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Satellite platforms

Vendela Paxal

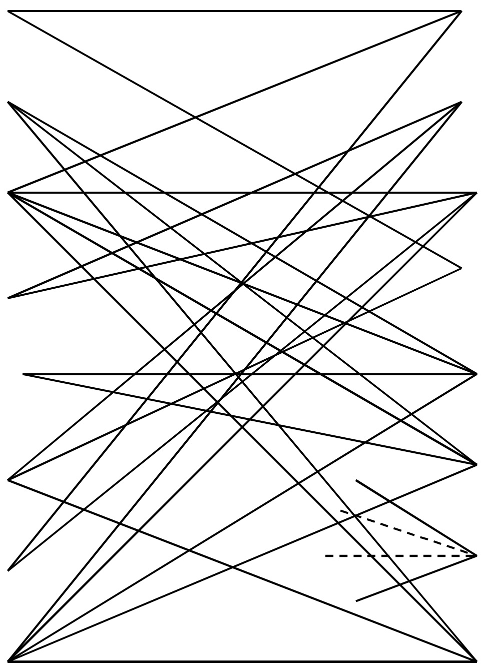
Overview

- Requirements
- Mass control
- Attitude and orbital control
- Power supply
- Telemetry, Tracking & Command
- Thermal control
- Mechanical
- Payload
- Debris

Requirements

- Requirements linked to the mission of the satellite
- Requirements linked to being able to operate in the hostile space environment
- Requirements on how long to operate
- Requirements in order to be able to launch
- Requirements for keeping the orbit, and orientation towards Earth
- Requirements to keep contact with the satellite
- Requirements for cost
- Requirements to control your satellite after its death

Controls:
- Mass
- Orbit
- Power
- Temperature
- Telemetry
- Mechanical
- Payload
- Debris



Requirements:
- Mission
- Environment
- Lifetime
- Launch
- Orbit keeping
- Contact
- Cost
- Debris

Mass control

”A satellite is worth its weight in gold”

The more sophisticated the satellite is, the longer development time, and the more things can go wrong.

The insurance price increases to a level out of proportions, or it may even become impossible to insure the satellite.

The mass to launch is extremely important.

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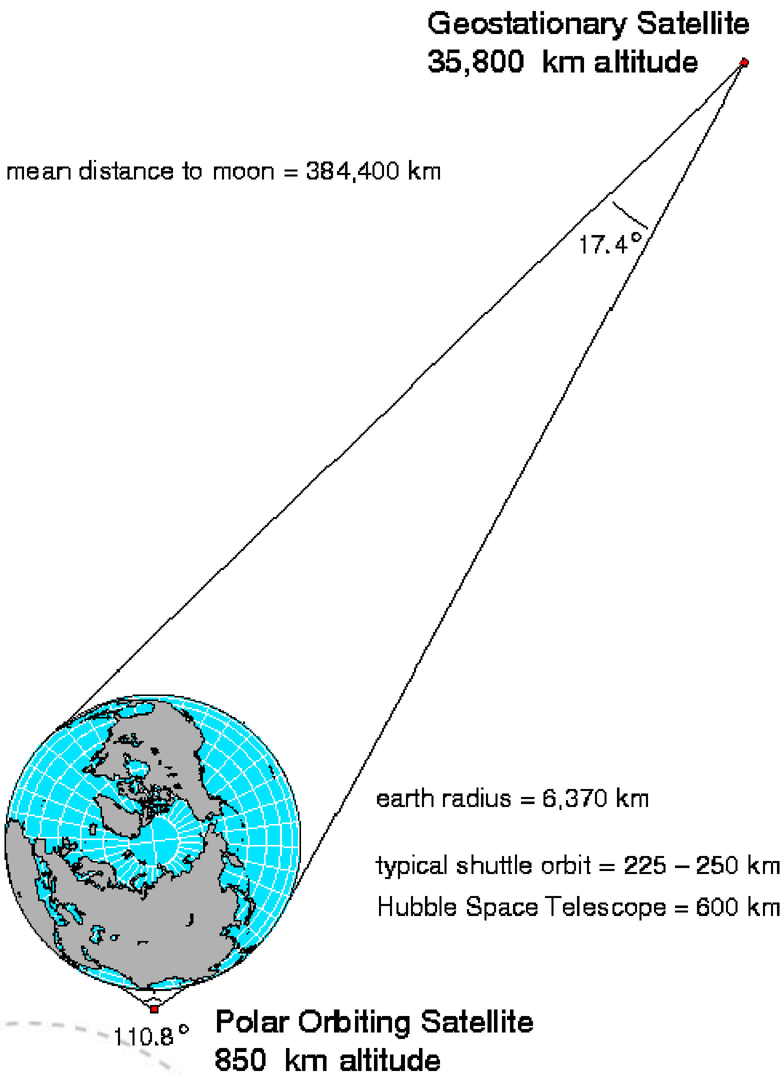
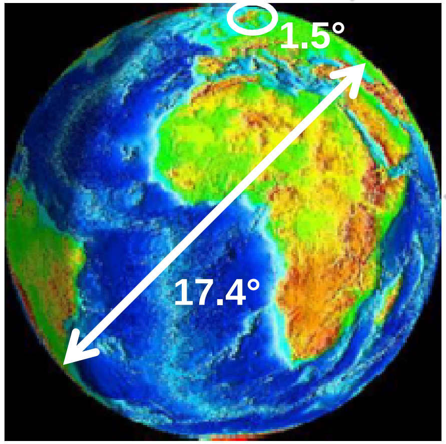
Orbit control





The satellite's view of Norway from our allocated position in space: 1°W

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Power control

It is usually the available power that will be determinant for the satellite’s lifetime, not the ageing of the structure or the components.
Fuel is used for orbit control, from launch, via correcting manouvers, to into graveyard fireing.
Solar panels are used for electronics and continuous operation.

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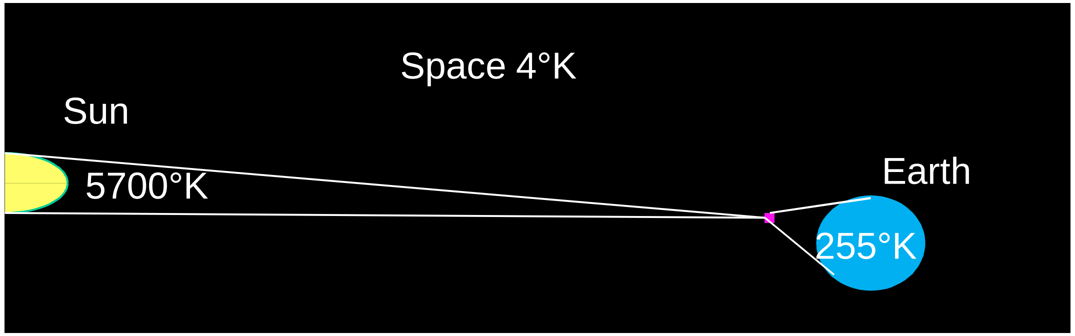
Temperature control



Operating temperature ranges:

Batteries:	0 to 20°C
Solar panels:	-100 to 50°C
Electronics:	-10 to 60°C
Fuel:	10 to 50°C

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Telemetry

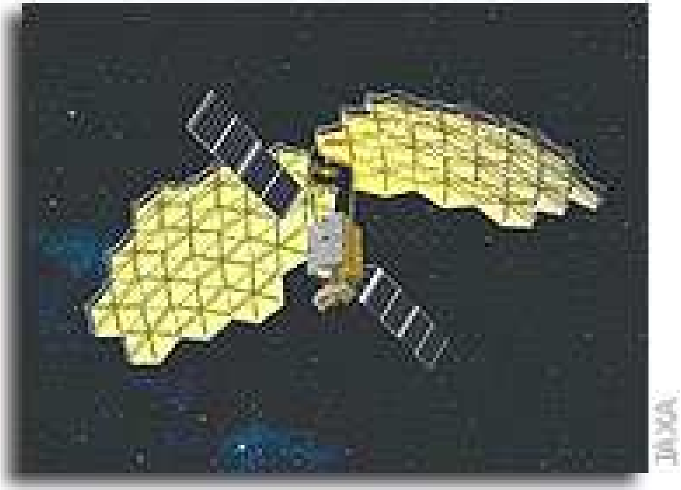
The communication channel between the control centre on Earth and the satellite.

