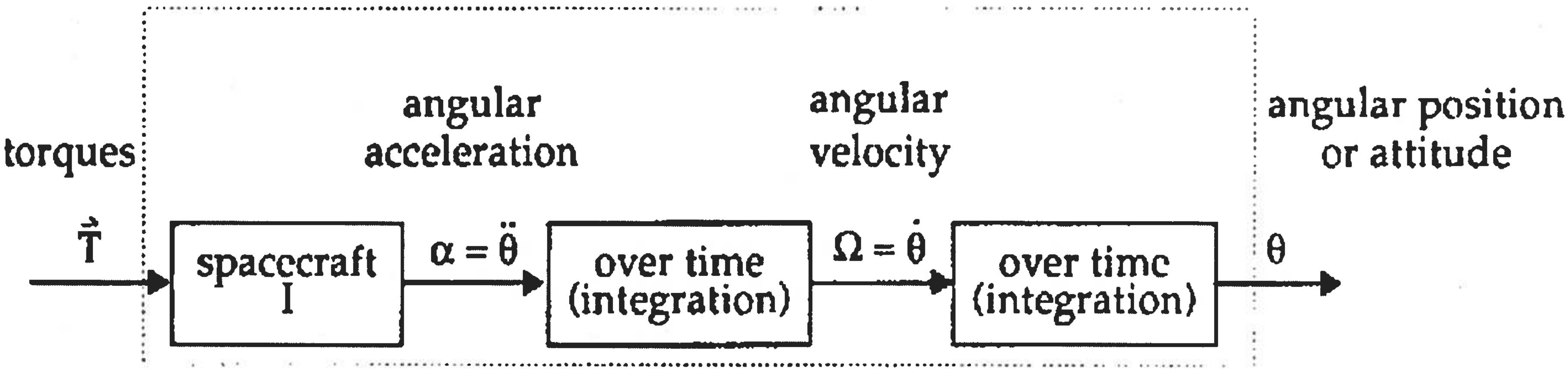


Text 19.1: Attitude Determination & Control

Environmental influences on spacecraft attitude:



Disturbing Torques



Attitude Dynamics for Spacecraft: "The Plant"

$$\vec{T} = \dot{\vec{H}} = I\vec{\alpha}$$

$$\vec{T} = \vec{r} \times \vec{F}$$

Disturbance Torques

Assessment of expected disturbance torques is an essential part of rigorous spacecraft attitude control design

Typical Disturbances

- Gravity Gradient: "Tidal" Force due to 1/r2 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.

Internal Disturbing Torques

- Examples
  - Uncertainty in S/C Center of Gravity (typically 1-3 cm)
  - Thruster Misalignment (typically 0.1° – 0.5°)
  - Thruster Mismatch (typically ~5%)
  - Rotating Machinery
  - Liquid Sloshing (e.g. propellant)
  - Flexible structures
  - Crew Movement

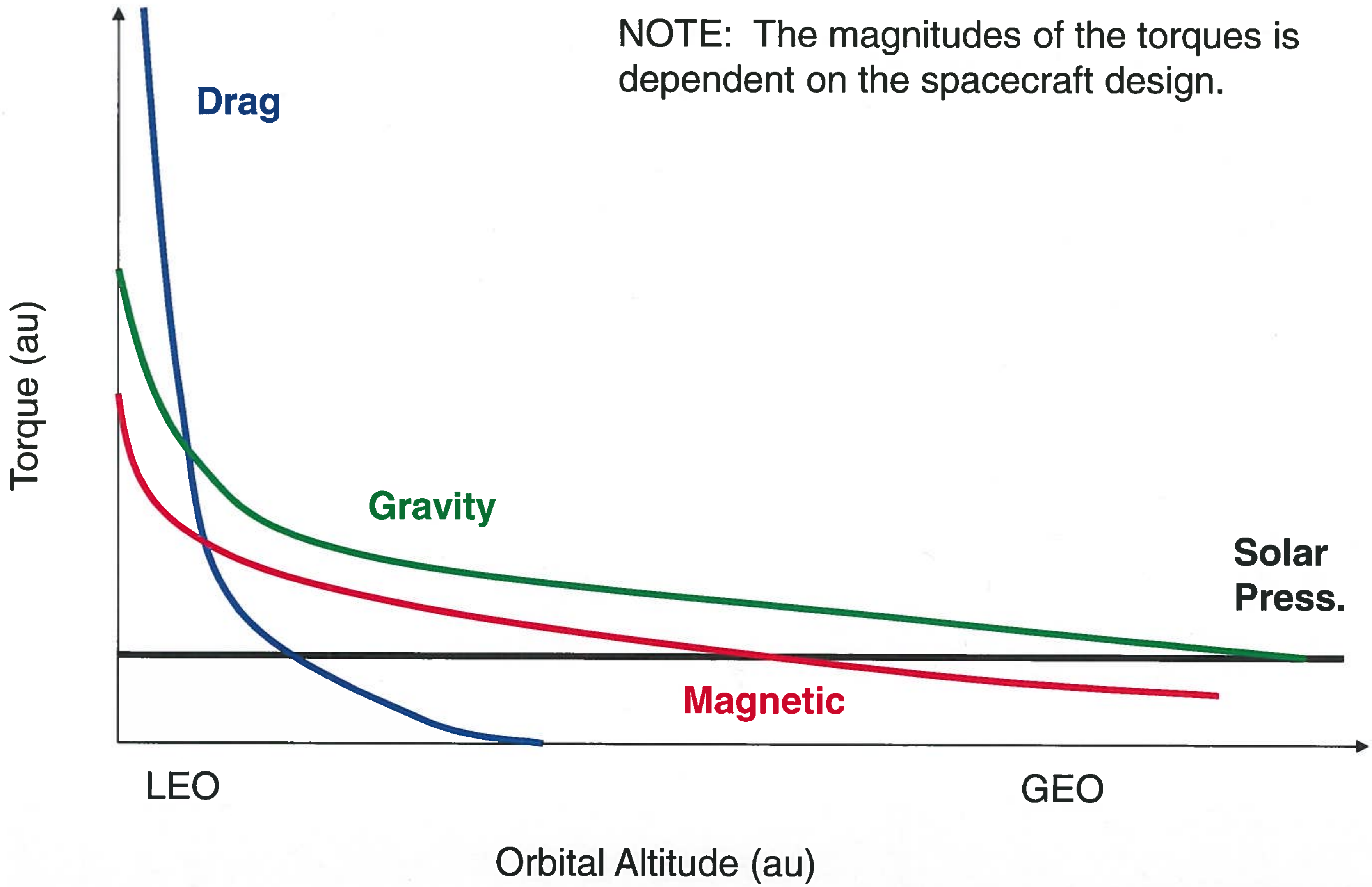
Text 1 9.1: Attitude Determination & Control

Spacecraft Self-Induced Attitude Disturbances:

Disturbances	Effect on Vehicle	Typical Values
Uncertainty in Center of Gravity (cg)	Unbalanced torques during firing of couples thrusters Unwanted torques during translation thrusting	1–3 cm
Thruster Misalignment	Same as cg uncertainty	0.1–0.5 deg
Mismatch of Thruster Outputs	Similar to cg uncertainty	±5%
Reaction Wheel Friction and Electromotive Force (i.e., back EMF)	Resistance that opposes control torque effort. These torques are the limiting mechanism for wheels speed.	Roughly proportional to wheel speed, depending on model. At top speed, 100% of control torque (i.e., saturation)
Rotating Machinery (pumps, filter wheels)	Torques that perturb both stability and accuracy	Dependent on spacecraft design; may be compensated by counter-rotating elements
Liquid Slosh	Torques due to liquid dynamic pressure on tank walls, as well as changes in cg location.	Dependent on specific design; may be mitigated by bladders or baffles
Dynamics of Flexible Bodies	Oscillatory resonance at bending/twisting frequencies, limiting control bandwidth	Depends on spacecraft structure; flexible frequencies within the control bandwidth must be phase-stabilized, which may be undesirable.
Thermal Shocks ("snap") on Flexible Appendages	Attitude disturbances when entering/leaving umbra	Depends on spacecraft structure. Long inertia booms and large solar arrays can cause large disturbances.

Table 19-5. Principal Internal Disturbance Torques. Spacecraft designers can minimize internal disturbances through careful planning and precise manufacturing, which may increase cost.

External Disturbance Torques



Gravity Gradient Torque

$$T_g = \frac{3\mu_\oplus}{2R^3}\left|I_z - I_y\right|\sin(2\theta)$$

where:

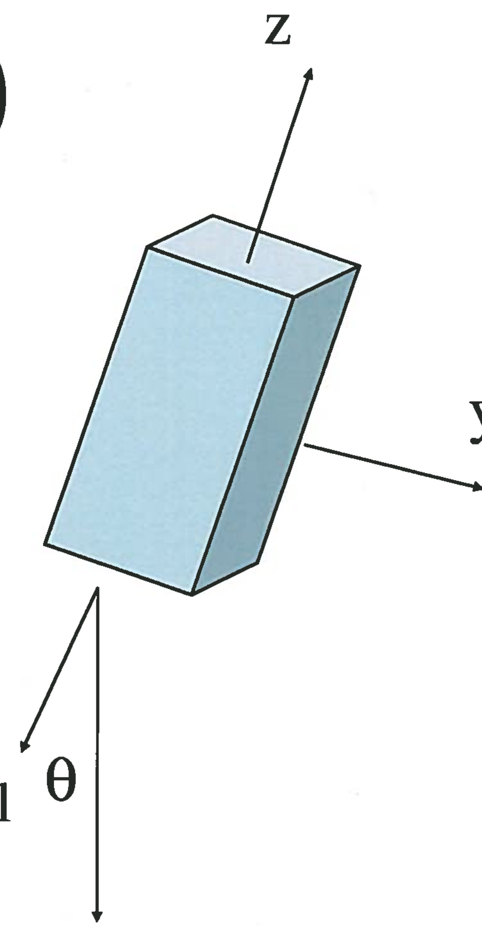
T_g = maximum gravity gradient

$\mu_\oplus$ = Earth's gravitational parameter

R = orbit radius

I_y, I_z = S/C mass moments of inertia

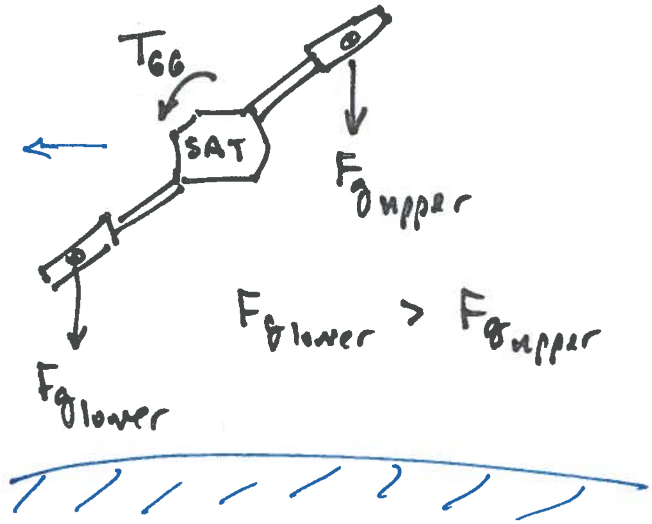
θ = maximum deviation away from vertical



Text 19.1: Attitude Determination & Control

Gravity Gradient

Force of gravity pulls very slightly harder on segment of satellite closer to Earth



- $$\frac{\text{gravitational force}}{\text{unit mass}} = \frac{F_g}{m} = g = \frac{\mu}{r^2}$$

- Radial gradient of gravitational force:

$$\frac{dg}{dr} = -\frac{2\mu}{r^3} \Rightarrow$$ gravitational attraction decreases with altitude (very small changes!)

alt (km)	g (m/s²)	dg/dr ($\frac{m/s^2}{m}$)
322	8.879	-2.651 E-6
552	8.300	-2.395 E-6
3586	0.223	-0.011 E-6

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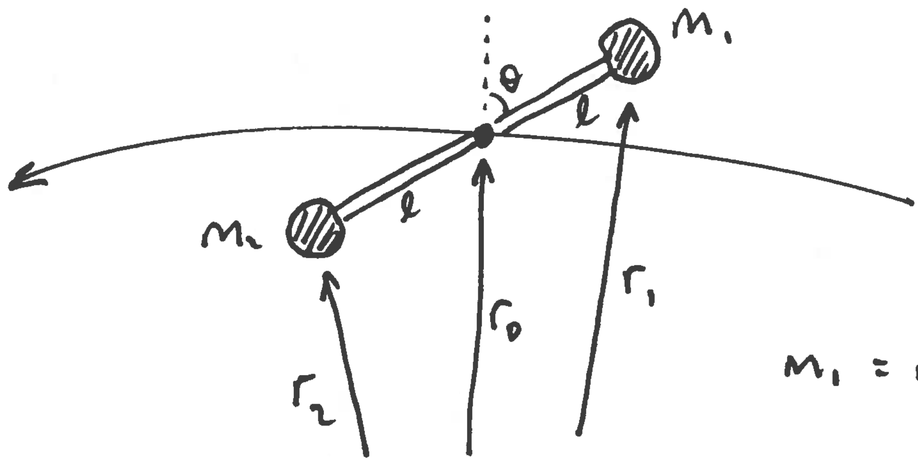
- Consider dumb-bell of point-masses in circular Earth orbit — estimate moment induced by gravity gradient :



$m_1 = m_2$

- Forces on m_1 and m_2 :

$$\Sigma F_1 = \underbrace{\frac{m_1 v_1^2}{r_1}}_{\text{centrifugal}} - \underbrace{m_1 g_1}_{\text{gravitational}}$$

$$= m_1 \left[\frac{v_1^2}{r_0 + l\cos\theta} - \frac{\mu}{(r_0 + l\cos\theta)^2} \right]$$