A satellite operator needs to monitor and operate all the control systems; the solar panels, the temperature, the orbital corrections, the deployment and orientation of antennas, control of transmit power, and maneuvers to avoid impacts from debris collisions and sun storms.

The application itself may also need the telemetry link, e.g. for upload of new software, correction of bugs, traffic monitoring and control (OBP).

Mechanical

- Mechanical structures are important for launch. The interface between the satellite and the rocket is challenging.
- The structures for mounting the components are important.
- Protection to radiation and debris may save a satellite.
- Mechanical deployment of antennas and solar panels is vital for power and telemetry.



Japanese KIKU 8 satellite launched in 2006

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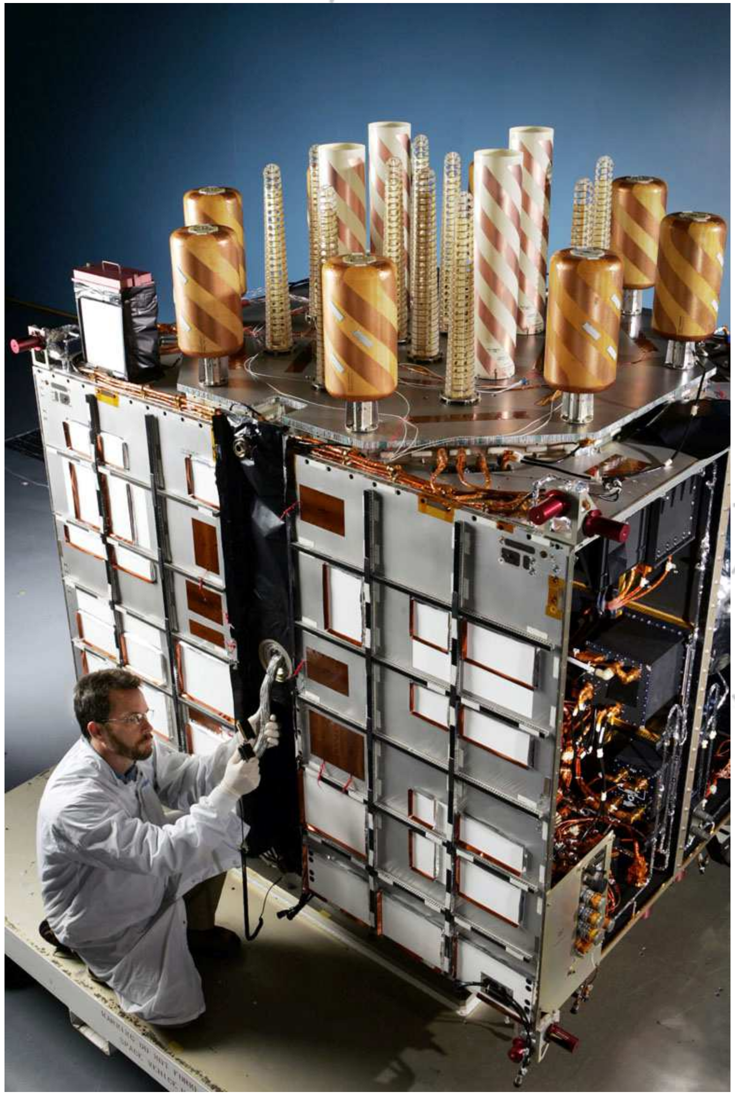
http://www.asc-csa.gc.ca/includes/video_window.asp?src=/videos/radarsat2_deployment_lr.wmv&titre=déploiement%20de%20l'antenne%20rso%20et%20des%20panneaux%20solaires%20du%20satellite%20radarsat-2%20après%20le%20lancement&404=1&ref=

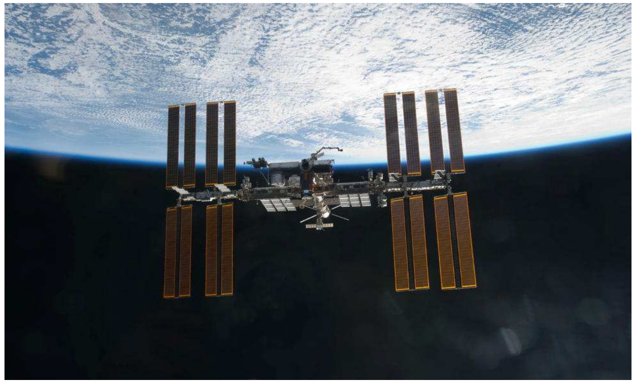
Payload

- Navigation
- Remote sensing
- Communication



Purpose: Generally to create revenue

Debris



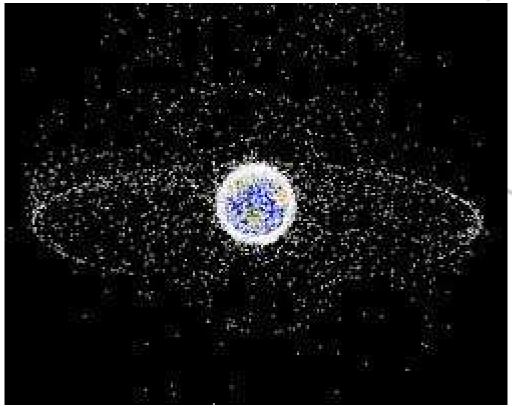
(Dagbladet) 06.02.2011: Mannskapet på Den internasjonale romstasjonen (ISS) har de siste timene vært klare til å evakuere inn i den tilliggende russiske Sojuz-romkapselen på grunn av faren for kollisjon med kinesisk romskrot - en kollisjon som kunne fått katastrofale følger.

Det eventuelle sammenstøtet ville ifølge beregningene ha funnet sted klokka 22.21 norsk tid - men Nasa kan allerede nå avkrefte at det blir noen kollisjon; romsøppelet vil ikke engang komme innenfor den svære sikkerhetssonen på 25 x 25 x 0,75 km.

Nasas kontrollsenter i Houston har holdt øye med skrotet - restene av en kinesisk satellitt Kina selv skjøt ned i 2007 - siden i dag tidlig, og informerte kommandør Dmitrij Kondratjev om at han og de to andre om bord burde stå klare til å søke tilflukt i Sojuz TMA-20-fartøyet.

Tre andre astronauter er for tida på vei til ISS, og skal etter planen legge til i morgen. Dermed vil romstasjonen ha fullt mannskap på seks, bestående av personer fra USA, Russland og Italia.

Romsøppel er et stadig voksende problem; rundt 12 500 kjente større biter skrot befinner seg i bane rundt jorda, det antas at tallet på biter store nok til å gjøre alvorlig skade er 600 000. Skrotet stammer stort sett fra utrangerte satellitter, og problemet blir ikke mindre i ei tid med mer og mer aktivitet i jordas umiddelbare nærhet.



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1. Mass control

”A satellite is worth its weight in gold”

The more sophisticated the satellite is, the longer development time, and the more things can og wrong.

The insurance price increases to a level out of proportions, or will be impossible to insure.

The mass at launch is extremely important.