$\Sigma F_2 = \ldots$ (similar)

Text 19.1: Attitude Determination & Control

- Now assume θ = small (nearly vertical) $\Rightarrow \cos\theta \cong 1$

- Moment about center of mass :

$$M_{cm} = F_2 \ell \theta - F_1 \ell \theta = \ell\theta (F_2 - F_1)$$

- Furthermore, assume $V_1 \cong V_2 \cong V_0$

$$\ell \ll r_0$$

$$m_1 = m_2 = m$$

algebra

$$M_{cm} \cong -2m\left(\frac{\ell}{r_0}\right)^2 \underbrace{\left(\frac{\mu}{r_0}\right)}_{V^2} \theta$$

Text 19.1: Attitude Determination & Control

- So, moment about CM of dumb-bell satellite due to gravity gradient:

$$M_{cm} \cong -2m\left(\frac{l}{r_0}\right)^2\left(\frac{\mu}{r_0}\right)\theta$$

- Observations:

  - For positive θ, M_{cm} is negative $\rightarrow$ restoring torque (statically stable)

  - Gravity gradient torque is small, and decreases with increasing altitude – only useful in LEO

  - Restoring torque $\propto$ mass of satellite
    & square of length
    $\propto \theta$, so difficult to get large θ

  - Dynamic stability:

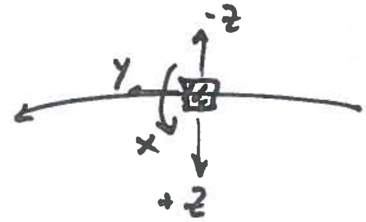
$$\omega_n = \sqrt{\frac{\mu}{r_0^3}}$$

undamped natural freq. independent of m and l

Text 19.1: Attitude Determination & Control

- SMAD Gravity Gradient:

  Torque always tries to align minimum principal axis with local gravity vector



- Spacecraft with min principal axis in its z-direction:

  axis about which I is min

$$T_g = \frac{3\mu}{2R} \left| I_z - I_y \right| \sin(2\theta)$$

where T_g = torque due to GG about X principal axis

μ = Earth gravitational const $\left(3.986E14 \ \frac{m^3}{s^2}\right)$

R = distance to Earth center (m)

θ = angle between "down" and z

I = moments of inertia ($kg\ m^2$)

Aerodynamic Torque

$$\underline{T} = \underline{r} \times \underline{F}_a$$

r = Vector from body CM to Aerodynamic CP

$$F_a = \frac{1}{2}\rho V^2 S C_D$$

Fa = Aerodynamic Drag Vector in Body coordinates

Aerodynamic Drag Coefficient

$$1 \le C_D \le 2$$

Typically in this Range for Free Molecular Flow

S = Frontal projected Area

V = Orbital Velocity

ρ = Atmospheric Density

Typical Values:

Cd = 2.0

S = 5 m^2

r = 0.1 m

r = 4 x 10^{-12} kg/m^3

T = 1.2 x 10^{-4} Nm

Notes

(1) r varies with Attitude

(2) ρ varies by factor of 5-10 at a given altitude

(3) C_D is uncertain by 50 %

2 x 10^{-9} kg/m^3 (150 km)

3 x 10^{-10} kg/m^3 (200 km)

7 x 10^{-11} kg/m^3 (250 km)

4 x 10^{-12} kg/m^3 (400 km)

Exponential Density Model

Aerodynamic Torque

$$T_a = F\left(c_{pa} - c_g\right)$$

where: $$F \cong \frac{1}{2}\rho C_D A v^2$$

T_a = aerodynamic disturbance torque

ρ = atmospheric density

C_D = coefficient of drag (typical S/C values are 2 - 2.5)

A = cross - sectional area

v = velocity

C_{pa} = center of atmospheric pressure

C_g = center of gravity

Text 19.1: Attitude Determination & Control

- Torque on spacecraft due to Atmospheric Drag

$$T_a = \underbrace{\tfrac{1}{2}\rho C_d A_r V^2}_{\text{force}} \underbrace{(C_{P_a} - CM)}_{\text{distance}}$$

where T_a = atmos drag torque

ρ = atmos density

C_d = drag coeff (2.0 - 2.5 for spacecraft)

A_r = "ram" area

↑ projection onto plane $\perp$ to velocity vector

V = spacecraft velocity relative to atmos.

misalignment $\perp$ to direction of force produces torque

CP_a = center of aerodynamic pressure

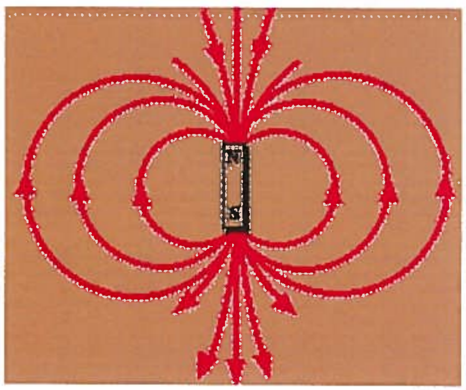
CM = center of mass

Magnetic Torque

$$\underline{T} = \underline{M} \times \underline{B}$$

M = Spacecraft residual dipole in AMPERE-TURN-m2 (SI) or POLE-CM (CGS)

M = is due to current loops and residual magnetization, and will be on the order of 100 POLE-CM or more for small spacecraft.



Typical Values:

B= 3 x 10^{-5} TESLA

M = 0.1 Atm^2

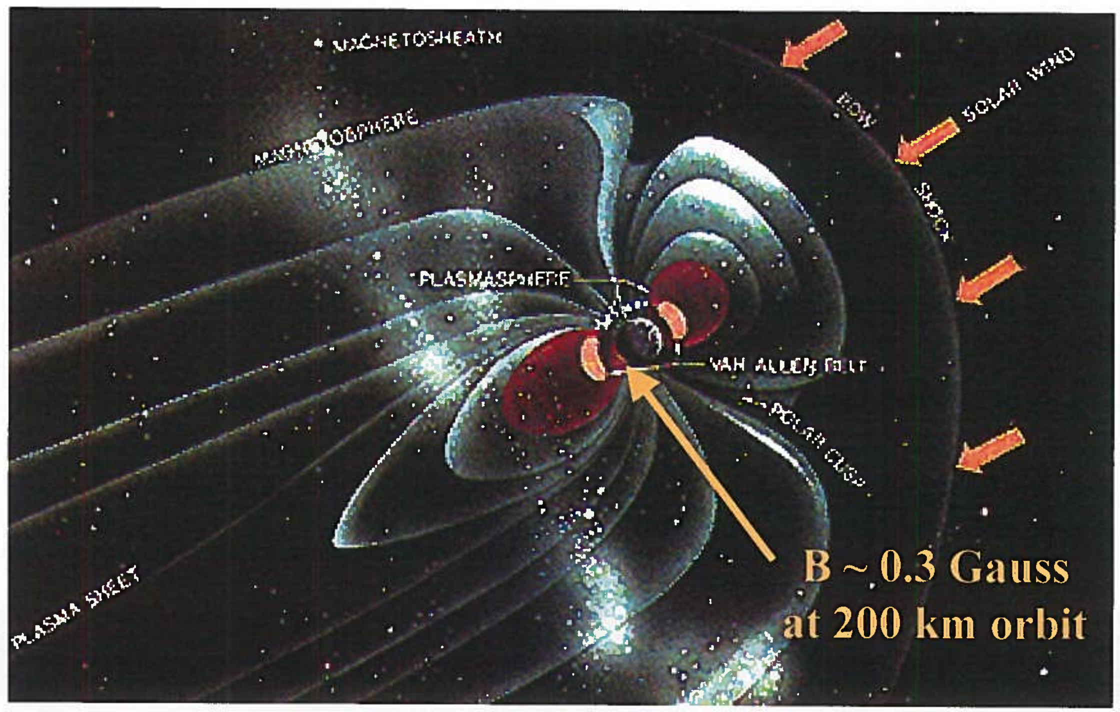
T = 3 x 10^{-6} Nm

B = Earth magnetic field vector in spacecraft coordinates (BODY FRAME) in TESLA (SI) or Gauss (CGS) units.

B varies as 1/r3, with its direction along local magnetic field lines.

Conversions:

1 Atm2 = 1000 POLE-CM , 1 TESLA = 104 Gauss



Magnetic Torque

$$\overline{T}_m = \overline{m} x \overline{B}$$

where:

$$T_m = \text{magnetic disturbance torque}$$

$$m = \text{S/C residual magnetic dipole} \left[\text{Amp} \cdot \text{m}^2\right]$$

$$B = \text{strength of Earth's magnetic field}$$

$$= \frac{M}{R^3} \text{ for points above the equator}$$

$$= \frac{2M}{R^3} \text{ for points above the poles}$$

$$M = \text{Earth's magnetic moment} \left(7.96 \times 10^{15} \text{ tesla} \cdot \text{m}^3\right)$$

$$R = \text{orbit radius} \left[\text{meters}\right]$$

*Note value of *m* depends on S/C size and whether on-board compensation is used

- values can range from 0.1 to 20 Amp-m^2
- $m = 1$ for typical small, uncompensated S/C

Text 19.1: Attitude Determination & Control

- Torque on spacecraft due to Magnetic Field:

Most spacecraft have their own magnetic field, the "residual magnetic moment," which can interact with Earth's magnetic field.
(if mag fields not aligned → magnetic torque)

- Model both Earth and spacecraft magnetic fields as dipoles

- $T_m = D B = D\left(\frac{M}{R^3}\lambda\right)$

where T_m = max magnetic torque

D = spacecraft resid. dipole moment

B = magnetic field strength of Earth at satellite location

M = magnetic moment of Earth

R = distance to Earth's center

λ = function of magnetic latitude
= 1 at mag. equator, = 2 at mag. poles

Solar Radiation Torque

$$\underline{T} = \underline{r} \times \underline{F}_s$$

$$F_s = (1+K) P_s S$$

$$P_s = I_s / c$$

$$I_s = 1400 \text{ W/m}^2 \quad @ \ 1 \ \text{A.U.}$$

Notes:

(a) Torque is always $\perp$ to sun line
(b) Independent of position or velocity as long as in sunlight

Typical Values:
K = 0.5
S =5 m^2
r =0.1 m
T = 3.5 x 10^{-6} Nm

$\underline{r}$ = Vector from Body CM to optical Center-of-Pressure (CP)

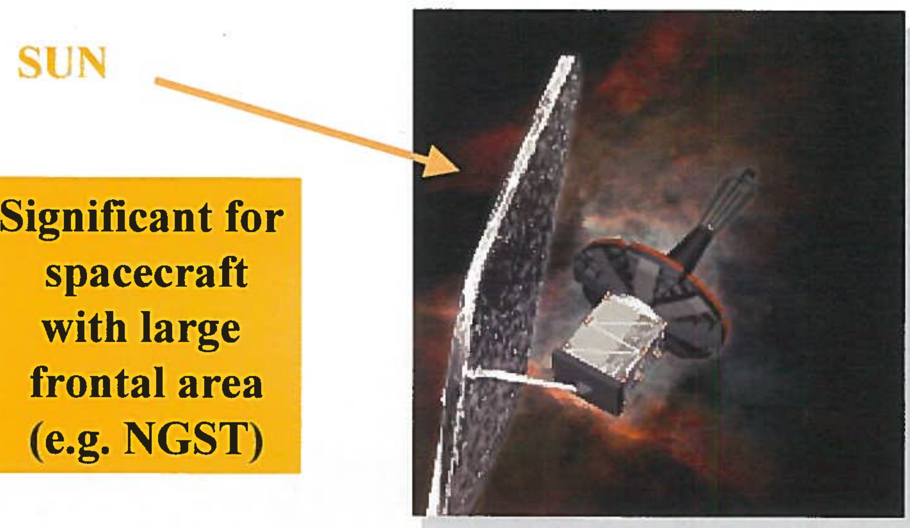
$\underline{F}_s$ = Solar Radiation pressure in BODY FRAME coordinates

K = Reflectivity , 0 < K <1

S = Frontal Area

I_s = Solar constant, depends on heliocentric altitude

Significant for spacecraft with large frontal area (e.g. NGST)



Text 19.1: Attitude Determination & Control

- Torque on spacecraft due to Solar Radiation Pressure (SRP):

$$T_s = \underbrace{\frac{\Phi}{c} A_s (1+q)}_{\text{force}} \underbrace{(c_{p_s} - CM)\cos\varphi}_{\text{distance}}$$

where T_s = SRP torque

Φ = solar const ($1366 \frac{W}{m^2}$ at 1AU)

c = speed of light (3E8 m/s)

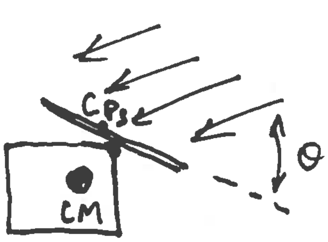
A_s = sunlit surface area (m^2)

q = reflectance factor = 0 for absorption, 1 for reflection

ϕ = angle of incidence of sunlight

c_{p_s} = center of SRP

CM = center of mass



Solar Pressure Torque

$$T_{srp} = F\left(c_{ps} - c_g\right)$$

where:

$$F = \frac{F_s}{c} A_s (1+\rho)\cos i$$

T_{srp} = solar radiation presure disturbance torque

c_{ps} = center of solar radiation pressure

$\mathrm{c_g}$ = center of gravity

F_s = solar flux density $\left[\frac{\mathrm{W}}{\mathrm{m}^2}\right]$

c = speed of light

A_s = area of illuminated surface

ρ = reflectance factor $(0 < \rho < 1$, typical value 0.6 for S/C$)$

i = sun incidence angle

Mass Expulsion and Internal Torques

Mass Expulsion Torque: $\underline{T} = \underline{r} \times \underline{F}$

Notes:

(1) May be deliberate (Jets, Gas venting) or accidental (Leaks)

(2) Wide Range of r, F possible; torques can dominate others

(3) Also due to jettisoning of parts (covers, cannisters)

Internal Torque:

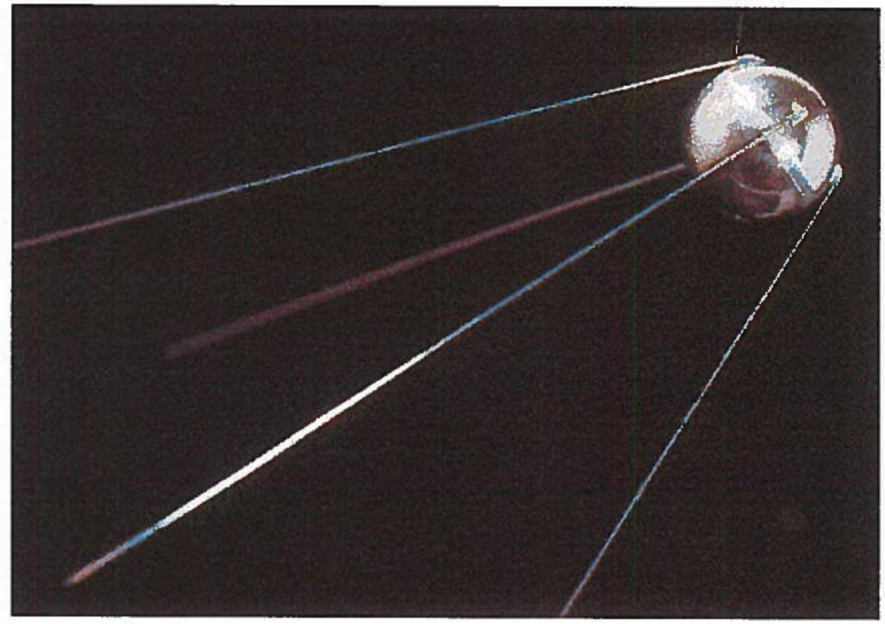
Notes:

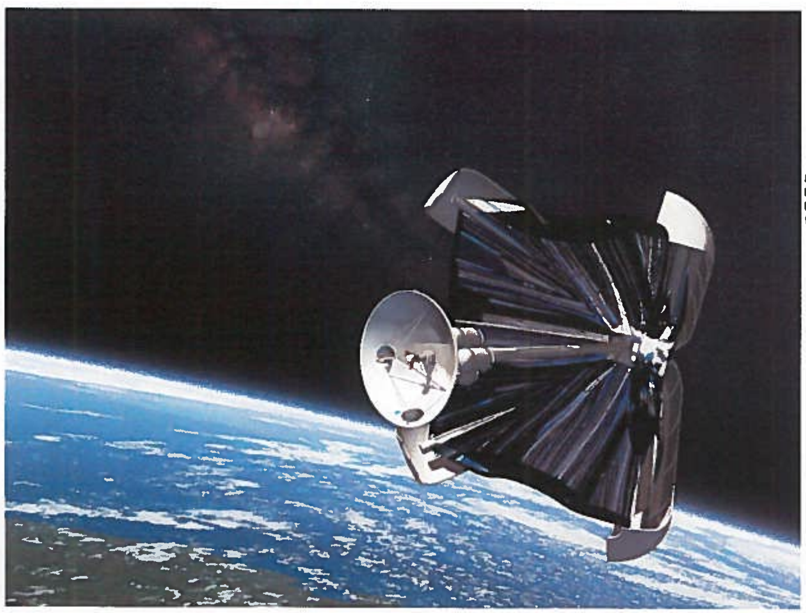
(1) Momentum exchange between moving parts has no effect on System H, but will affect attitude control loops

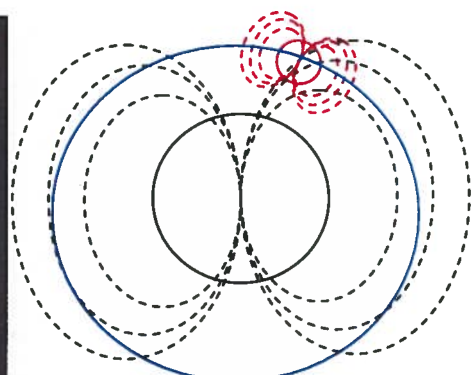
(2) Typically due to antenna, solar array, scanner motion or to deployable booms and appendages

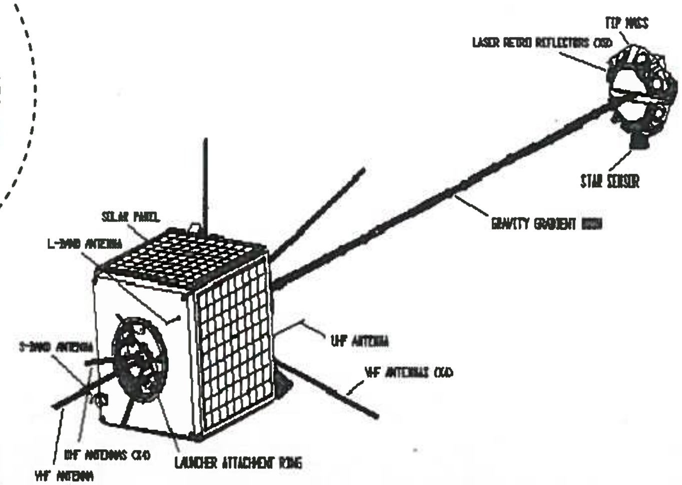
Disturbing Torques

- All of these disturbing torques can also be used to control the satellite
  - Gravity Gradient Boom
  - Aero-fins
  - Magnetic Torque Rods
  - Solar Sails



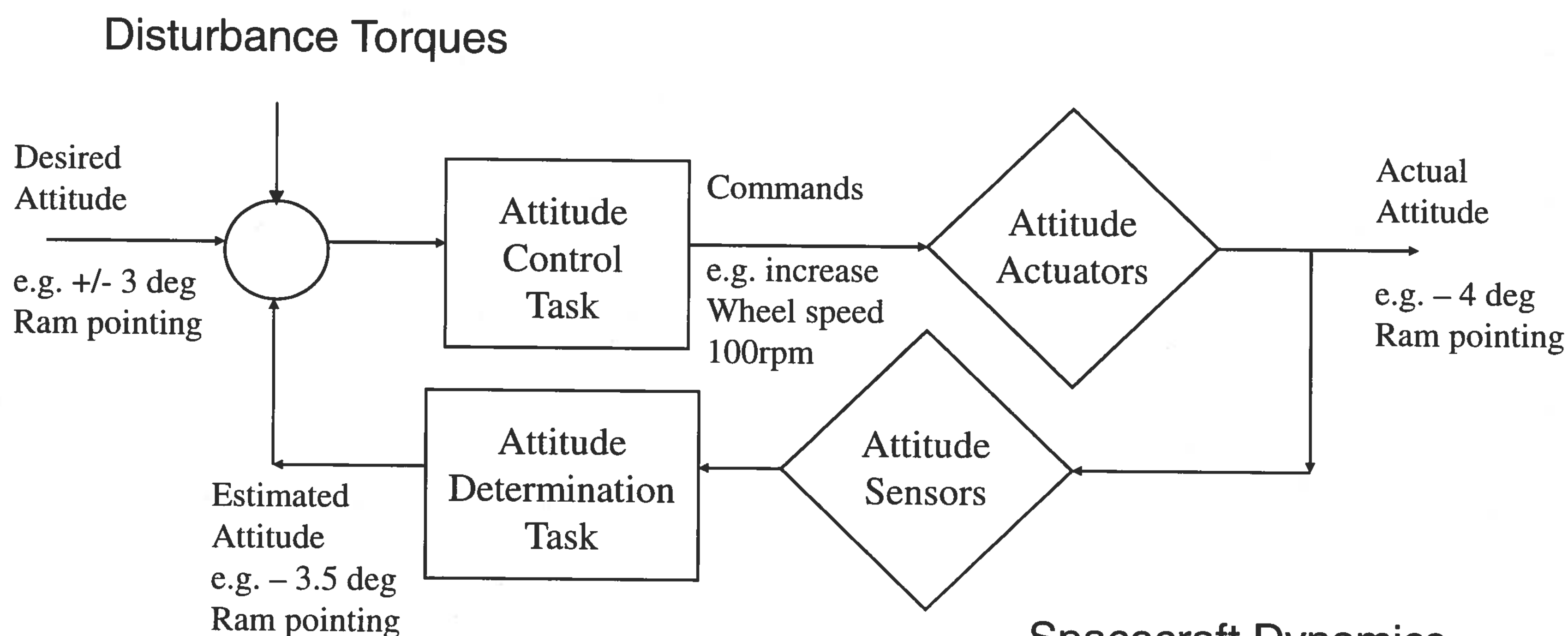






Sensors

Control Loops



Spacecraft Dynamics

- Rigid Body
- Flexible Body (non-rigid)

Attitude Measurement

- Attitude reference: inertial
  spacecraft
  Earth
  etc

- 2 sensor categories: need both in most cases

  Reference Sensor: absolute attitude measurement

  Inertial Sensor: changes in attitude, relative to an instantaneous attitude snapshot, or "fix"

- Inertial sensors drift, so are periodically (often) calibrated by Reference sensor

- $\underbrace{\text{Multi-sensors}}$ → Kalman Filter → inst. attitude

  Reference sensors
  Inertial sensors
  Multiple types
  Redundant sensors

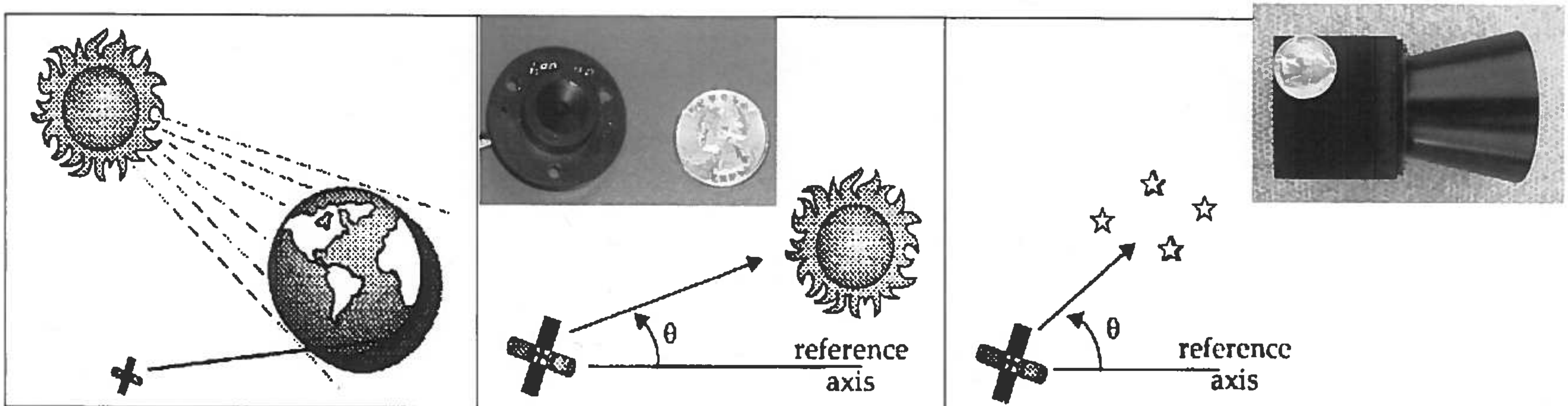
Attitude Determination

- Earth Sensor (horizon sensor)
  - Use IR to detect boundary between deep space & upper atmosphere
  - Typically scanning (can also be an actuator)
- Sun Sensor
- Star Sensor
  - Scanner: for spinning S/C or on a rotating mount
  - Tracker/Mapper: for 3-axis stabilized S/C
    - Tracker (one star) / Mapper (multiple stars)
- Inertial Measurement Unit (IMU)
  - Rate Gyros (may also include accelerometers)
- Magnetometer
  - Requires magnetic field model stored in computer
- Differential GPS

Typical Spacecraft Sensor Configurations

- Most precise measurements (e.g., scientific satellites)
  - star trackers
- Moderate accuracy requirements
  - coarse digital sun sensors
  - horizon sensors
  - magnetometers
- Spinning satellites
  - single-axis sun sensors
  - magnetometers
  - horizon sensors
- High-altitude (e.g., geosynchronous) satellites
  - optical sensors
  - gyroscopes
  - magnetic field too weak for use

Attitude Determination



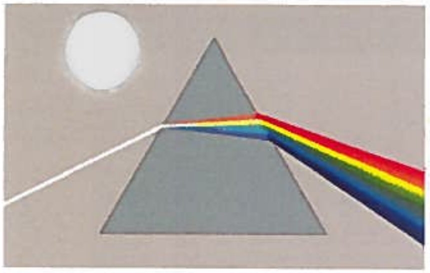
Earth Horizon Sensor | Sun Sensor | Star Tracker

Sensor	Accuracies	Comments
IMU	Drift: 0.0003 – 1 deg/hr 0.001 deg/hr nominal	Requires updates
Star Sensor	1 arcsec – 1 arcmin (0.0003 – 0.001 deg)	2-axis for single star Multiple stars for map
Sun Sensor	0.005 – 3 deg 0.01 deg nominal	Eclipse
Earth Sensor GEO LEO	< 0.1 – 0.25 deg 0.1 – 1 deg	2-axis
Magnetometer	0.5 – 3 deg	< 6000 km Difficult for high *i*