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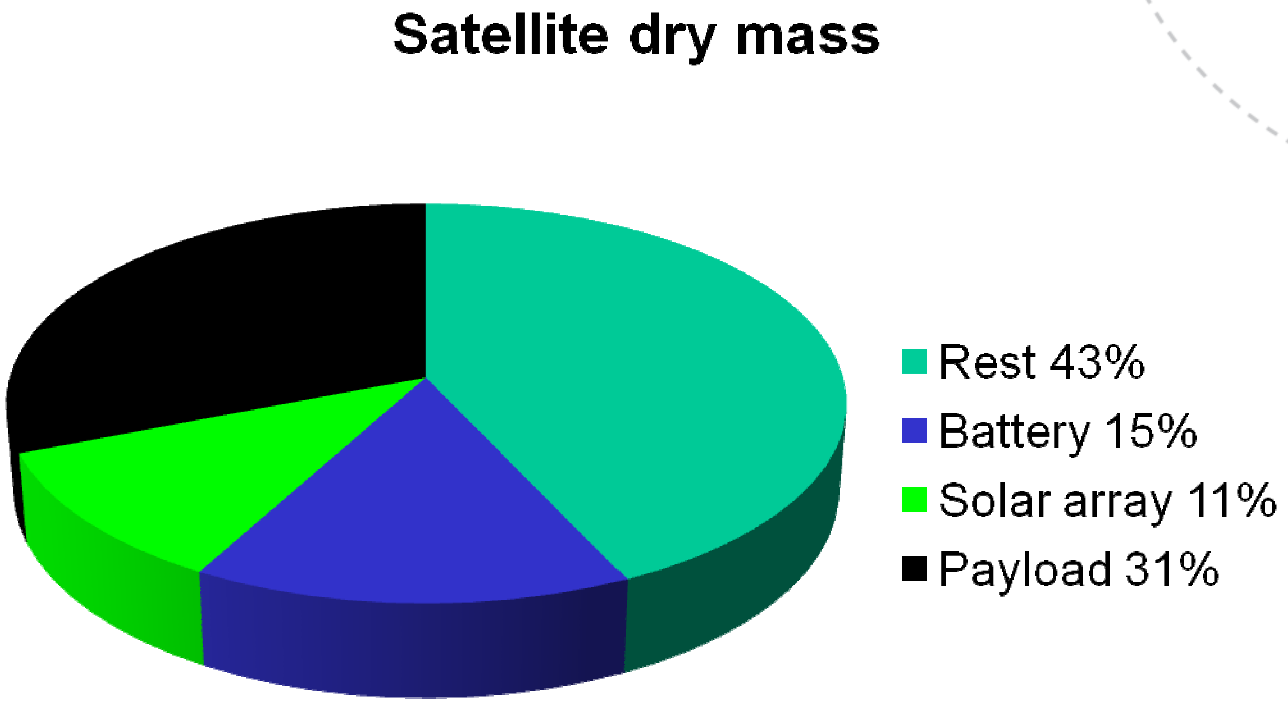
Dry mass brakdown



The Rocket equation

The Rocket equation (from Newton's 2nd law):

$\Delta v = I_{sp} \cdot g_0 \cdot \ln(m_i/m_f) = I_{sp} \cdot g_0 \cdot \ln(m_i/(m_i - m_p))$

Δv = the velocity change induced by the ignition of the propellant (m/s)

I_{sp} = propellant's specific impulse (s)

g_0 = gravitational acceleration at sea level (9.81m/s^2)

m_i = the total initial mass (kg) = $m_s + m_p$

m_f = the total final mass = $m_i – m_p = m_s$

m_s = satellite mass (structure + payload)

m_p = propellant mass

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See appendix C9 in Understanding Space

1^{st} and 2^{nd} cosmic speeds

The 1^{st} cosmic velocity, v, is the velocity needed to set an object into orbit at the earth's surface (assuming vacuum):

$F_{sentr} = F_{grav} \Rightarrow mv^2/r = mg(r) \Rightarrow v = \sqrt{R_E g_0} = 8km/s$

The 2^{nd} cosmic velocity is the velocity needed to escape the gravitaional force of the earth. When the kinetic energy of an object is higher or equal to the gravitational potential enegy:

$E_{kin}=E_{grav} \Rightarrow mv^2/2 = GMm/r \Rightarrow v = \sqrt{2GM/R_E} = 11.2km/s$

G is the universal gravitational constant

M is the mass of the Earth

R_E is the mean radius of the Earth

g_0 is the gravitational acceleration at sea level, and $g_0 = GM/R^2_E$

Consequence

Even if the payload mass equals zero, it is (from the rocket equation) impossible to attain the first cosmic speed with todays propulsion systems (giving I_{sp}). A typical value for possible maximum speed would be about 7km/s.

This is why we always need two stage rockets with today's technology, and this is why the mass budget is of such importance.

Also the higher mass, the higher is the cost.

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Example of component's weight

Part of the proposed exercises.

ΔV_{design} = 10km/s, $I_{sp} \approx 350 - 400$s

$m_{payload}$ = 2 tons

$m_{structure}$ (stage 1 and 2) = 11 tons

$m_{propellant}$ (stage 1 and 2) = 112 tons

$\Rightarrow$ 82% of the initial mass is propellant

$\Rightarrow$ less than 2% of the initial mass is payload

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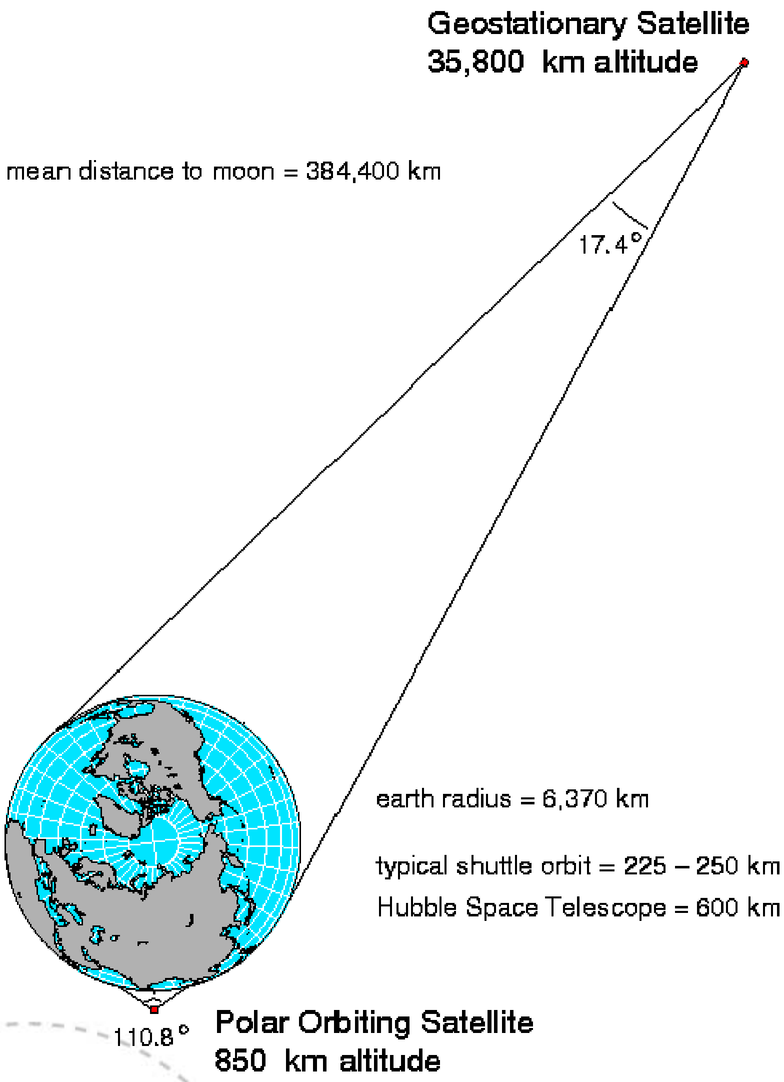
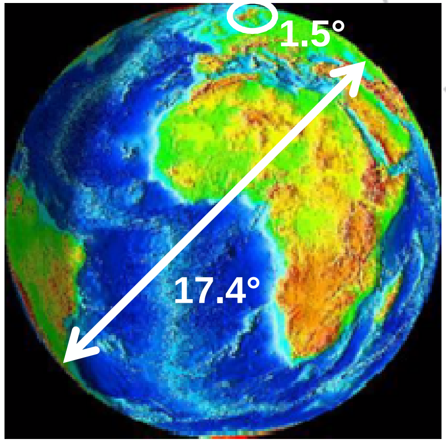
2. Orbit control





The satellite's view of Norway from our allocated position in space: 1°W

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Attitude control

Attitude defines a vehicle’s orientation in space.

The attitude is controlled by the ADCS = Attitude Determination and Control Subsystem

The desired attitude depends on the space craft’s mission.