Text 19.1: Attitude Determination & Control

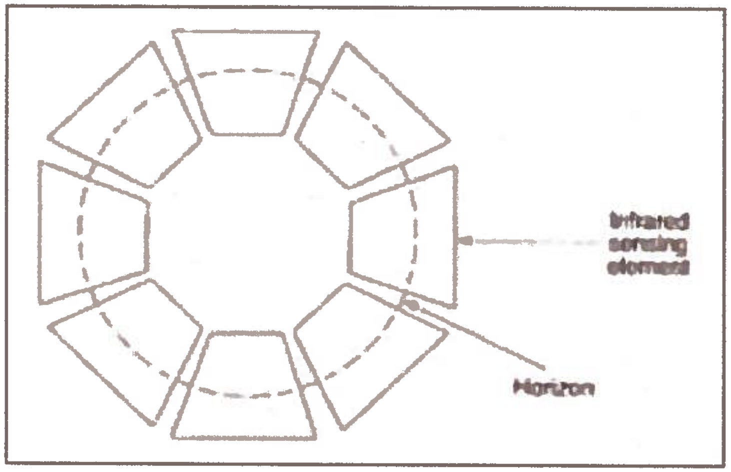
Attitude Sensors:

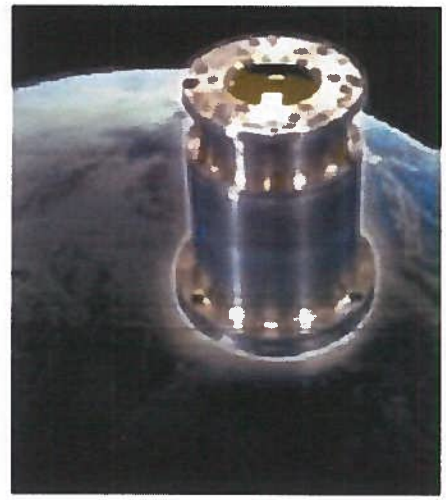
Sensor	Typical Performance Range	Mass (kg)	Power (W)
Gyroscopes	Drift Rate = 0.003 deg/hr to 1 deg/hr Drift rate stability varies widely	< 0.1 to 15	< 1 to 200
Sun Sensors	Accuracy = 0.005 deg to 3 deg	0.1 to 2	0 to 3
Star Sensors *(Scanners & Cameras)*	Accuracy = 1 arcsecond to 1 arcminute = 0.0003 deg to 0.01 deg	2 to 5	5 to 20
Horizon Sensors *Scanner/Pipper* *Fixed Head (Static)*	Accuracy: 0.05 deg to 1 deg (0.1 deg is best for LEO) < 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



Earth Horizon Sensor

- Infrared sensing to reduce optical error
- Static horizon sensor has field of view larger than the entire earth's edge (limb)
- Provides orientation with respect to the nadir





Goodrich Multi-Mission Horizon Sensor

Characteristics

Infrared spectral band: 14 to 16 μm

Detectors: Micro-machined (MEMS) thermopiles

Total field-of-view: 16° long by 10° wide

Spacecraft keep-out-zone: 15° cone around aperture

GEO Performance (redundant, 2-telescope assembly)

Pitch accuracy (E-W): +/- 0.013°, 3-σ for either telescope

Roll accuracy (N-S): +/- 0.030°, 3-σ for either telescope

(Above is RSS accuracy at null, based on 30°C long-term thermal drift and 18° peak-to-peak diurnal)

Operating range: +/- 6° fine pitch measurement

+/- 2° fine roll measurement

Acquisition range: 16° maximum Earth detection

Earth Horizon Sensor

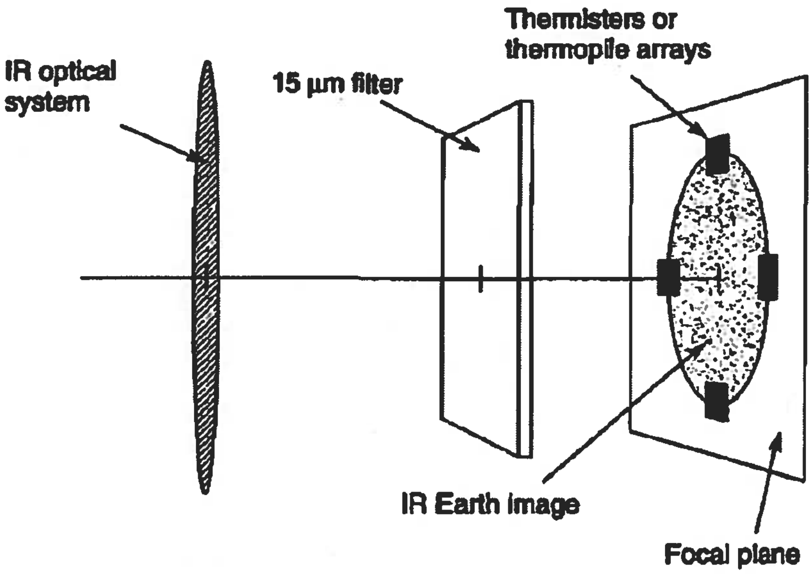


Figure 9.13 Schematic of a static Earth-horizon sensor

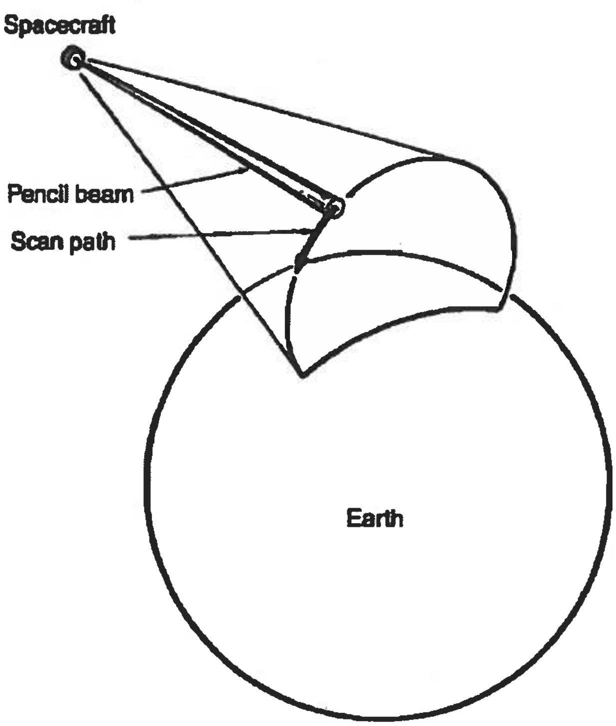


Figure 9.14 Scanning technique for an Earth horizon sensor

Scanning Earth Horizon Sensor

- Spinning assembly identifies light and dark areas (infrared)
- Width of light area identifies spacecraft roll angle, ϕ

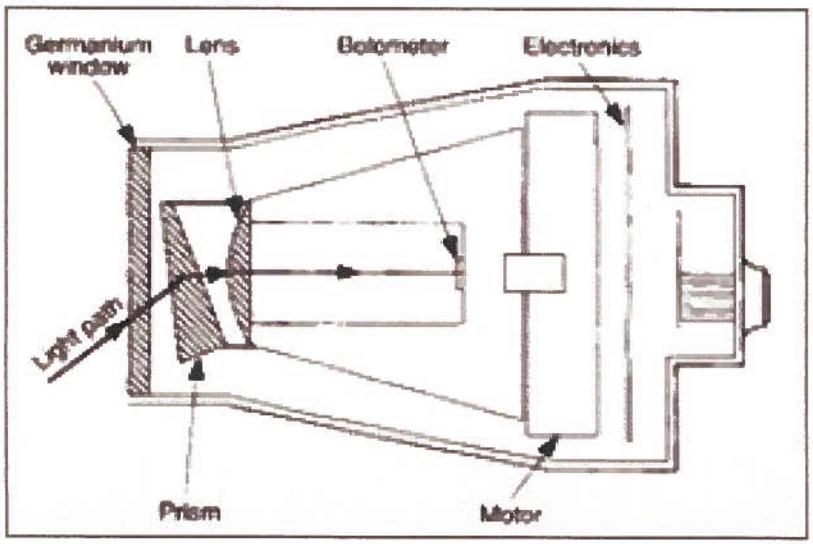
$$\cos\rho = \cos\gamma\cos\phi + \sin\gamma\sin\phi\cos(\Omega/2)$$

ρ: *Earth angular radius*

γ: *Half-cone angle*

$$\Omega = \omega_{scanner}\left(t_{LOS} - t_{AOS}\right)$$

$t_{LOS/AOS}$: *Time of loss/acquisition of signal*



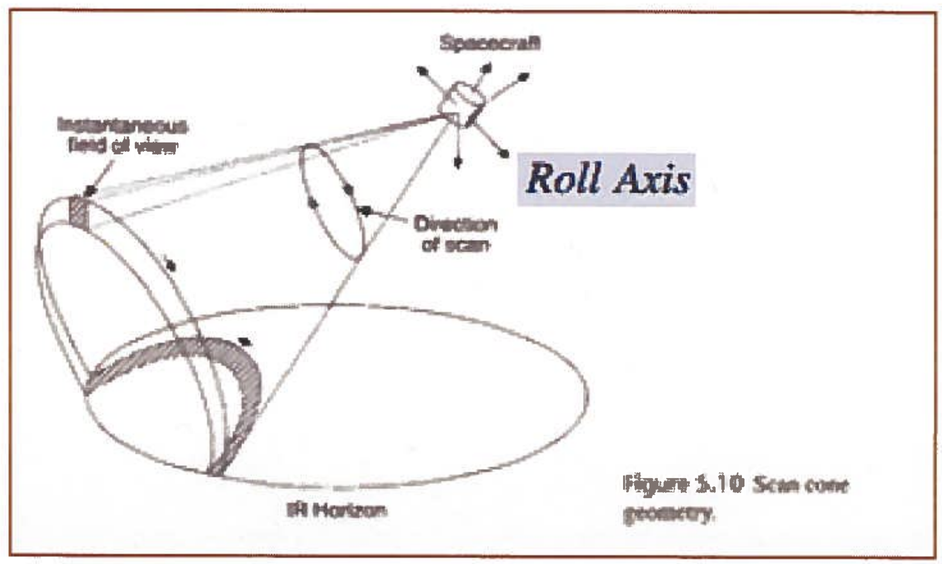
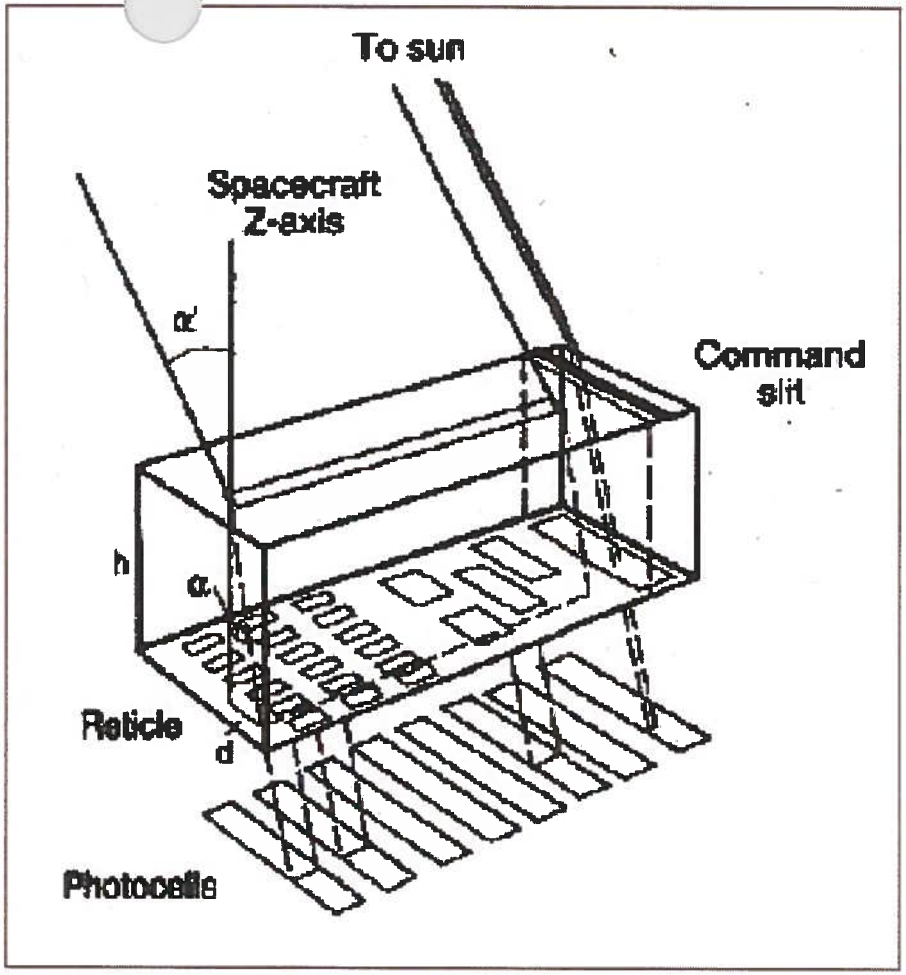
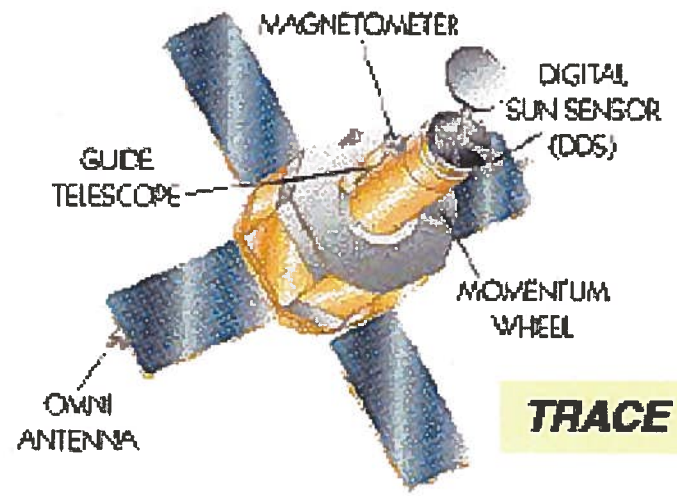


Figure 5.10 Scan cone geometry.