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Orbital perturbations

- Atmospheric friction
- Irregularities of the earths gravitational field
- Lunar gravity
- Solar gravity
- Solar pressure

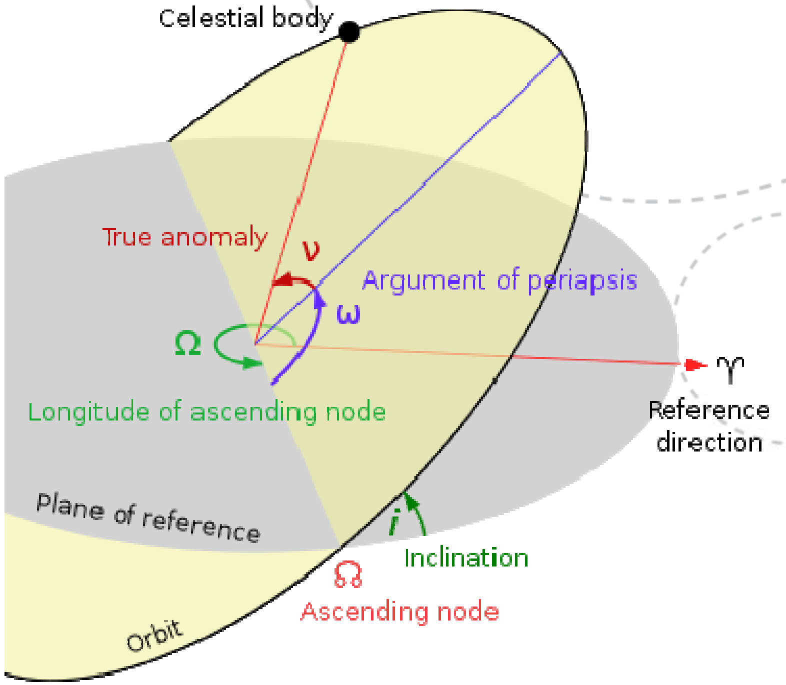
Drift of node and perigee

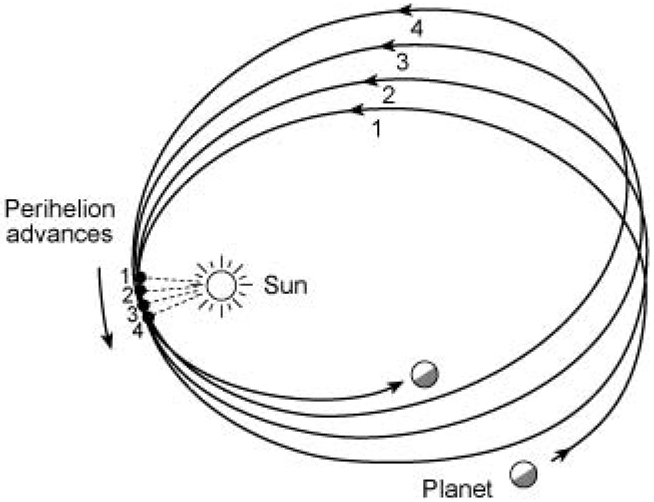
Orbital precession (change in the orientation of the rotation axis of a rotating body).

Nodal drift: $\Delta\Omega/\Delta t = -9.98 \cdot \cos(i) / ((1-e^2)^2 \cdot (a/R)^{7/2})$

Perigee drift $\Delta\omega/\Delta t = 5 \cdot (5 \cdot \cos^2(i)-1)/((1-e^2)^2 \cdot (a/R)^{7/2})$

$\Delta\omega/\Delta t=0$ when $i=63.4°$ and $i=180°-63.4°=116.6°$
which is the inclination of a Molniya orbit.





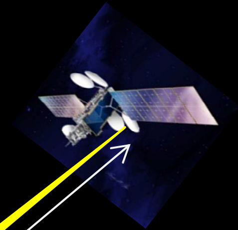
Pointing accuracy

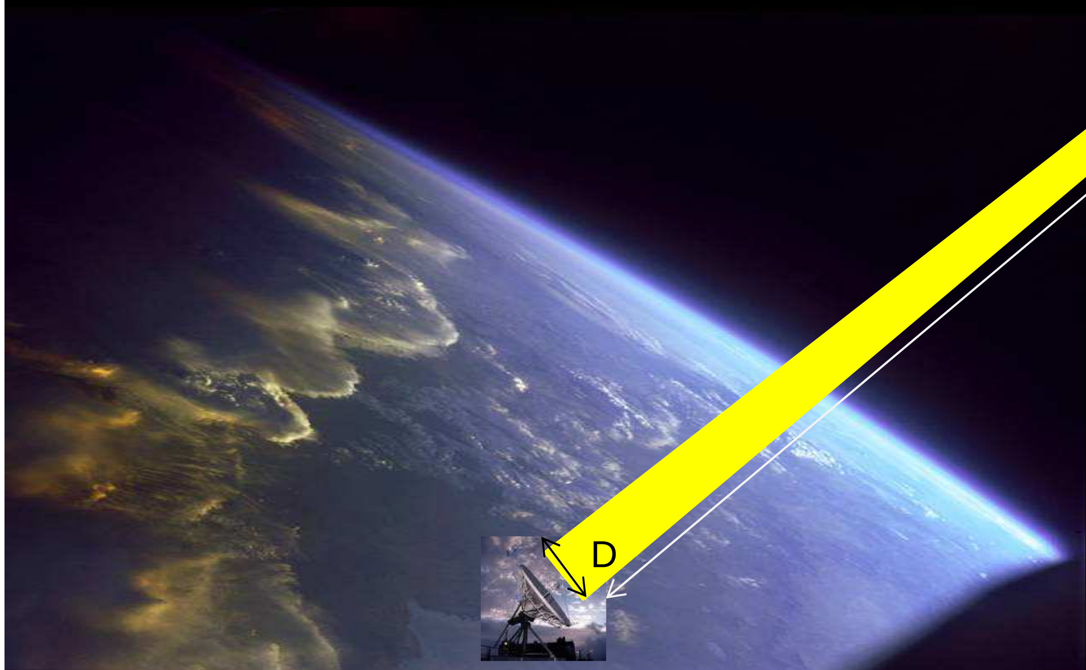
Ψ=D/h

Ψ is the pointing accuracy

h is the distance to target

D is the diameter of the target



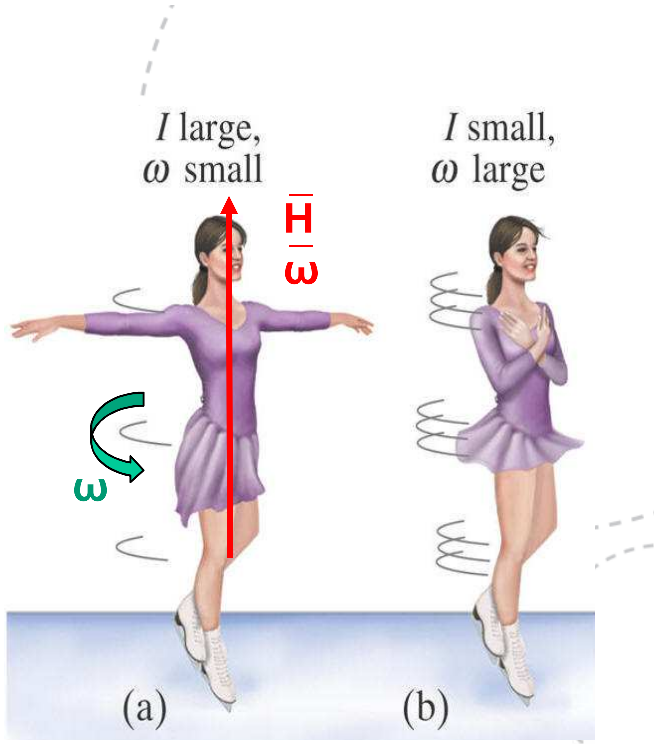


<u>Antenna Beamwidth (deg):</u>

$\theta_{3dB} = k \times \lambda / A$

k reflector characteristic coefficient
A is the antenna diameter,
λ is the wavelength

Fundamental dynamics



$\bar{H} = I \cdot \bar{\omega}$

H = angular momentum [kg·m^2/s]

I = moment of inertia [kg·m^2]

ω = angular velocity [rad/s]

Force: a change in momentum $\bar{F} = d\bar{p}/dt$

$\bar{T} = \bar{R} \times \bar{F}$

T = Torque [N·m]

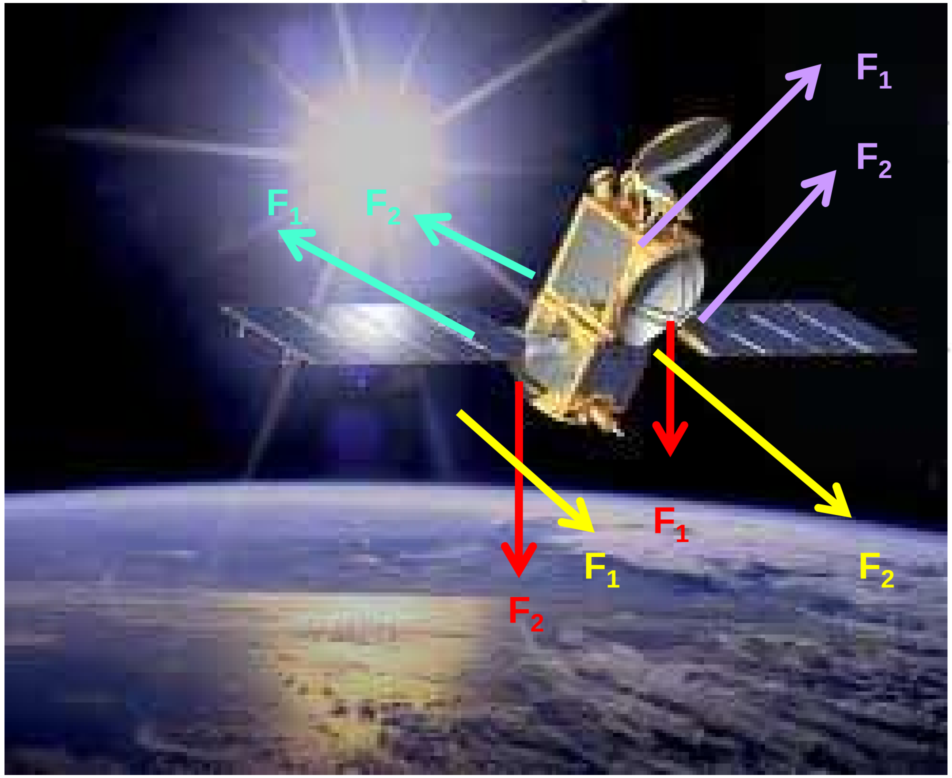
R = distance from cm to the force attac point

F = force applied

$\bar{T} = d\bar{H}/dt$

Torque on a satellite

- Gravity-gradient torque
- Solar radiation pressure
- Magnetic torque
- Aerodynamic torque



The torques appear as the forces on different parts of the satellite.

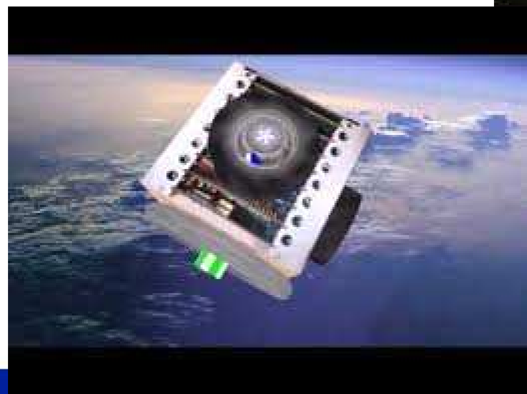
Stabilizing satellites

Gravity-gradient stabilization:
work with the torque, not against it

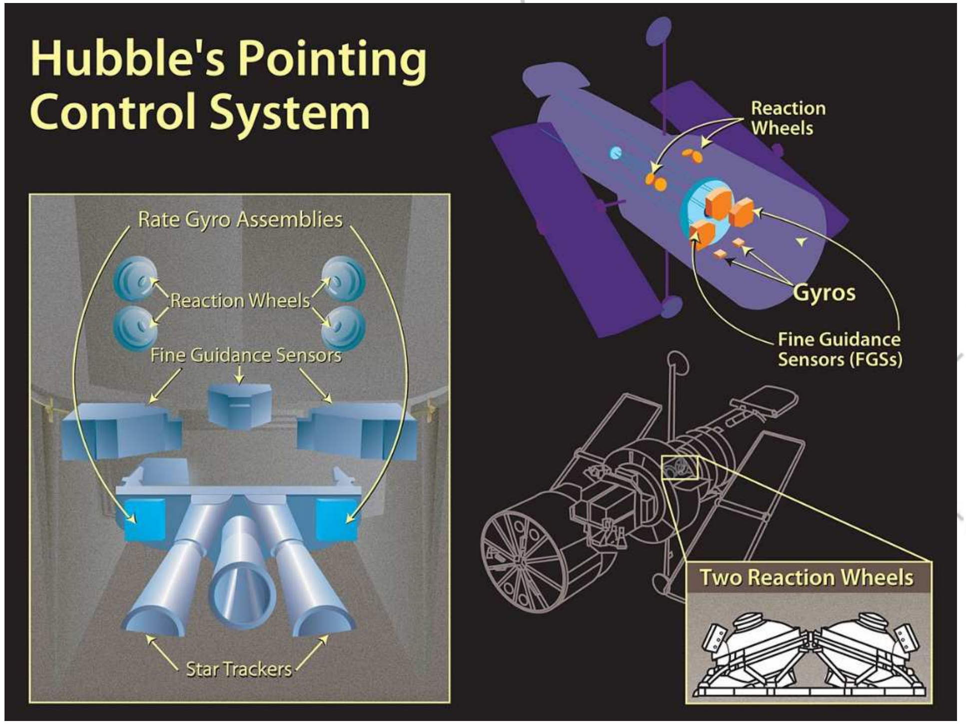
Spin stabilized: profit from gyroscopic stiffness; the faster an object spins, the more stable it is

Thrusters: fire off engines to exercise opposite torque

Momentum-control:
conservation of angular momentum



Use of wheels



Momentum wheels

The same as spin stabilised

Reaction wheels

Conservation of momentum

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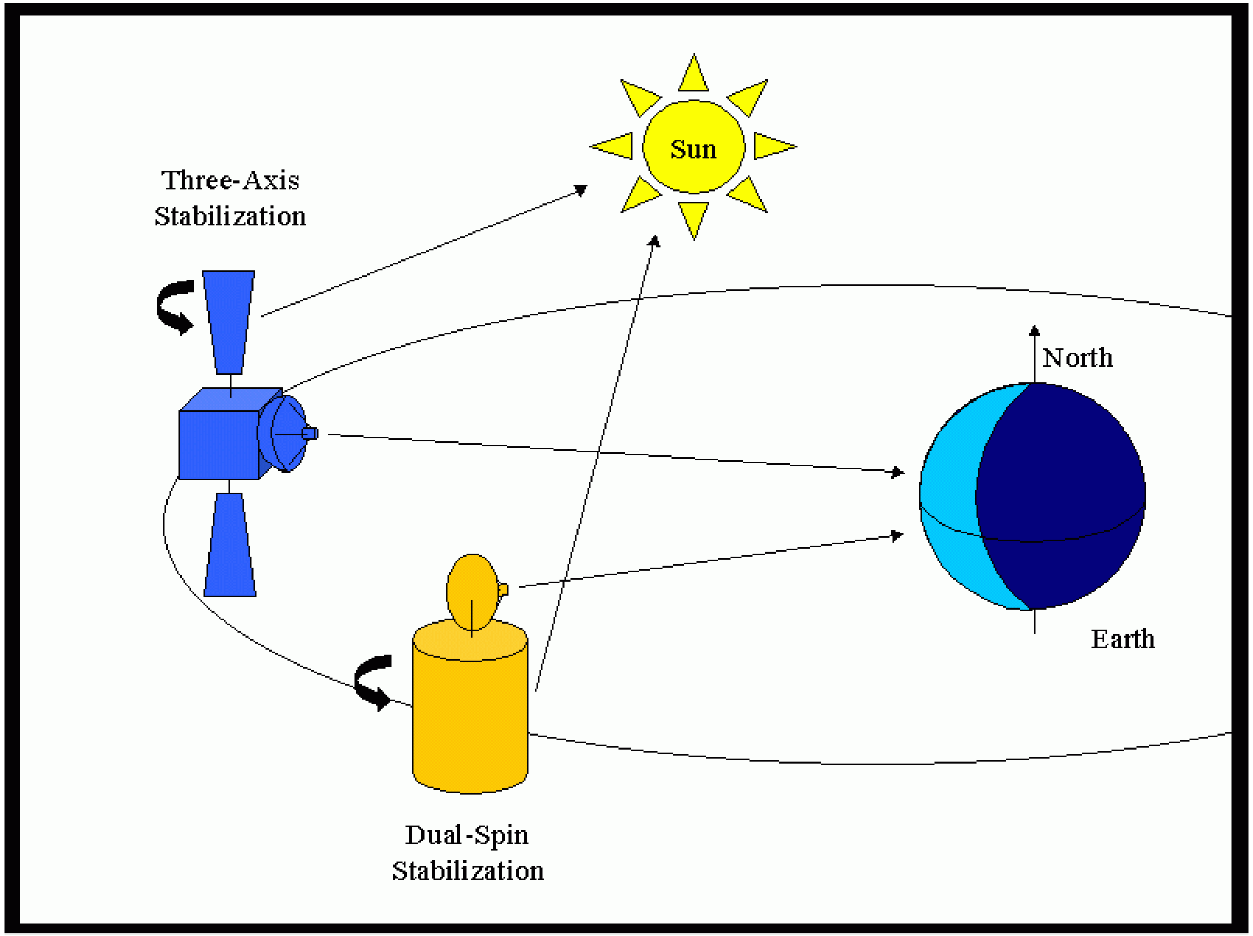
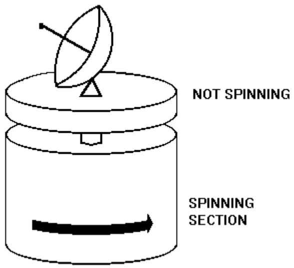
Dual spin and three axis stabilazation, antenna orientation