Sun Sensor



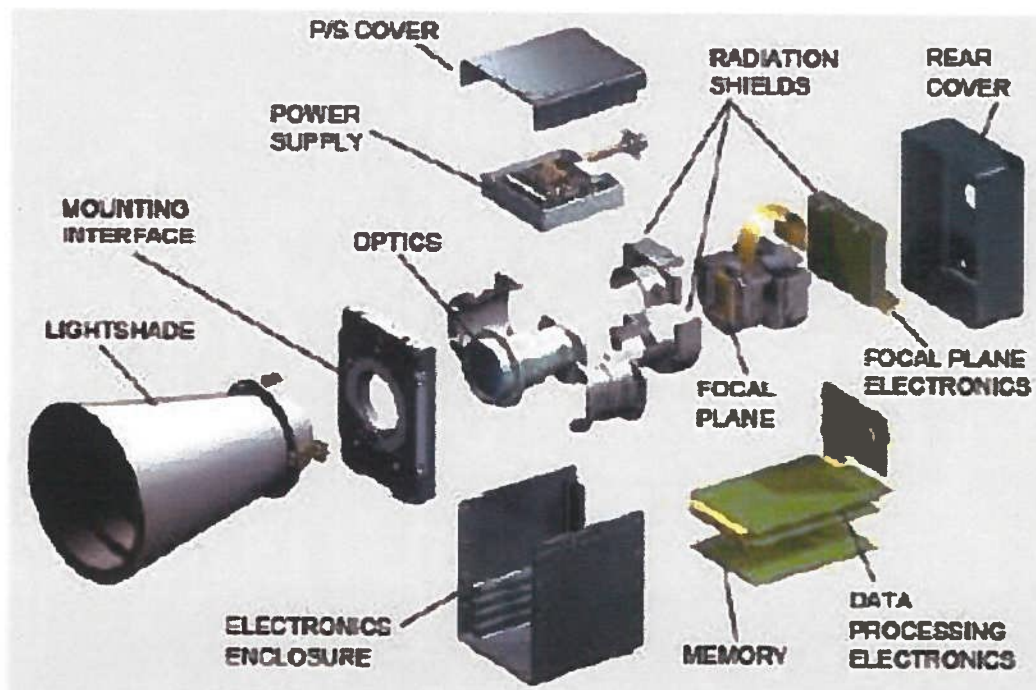
- Transparent block of material with known refractive index, *n*, coated with opaque material
- Slit etched in top, receptive areas etched in bottom
- Light from sun passing through slit forms a line over photodetectors

Star Sensor/Tracker

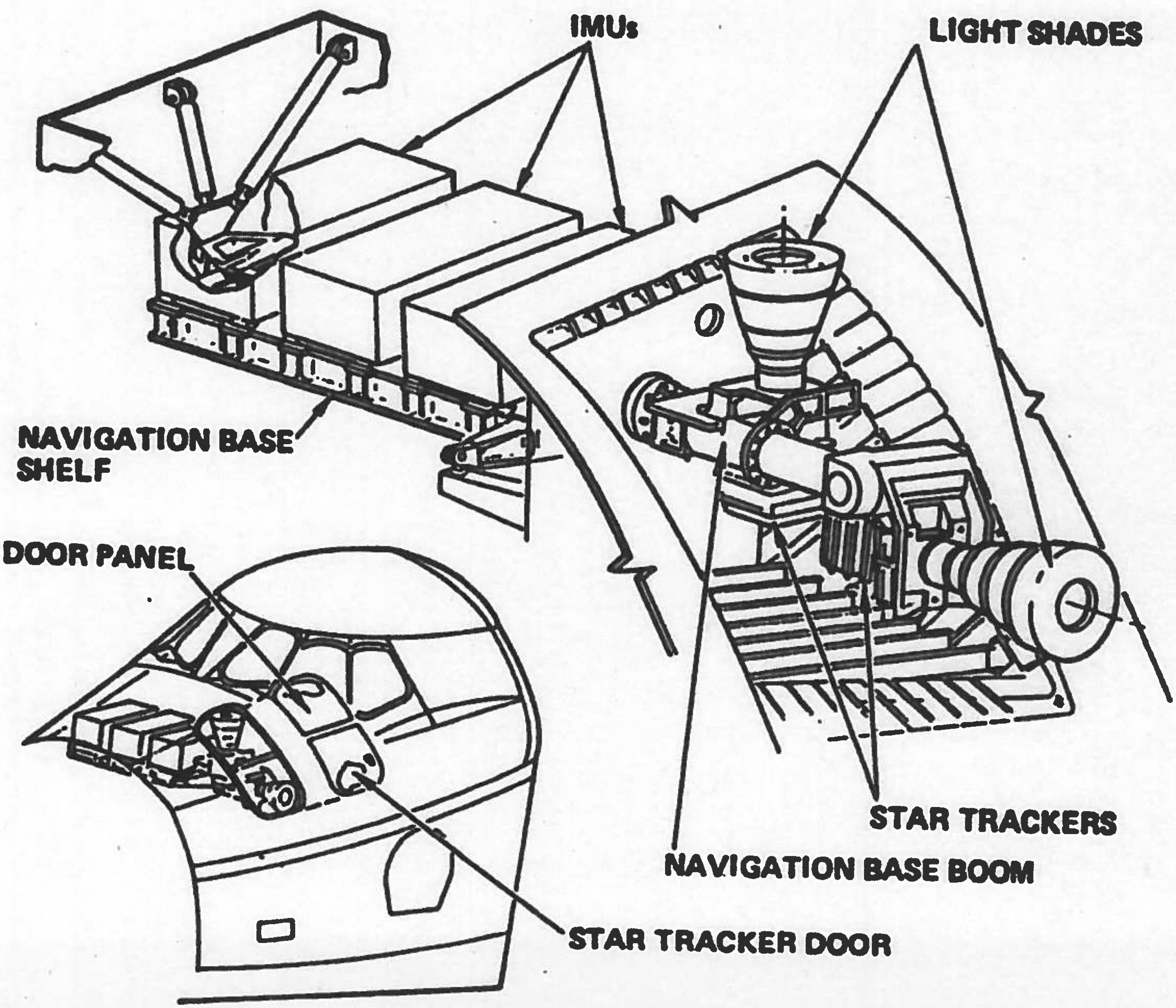
- Instrument has narrow field of view
- Star location catalog helps identify target
- Instrument must have low angular velocity
- *x* and *y* location of star on focal plane determines angles to the star

Goodrich Star Tracker

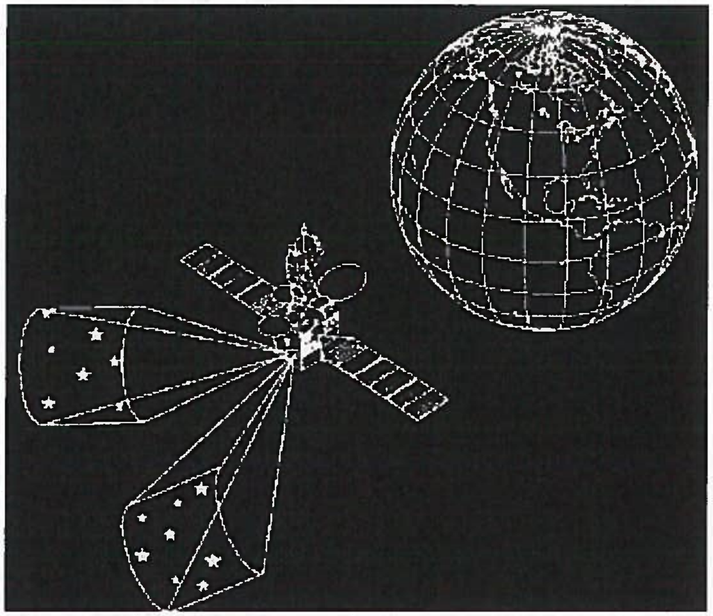
Performance Category	*Narrow FOV*	*Wide FOV*
Field of View Magnitude Sensitivity Power (avg. at +45°C)	8° x 8° +6.5 10W	20° circular +5.1 10W
Weight (with lightshade) Update Rate Stars Simultaneously Tracked	8.5 lb 6 Hz 6	7.5 lb 2 Hz 6
Overall Accuracy -Pitch/Yaw, rms -Roll, rms	 2 arc sec 40 arc sec	 5 arc sec 40 arc sec

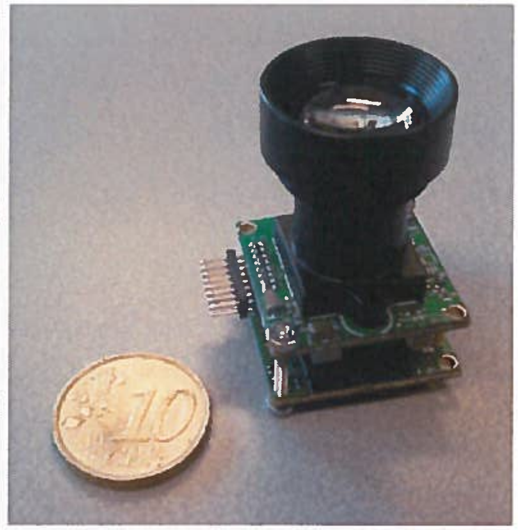


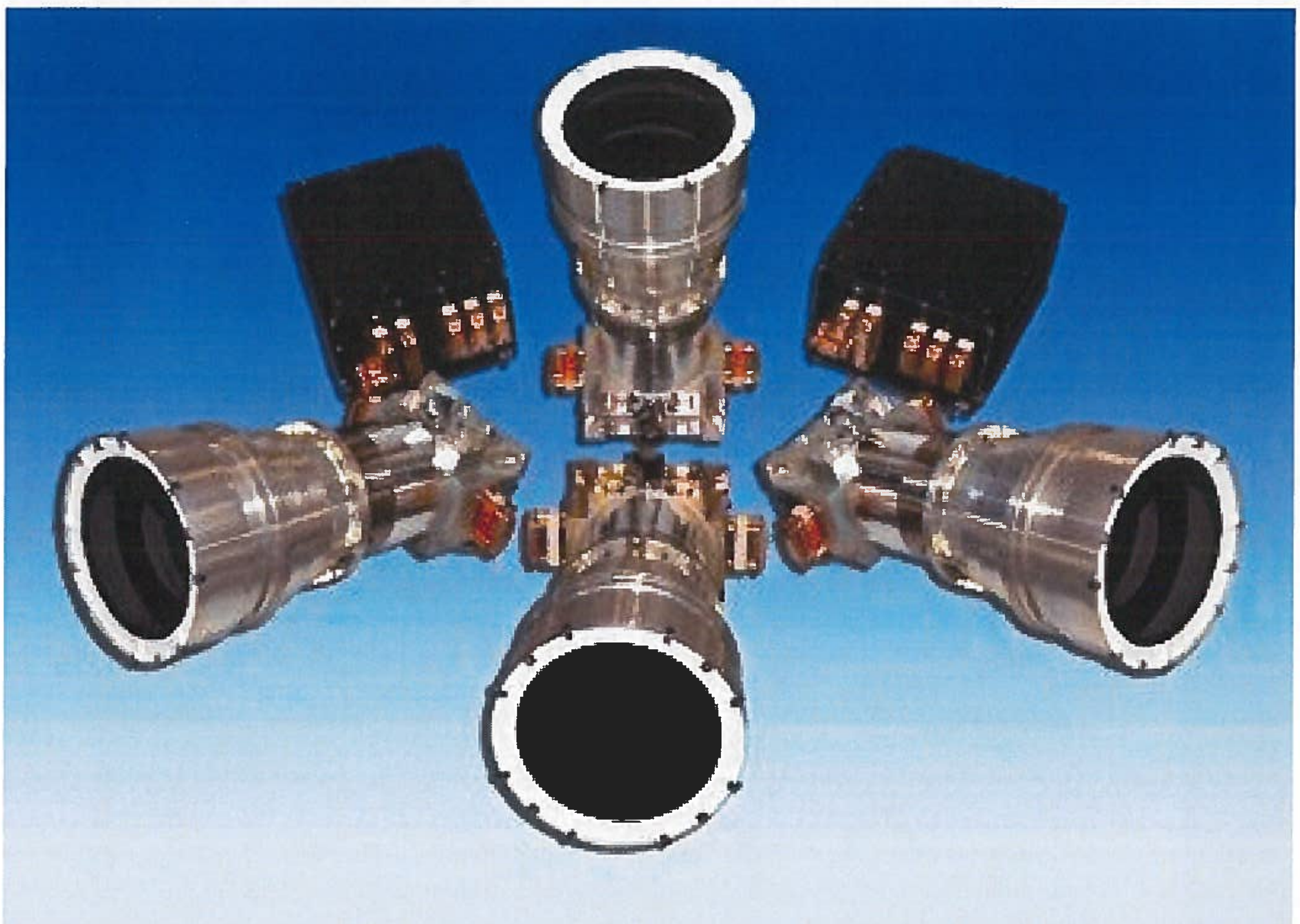
Shuttle Star Trackers



Star Sensors

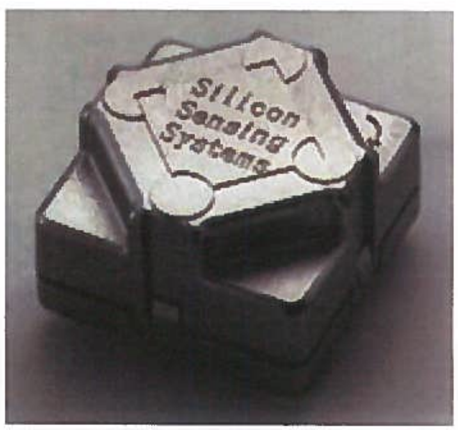






ACS Sensors: Rate Gyros and IMUs

- Rate Gyros (Gyroscopes)
  - Measure the angular rate of a spacecraft relative to inertial space
  - Need at least three. Usually use more for redundancy.
  - Can integrate to get angle. However,
    - DC bias errors in electronics will cause the output of the integrator to ramp and eventually saturate (drift)
    - Thus, need inertial update