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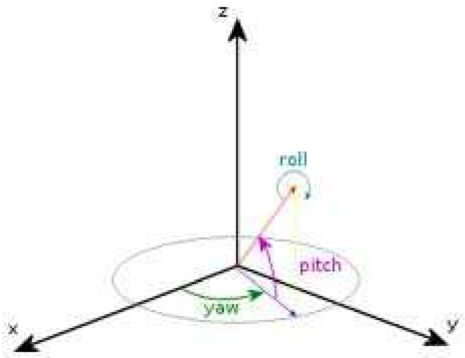
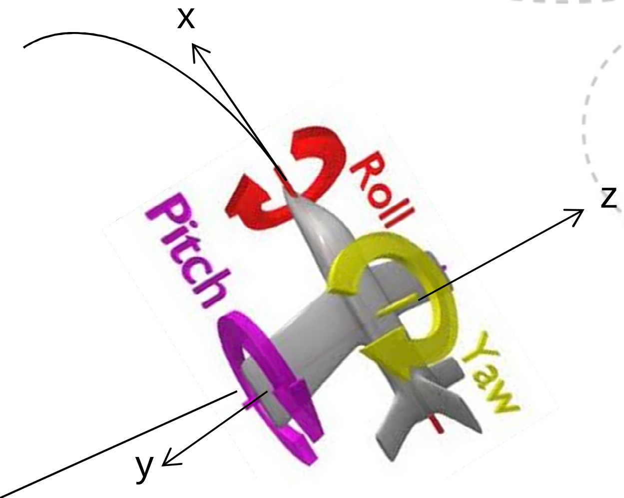
Yaw (gir), pitch (stamp) and roll (rull)





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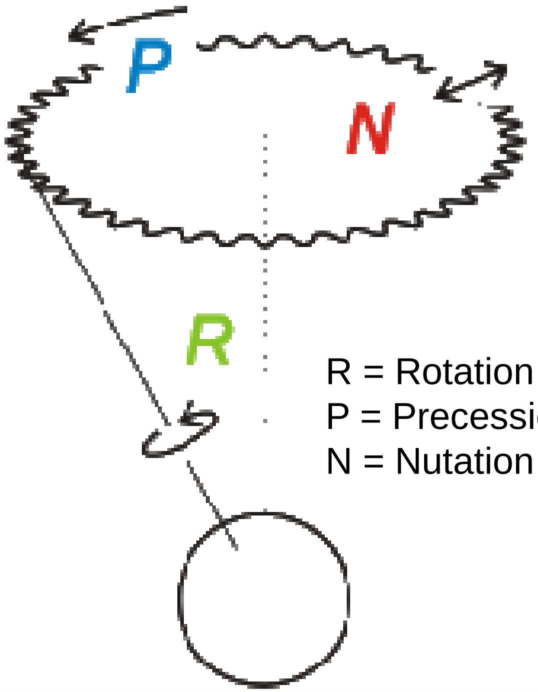
Reduce precession

May be reduced e.g. with a closed tube filled with viscous liquid.

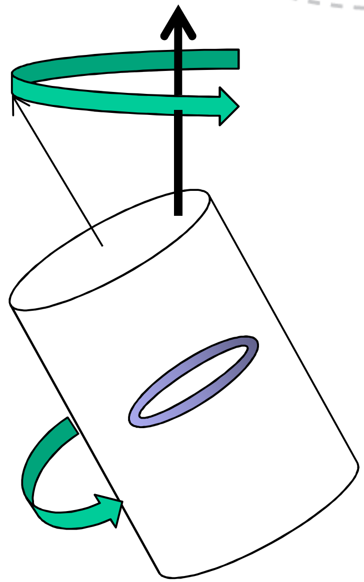
Passive: friction of the liquid, heat dissipation.

Active: pulse mode thrusters.

Nutation is a nodding or rocking movement of the axis of rotation.



R = Rotation
P = Precession
N = Nutation



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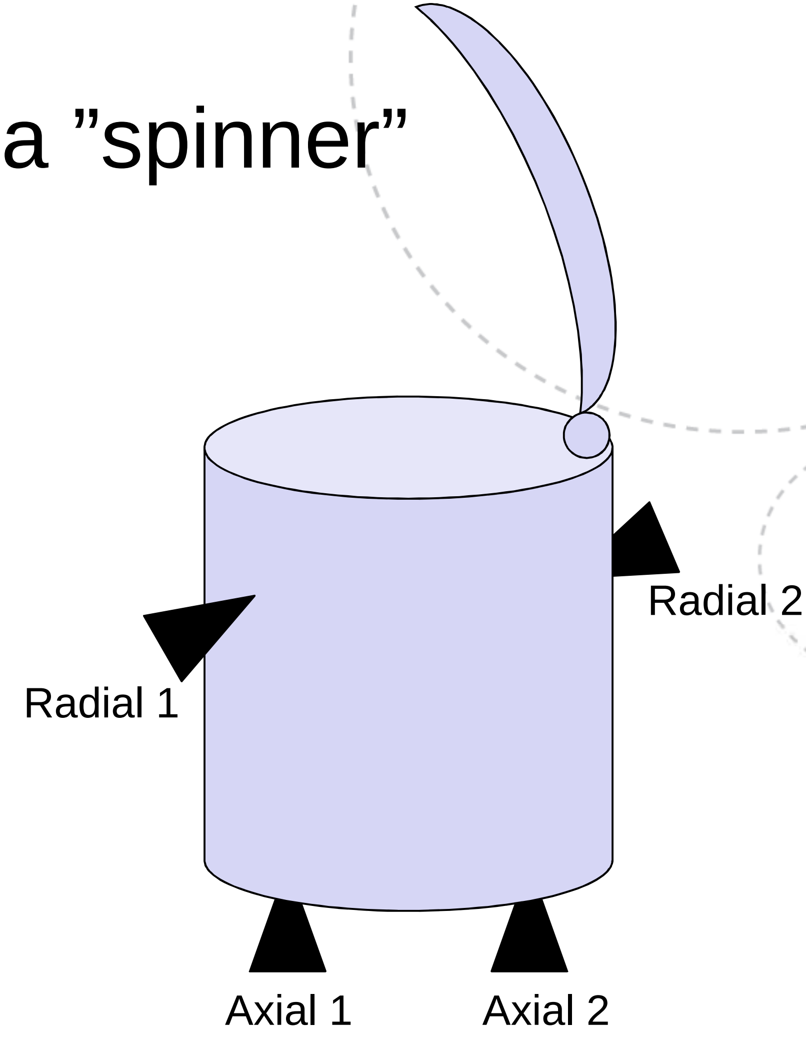
Location of thrusters on a ”spinner”

Operational example, Thor satellite:

Radial 1 or 2 are used for drift and eccentricity (east/west) maneuver every 14 days.

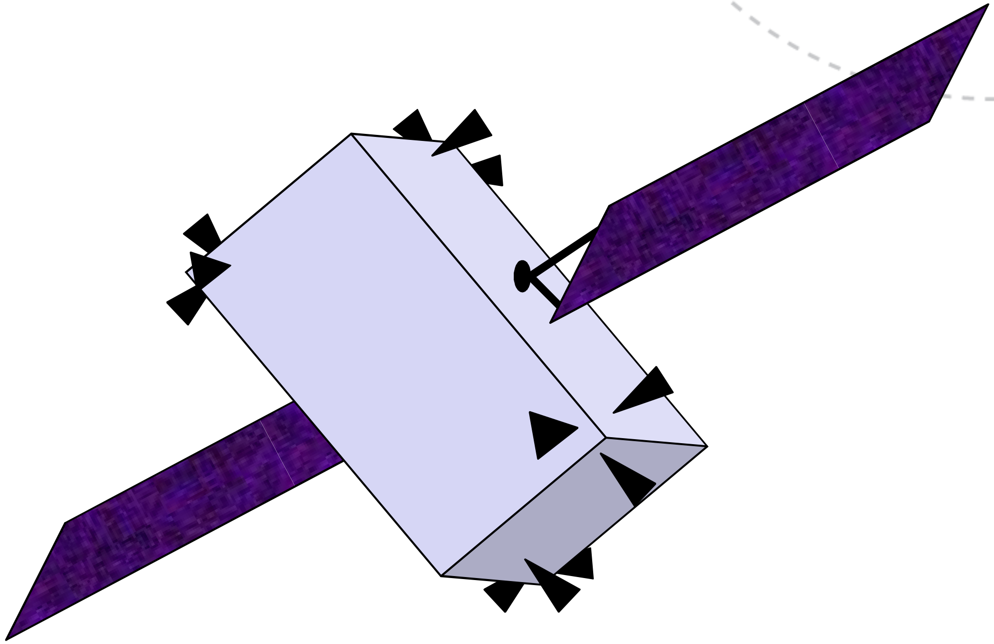
Axial 1 or 2 are used for attitude control every week.

Axial 1 and 2 are used for inclination maneuver once per month.

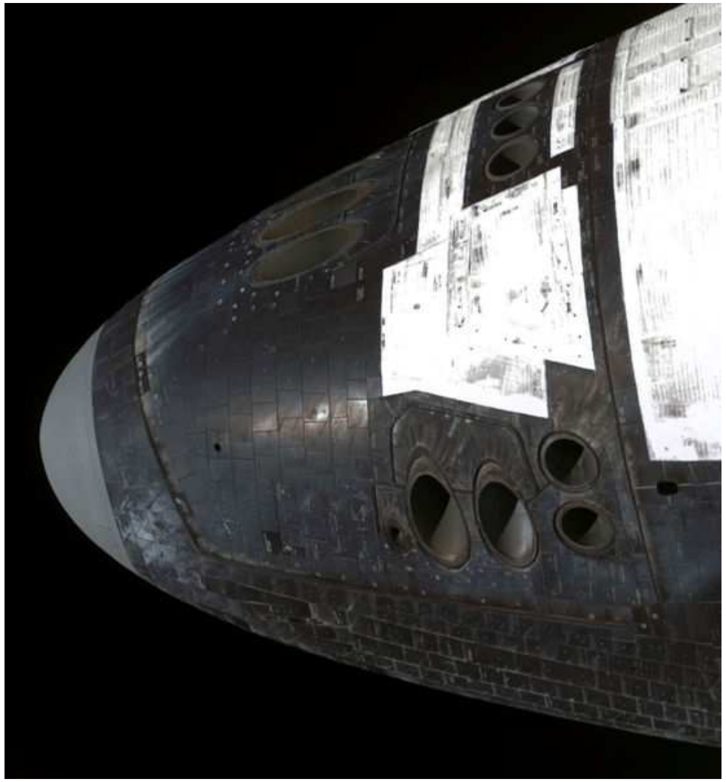


Location of thrusters on a ”3-axis”-stabilized satellite

Movements in x, y and z directions with torque.



Examples



Space shuttel

Apollo – Lunar orbiter

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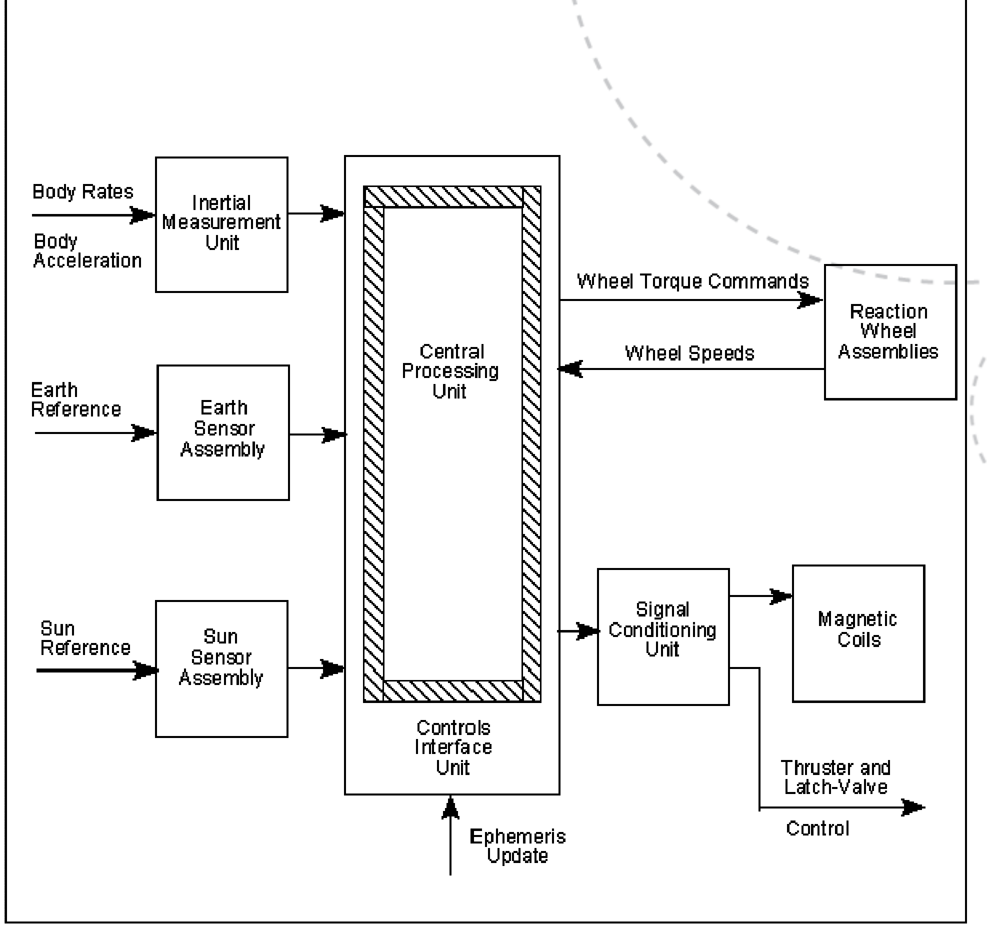
Extra-slide

The Attitude Determination and Control Subsystem (ADACS) provides, in conjunction with the RCS and Command and Control Subsystems (CCS), the functions of the on-orbit attitude control and ascent guidance. It is a zero-momentum system consisting of reaction wheels and Earth, Sun, and inertial reference sensors.