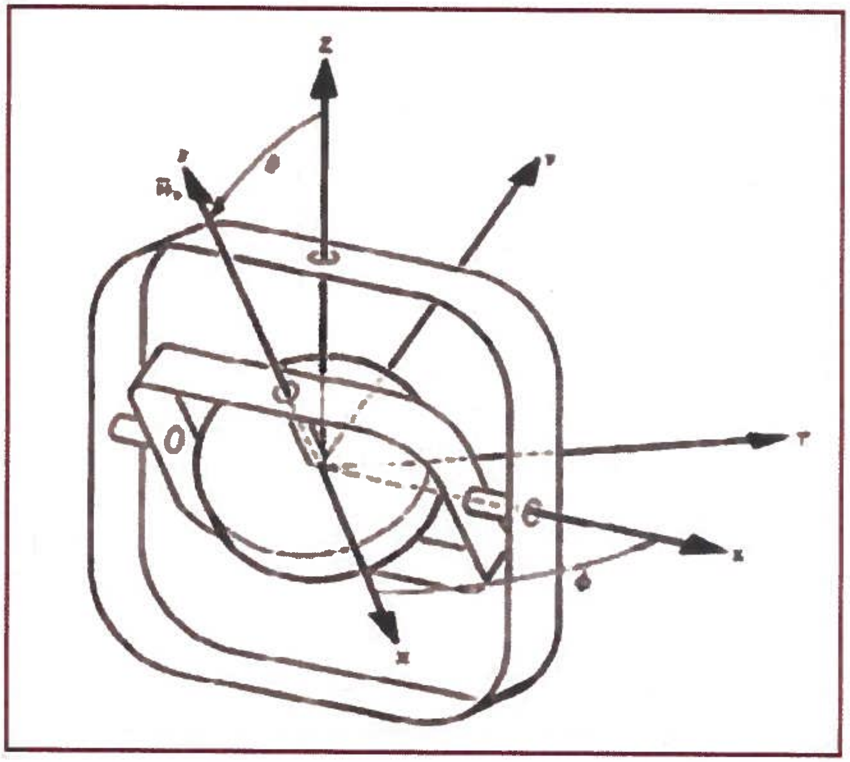


Courtesy of Silicon Sensing Systems, Ltd. Used with permission.

- Mechanical gyros (accurate, heavy)
- Ring Laser (RLG)
- MEMS-gyros

- Inertial Measurement Unit (IMU)
  - Integrated unit with sensors, mounting hardware,electronics and software
  - measure rotation of spacecraft with rate gyros
  - measure translation of spacecraft with accelerometers
  - often mounted on gimbaled platform (fixed in inertial space)
  - Performance 1: gyro drift rate (range: 0 .003 deg/hr to 1 deg/hr)
  - Performance 2: linearity (range: 1 to 5E-06 g/g^2 over range 20-60 g
  - Typically frequently updated with external measurement (Star Trackers, Sun sensors) via a Kalman Filter

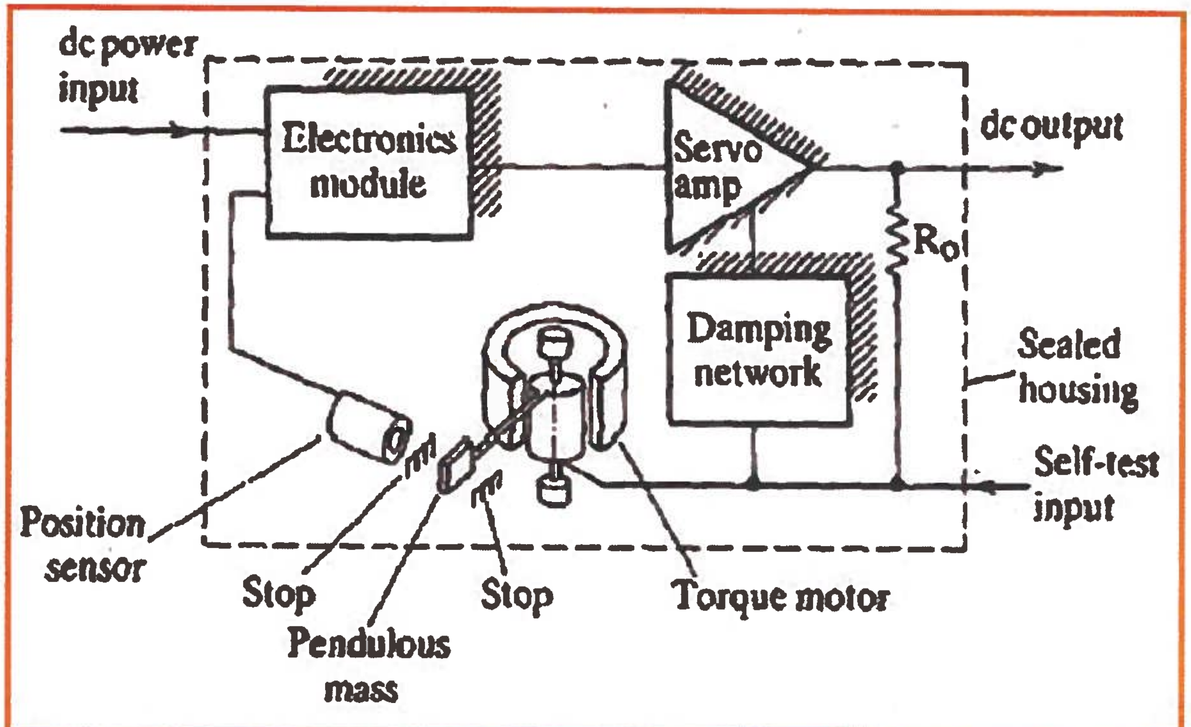
Two-Degree of Freedom Gyroscope



- Free gyro mounted on a gimbaled platform
- Gyro "stores" reference direction in space
- Angle pickoffs on gimbal axes measure pitch and yaw angles
- Direction can be precessed by applying a torque

Force Rebalance Accelerometer

$$\boxed{f = ma}$$



$$\Delta\ddot{x} = f_x/m = \left(-k_d\Delta\dot{x} - k_s\Delta x\right)/m$$

- Voltage required to re-center the proof mass becomes the measure of acceleration

Inertial Measurement Units

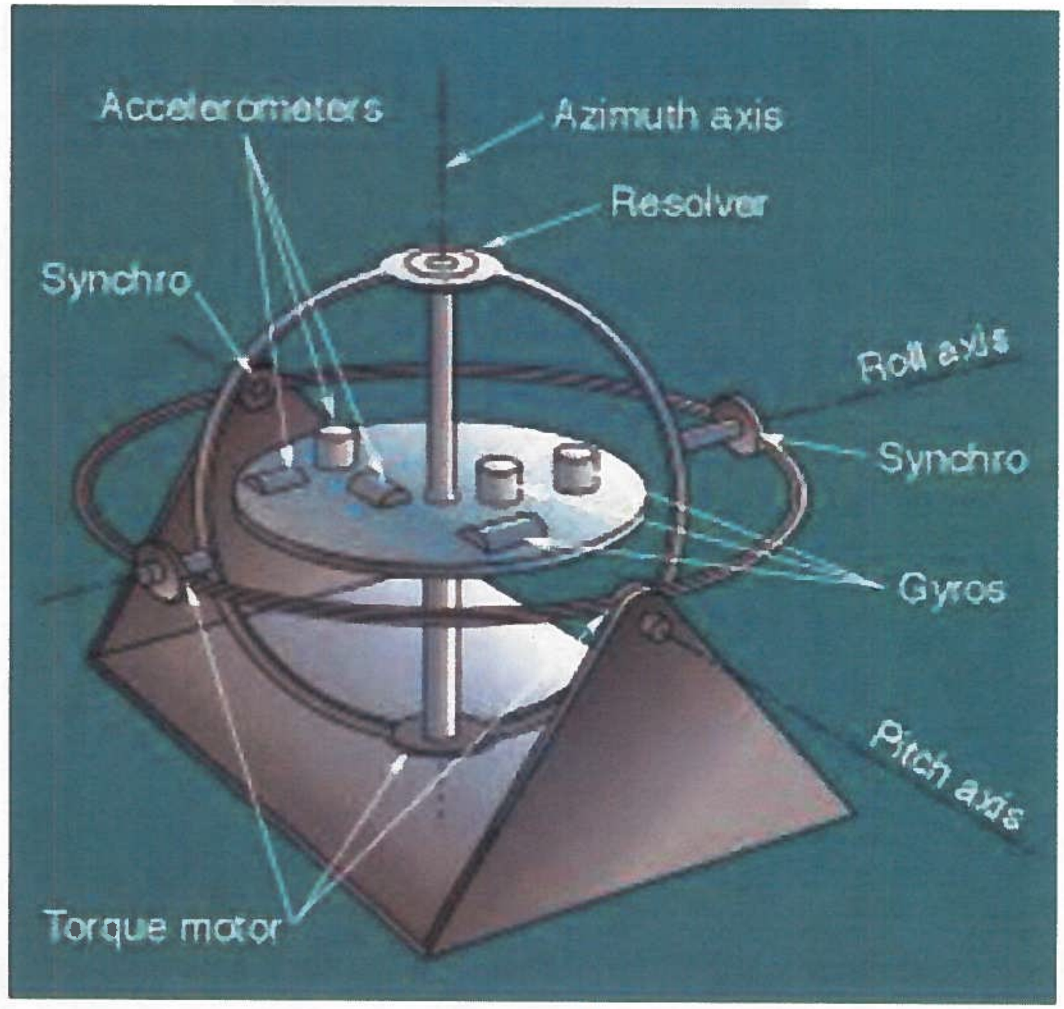
$$\begin{bmatrix} a_x \\ a_y \\ a_z \end{bmatrix} \rightarrow \begin{bmatrix} v_x \\ v_y \\ v_z \end{bmatrix} \rightarrow \begin{bmatrix} x \\ y \\ z \end{bmatrix}$$

$$\begin{bmatrix} \omega_x \\ \omega_y \\ \omega_z \end{bmatrix} \rightarrow \begin{bmatrix} \phi \\ \theta \\ \psi \end{bmatrix}$$

- 3 accelerometers
- 3 rate or rate-integrating gyroscopes
- Platform orientation "fixed" in space
- Vehicle rotates about the platform

- Need for high precision
- Drift due to errors and constants of integration
- Platform re-oriented with external data (e.g., GPS)

Gimbaled Physical Platform



Types of Gyroscopes

- Classic Spinning Disk
  - Based on gyroscopic effect of a of rotating object with angular momentum.

$$\tau = \frac{dh}{dt}$$

- Vibratory
  - Based on Coriolis force on an object in a rotating reference frame.

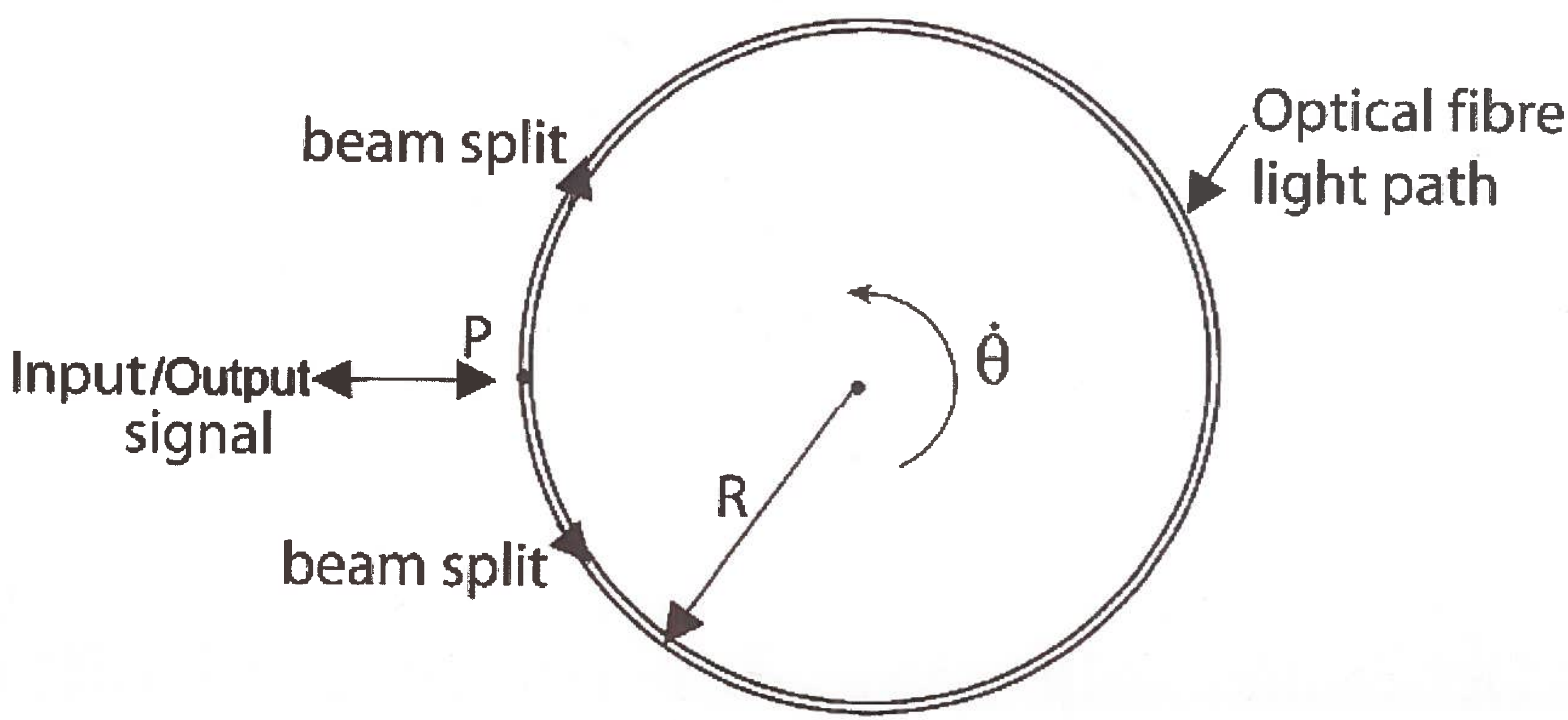
$$F_{coriolis} = 2m\left(\vec{v} \times \vec{\vec{U}}\right)$$

- Laser (Ring, IFOG)
  - Based on "timing" the travel of light through a course whose length varies with the rotation of the course.

$$Speed\ of\ \ Light = c = \text{Constant}$$

Interferometric Fiber Optic Gyroscope (IFOG)

- Coherent (laser) light travels in opposite directions around a fiber optic coil.
- Rotation of the coil creates a path difference between the signals.
- Measuring the phase shift between the signals provides a rotation rate measurement.



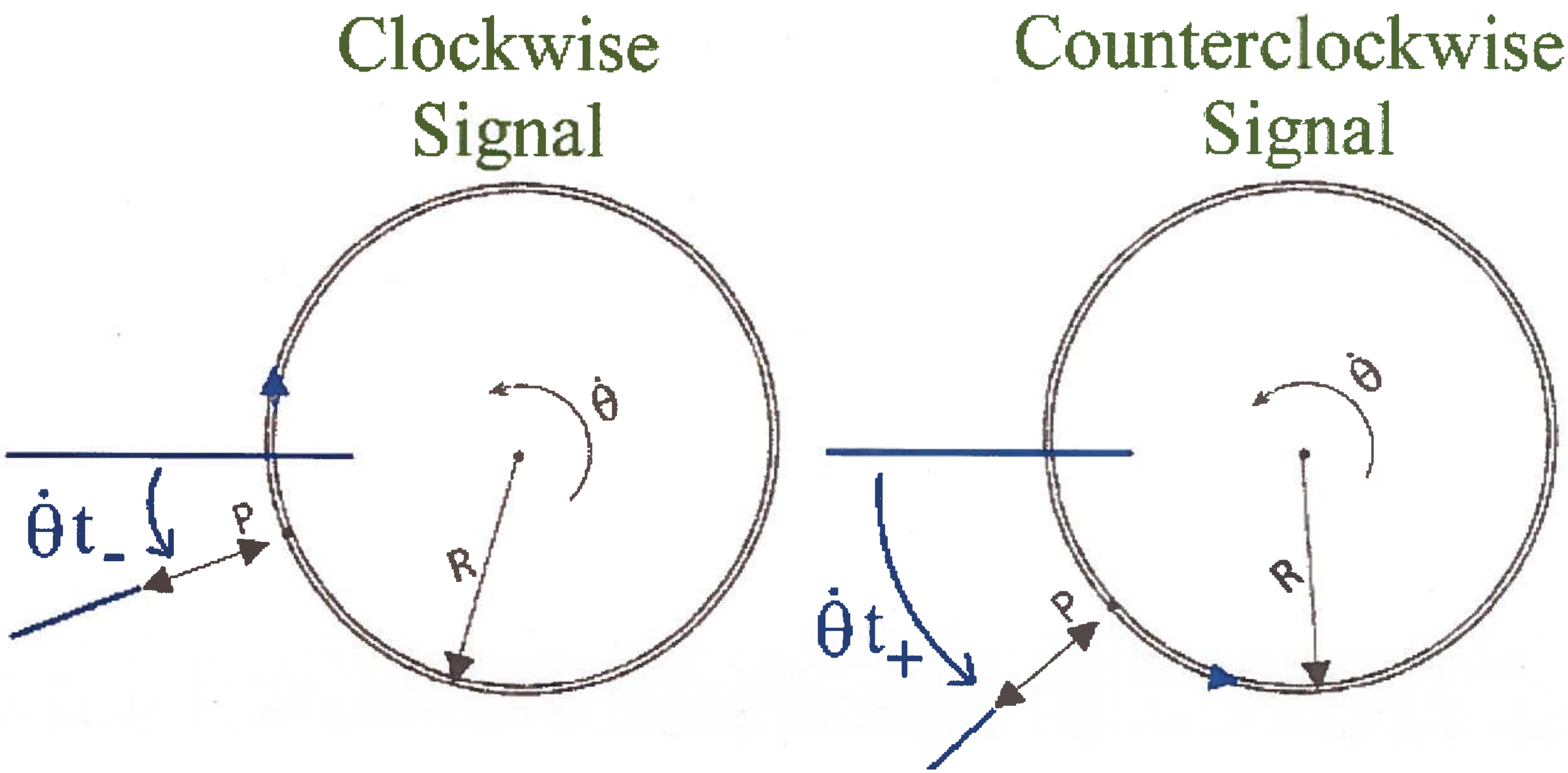
Path Difference

- Distance Traveled by Clockwise Signal:

$$ct_- = 2\pi R - R\dot{\theta}\, t_- \Rightarrow ct_- = \frac{2\pi cR}{c + R\dot{\theta}}$$

- Distance Traveled by Counterclockwise Signal:

$$ct_+ = 2\pi R + R\dot{\theta}\, t_+ \Rightarrow ct_+ = \frac{2\pi cR}{c - R\dot{\theta}}$$



Path Difference

- Take difference of distance traveled by each signal.

$$dL = c(t_+ - t_-) = 2\pi cR\left[\frac{1}{(c - R\dot{\theta})} - \frac{1}{(c + R\dot{\theta})}\right]$$

$$dL = 2\pi cR\left[\frac{(c + R\dot{\theta})}{(c^2 - R^2\dot{\theta}^2)} - \frac{(c - R\dot{\theta})}{(c^2 - R^2\dot{\theta}^2)}\right] = 2\pi cR\left[\frac{2R\dot{\theta}}{(c^2 - R^2\dot{\theta}^2)}\right]$$

$$dL = \frac{4\pi cR^2\dot{\theta}}{(c^2 - R^2\dot{\theta}^2)}$$

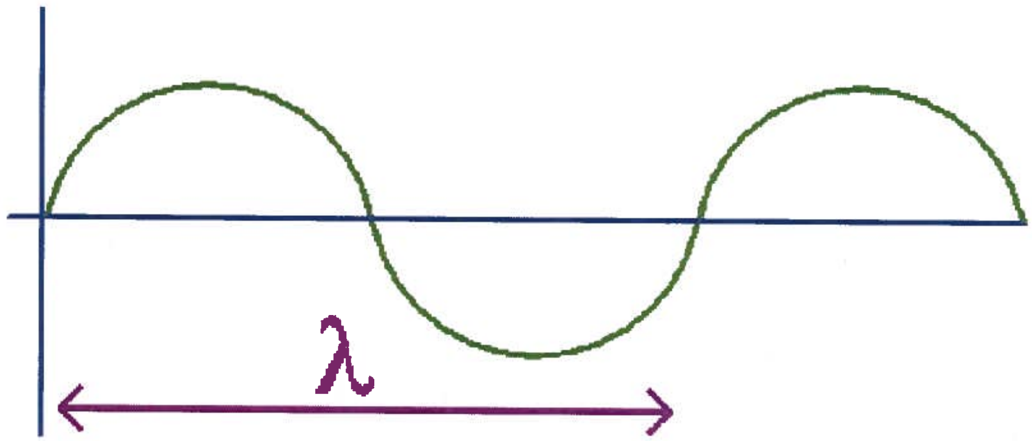
$$c >> R^2\dot{\theta}^2$$

- Path Difference: $$\boxed{dL = \frac{4\pi R^2\dot{\theta}}{c}}$$

Light Interference

- The coherent light exiting after traveling different distances have a phase difference proportional to rotation rate.

$$dL = \frac{4\pi R^2 \dot{\theta}}{c}$$



$$dL = \lambda \frac{d\phi}{2\pi}$$

$$\boxed{\Rightarrow \dot{\theta} = \frac{\lambda c}{8\pi^2 R^2} d\phi}$$

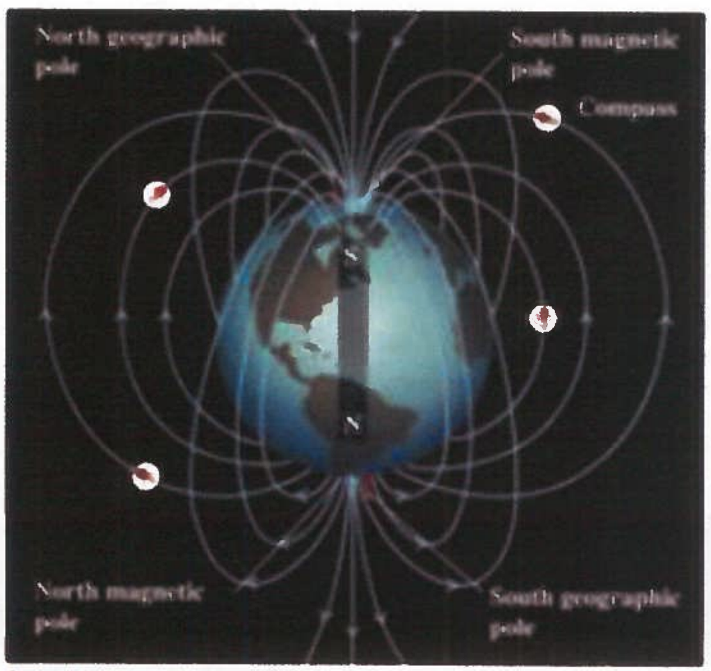
IFOG Summary

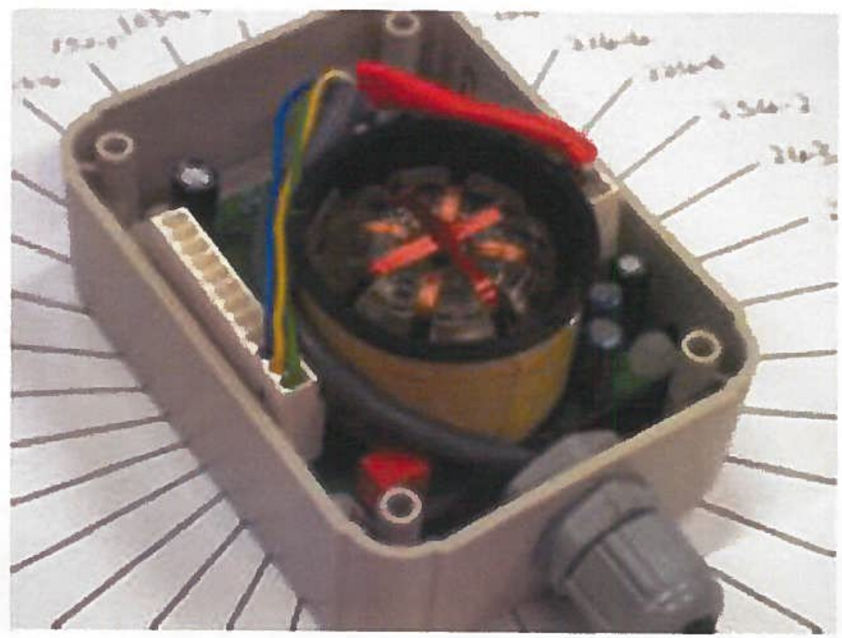
- Advantages
  - No mechanical parts
  - Resistant to shock and vibration
  - Long-lived
  - Accurate
  - Commercially Available: First Used in Boeing 777

- Disadvantages
  - Speed of light is “fast” and thus requires many loops of fiber optic fiber to create a detectable phase angle.

Magnetometer

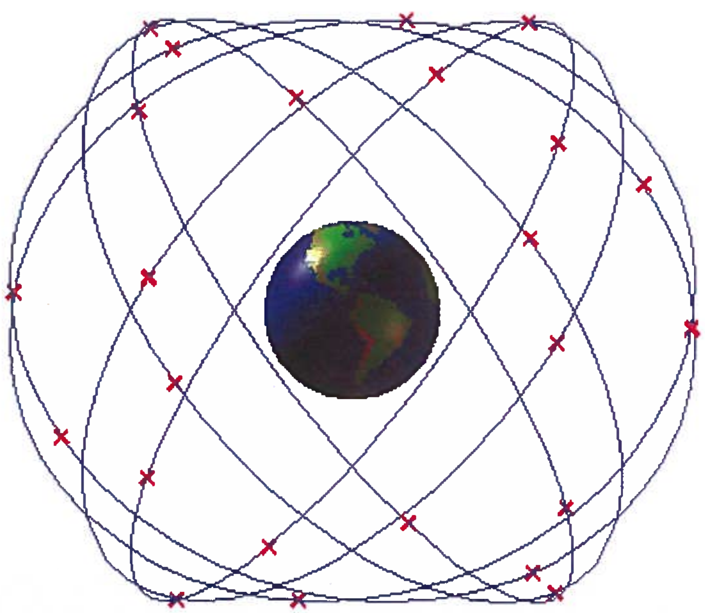
- Flux gate magnetometer
  - Alternating current passed through one coil
  - Permalloy core alternately magnitized by electromagnetic field
  - Corresponding magnetic field sensed by second coil
  - Distortion of oscillating field is a measure of one component of the Earth's magnetic field
- Three magnetometers required to determine Earth's magnetic field vector





ACS Sensors: GPS and Magnetometers

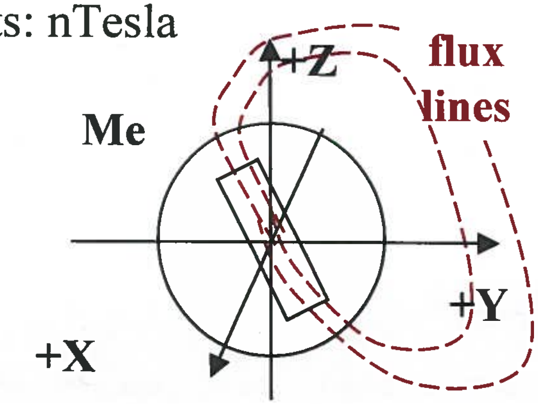
- Global Positioning System (GPS)
  - Currently 27 Satellites
  - 12hr Orbits
  - Accurate Ephemeris
  - Accurate Timing
    - Stand-Alone 100m
    - DGPS 5m
    - Carrier-smoothed DGPS 1-2m



- Magnetometers
  - Measure components Bx, By, Bz of ambient magnetic field B
  - Sensitive to field from spacecraft (electronics), mounted on boom
  - Get attitude information by comparing measured B to modeled B
  - Tilted dipole model of earth's field:

$$\begin{bmatrix} B_{north} \\ B_{east} \\ B_{down} \end{bmatrix} = \left(\frac{6378}{r_{km}}\right)^3 \begin{bmatrix} -C_\varphi & S_\varphi C_\lambda & S_\varphi S_\lambda \\ 0 & S_\lambda & -C_\lambda \\ -2S_\varphi & -2C_\varphi C_\lambda & -2C_\varphi S_\lambda \end{bmatrix} \begin{bmatrix} -29900 \\ -1900 \\ 5530 \end{bmatrix}$$

Where: C=cos , S=sin, ϕ=latitude, λ=longitude
Units: nTesla



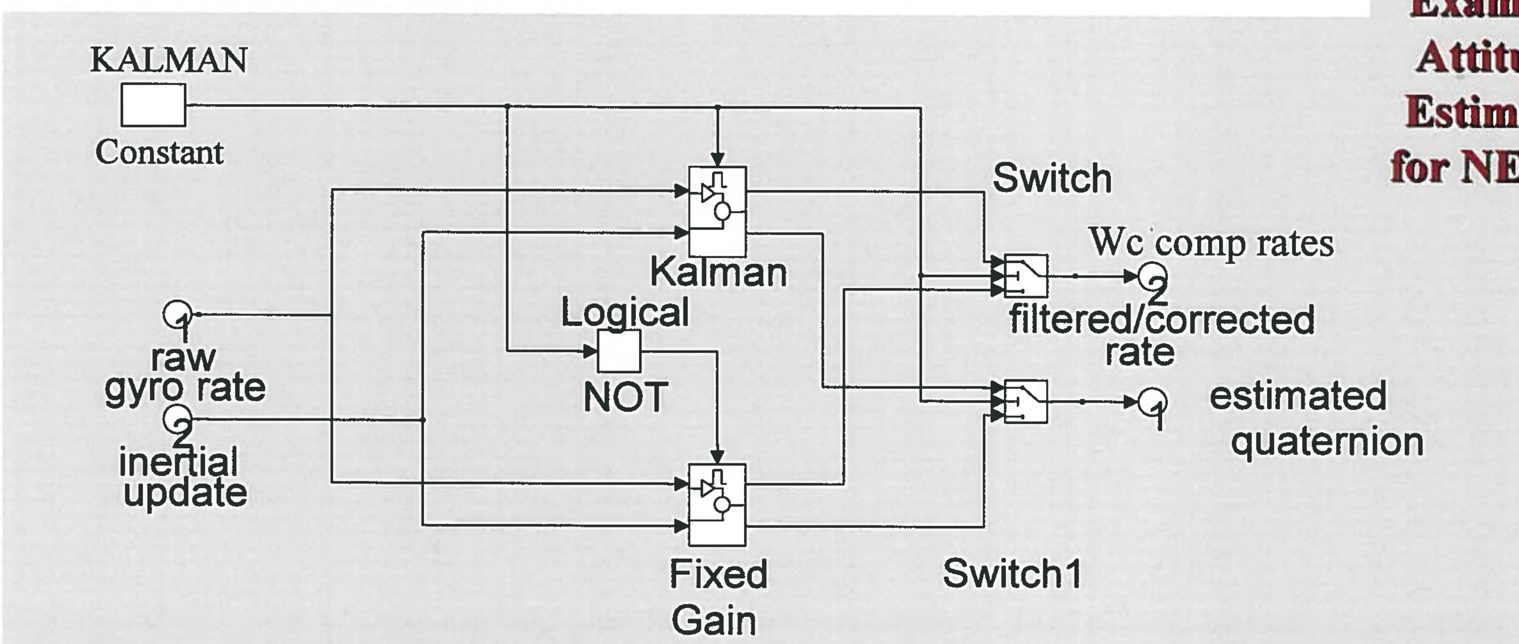
ACS Sensor Performance Summary

Reference	Typical Accuracy	Remarks
Sun	1 min	Simple, reliable, low cost, not always visible
Earth	0.1 deg	Orbit dependent; usually requires scan; relatively expensive
Magnetic Field	1 deg	Economical; orbit dependent; low altitude only; low accuracy
Stars	0.001 deg	Heavy, complex, expensive, most accurate
Inertial Space	0.01 deg/hour	Rate only; good short term reference; can be heavy, power, cost

Attitude Determination

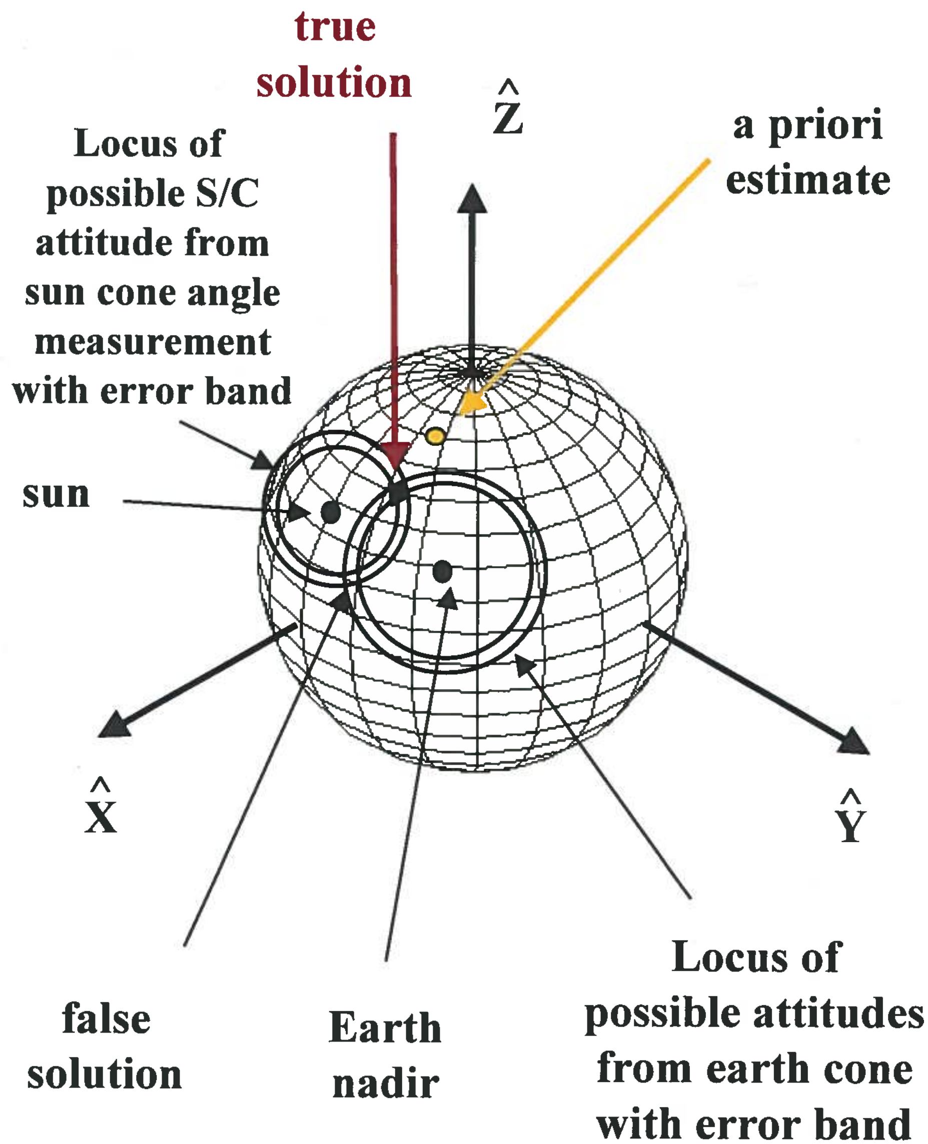
- Attitude Determination (AD) is the process of of deriving estimates of spacecraft attitude from (sensor) measurement data. Exact determination is NOT POSSIBLE, always have some error.
- Single Axis AD: Determine orientation of a single spacecraft axis in space (usually spin axis)
- Three Axis AD: Complete Orientation; single axis (Euler axis, when using Quaternions) plus rotation about that axis

Example: Attitude Estimator for NEXUS



Single-Axis Attitude Determination

- Utilizes sensors that yield an arc-length measurement between sensor boresight and known reference point (e.g. sun, nadir)
- Requires at least two independent measurements and a scheme to choose between the true and false solution
- Total lack of a priori estimate requires three measurements
- Cone angles only are measured, not full 3-component vectors. The reference (e.g. sun, earth) vectors are known in the reference frame, but only partially so in the body frame.



Three-Axis Attitude Determination

- Need two vectors (u,v) measured in the spacecraft frame and known in reference frame (e.g. star position on the celestial sphere)
- Generally there is redundant data available; can extend the calculations on this chart to include a least-squares estimate for the attitude
- Do generally not need to know absolute values

$$|u|,|v|$$

Define:

$$\hat{i} = u/|u|$$

$$j = (u \times v)/|u \times v|$$

$$\hat{k} = \hat{i} \times \hat{j}$$

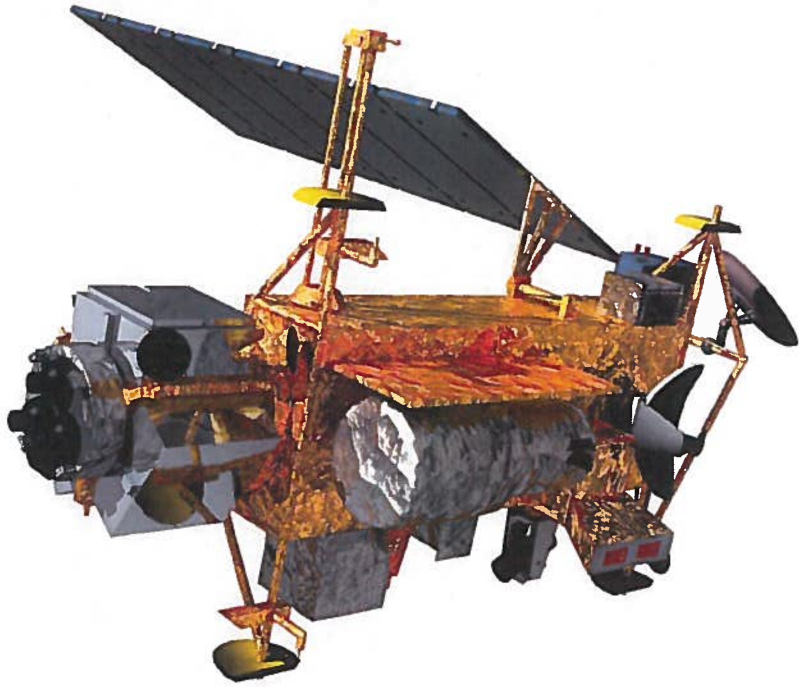
Want Attitude Matrix T:

$$\underbrace{\begin{bmatrix} \hat{i}_B & \hat{j}_B & \hat{k}_B \end{bmatrix}}_{M} = T \cdot \underbrace{\begin{bmatrix} \hat{i}_R & \hat{j}_R & \hat{k}_R \end{bmatrix}}_{N}$$

So: $T = MN^{-1}$

<u>Note:</u> N must be non-singular (= full rank)

UARS: Upper Atmosphere Research Satellite



Mission type	Earth observation
Operator	NASA
COSPAR ID	1991-063B
Website	http://umpgal.gsfc.nasa.gov/
Mission duration	14 years, 3 months
	Spacecraft properties
Bus	Multimission Modular Spacecraft
Manufacturer	Martin Marietta
Launch mass	5,900 kilograms (13,000 lb)
Power	1600.0 watts
	Start of mission
Launch date	12 September 1991, 23:11:04 UTC
Rocket	Space Shuttle *Discovery* STS-48

Example:

UARS Attitude Control System

Upper Atmosphere Research Satellite (1991 - 2005)