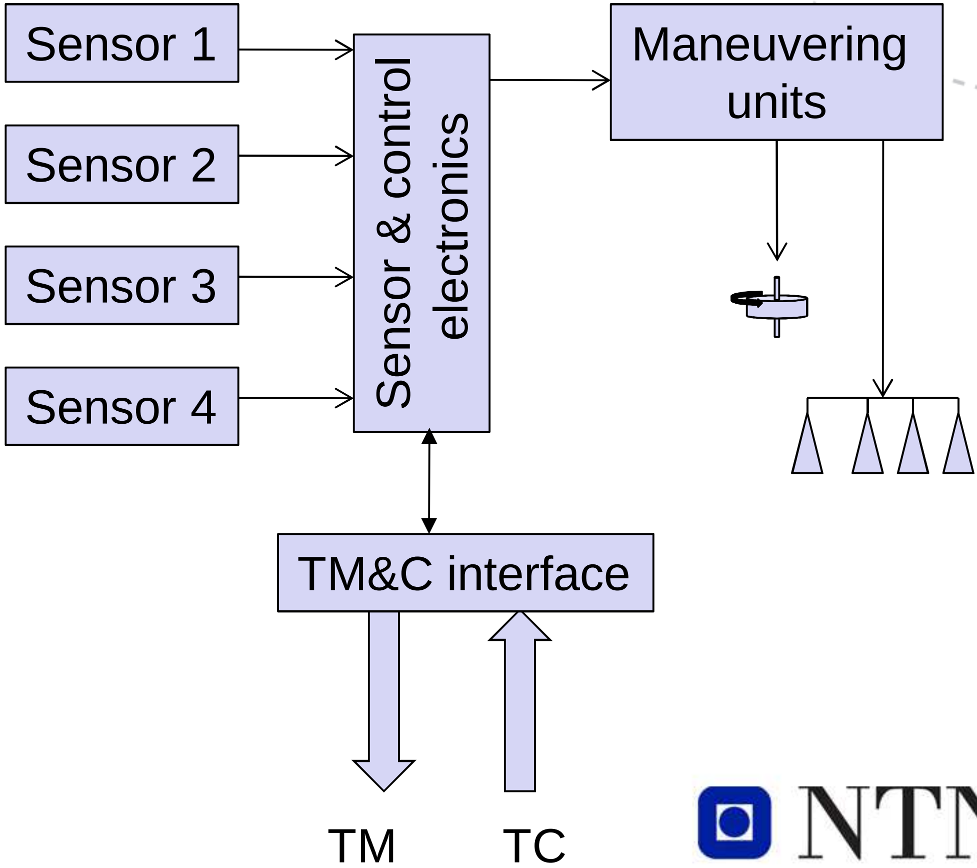
Figure 1.2.4-1. ADACS Simplified Block Diagram.



http://www2.ncdc.noaa.gov/docs/klm/html/c1/sec12-4.htm

Sensors- to know what to do, you must know where you are and how you are moving.

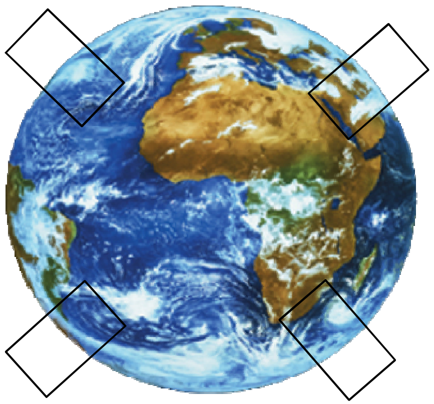
- Earth
- Sun
- Stars
- Gyroscopes
- Magnetometers
- GPS
- A combination of any of the above

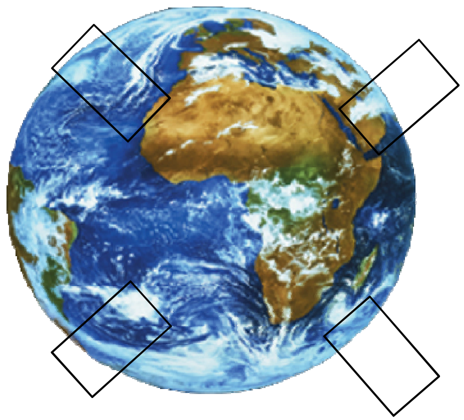


Some sensor examples

- Sun sensor
  - Measures the solar intensity with solar cells. The cell output current is proportional to the solar incidence angle
- Earth sensor
  - Infrared sensors detecting the difference in temperature between the earth and cold space
  - Infrared sensors on spinning satellites detecting the crossing of the earth horizon
- Star sensor
  - Compares the measured star constellation with a stored star map
- Gyroscope sensor
  - Uses the principle of conservation of momentum
  - Ring-laser gyroscope measures the frequency shift between two laser beams in a rotating cavity

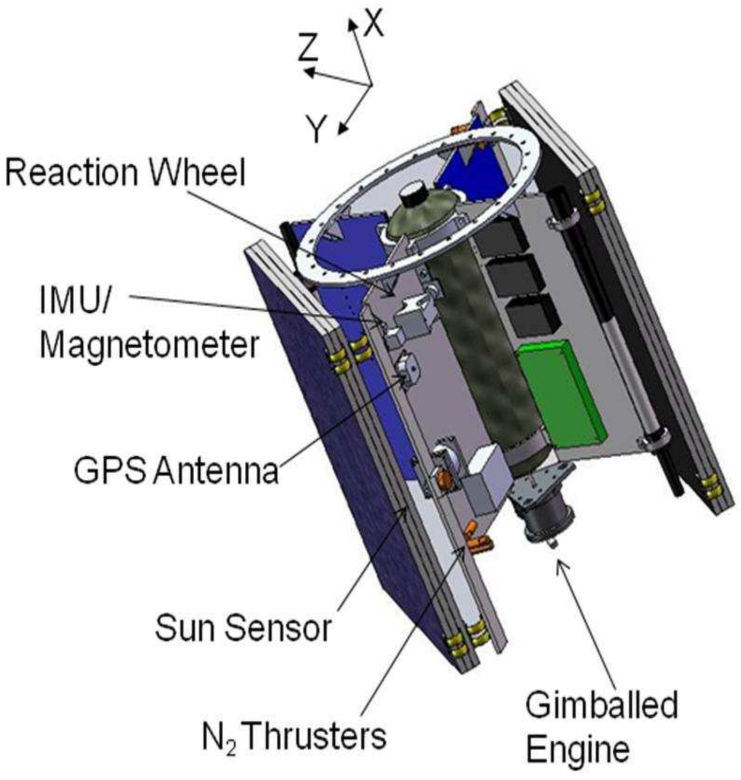
Earth sensor examples





MIT Nano-satellite

- The ACS/GNC team is responsible for tracking and maintaining the attitude and position of the vehicle. The team is responsible for the design and integration of all sensing and actuating mechanisms on the OSMV, including the necessary code or electronics boards.
- For attitude sensing, the OSMV will have a sun sensor that rotates with one solar array at its base, a 3-axis rate gyro and accelerometer, and 2 magnetometers. To correct and maintain the attitude, a reaction wheel parallel to the thrust axis will provide attitude correction around that axis, the gimbaled main engine will further correct or maintain the attitude in the remaining 2 directions, and 12 N2 cold gas thrusters will provide 3-axis stabilization particularly for de-tumbling prior to commissioning and after each eclipse.



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http://satellite.mit.edu/acs.php

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NIFRO-prisen

NIFRO = Norsk industriforum for romvirksomhet

DAVID MICHAEL BANG, UNIVERSITETET I OSLO, ER VINNER AV NIFRO-PRISEN FOR BESTE MASTEROPPGAVE 2013.

Norsk Industriforum for romvirksomhets pris for beste masteroppgave innen romteknologi 2013 deles under Romindustriens årskonferanse; Space Dinner på Grand Hotel i Oslo 11. februar 2014.

Juryen begrunner avgjørelsen slik: «Kandidaten har gjennom sin oppgave dokumentert en god forståelse for hvilke krav som stilles til sensorer og elektroniske komponenter som skal virke ombord i satellitter i bane rundt jorda. Kandidaten har kommet med gode innovative løsningsforslag, og har etter juryens mening utvist både forståelse, refleksjon og modenhet innen sitt fagområde.

Kandidaten har arbeidet med en solsensor som skal «fly» på UiO's studentsatellitt CubeSTAR. Det er utviklet en fungerende solsensor for måling av satellittens pekevinkler i rommet, og det er gjennomført en rekke tester for verifikasjon av systemet. Sensoren tilfredsstiller de krav som er satt til pekenøyaktighet, og den vil inngå i studentsatellittens retnings- og kontrollsystem.

Slik juryen ser det, har har kanditaten en dyp forståelse av problemet som

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Space Dinner 2014

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Kina har landet på månen

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Ti nye Vega

India på tur til Mars

Laser bedre enn radio?

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Ny japansk bærerakett

Skal studere månens

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From 09:30 until 16:00

At Stavanger

Find out more...

03 feb 2015

Space Dinner - Romindustriens årskonferanse 2015

Space Dinner - Romindustriens årskonferanse

NO 09:00

3. Power control

It is usually the available power that will be determinant for the satellite's lifetime, not the ageing of the structure or the components.

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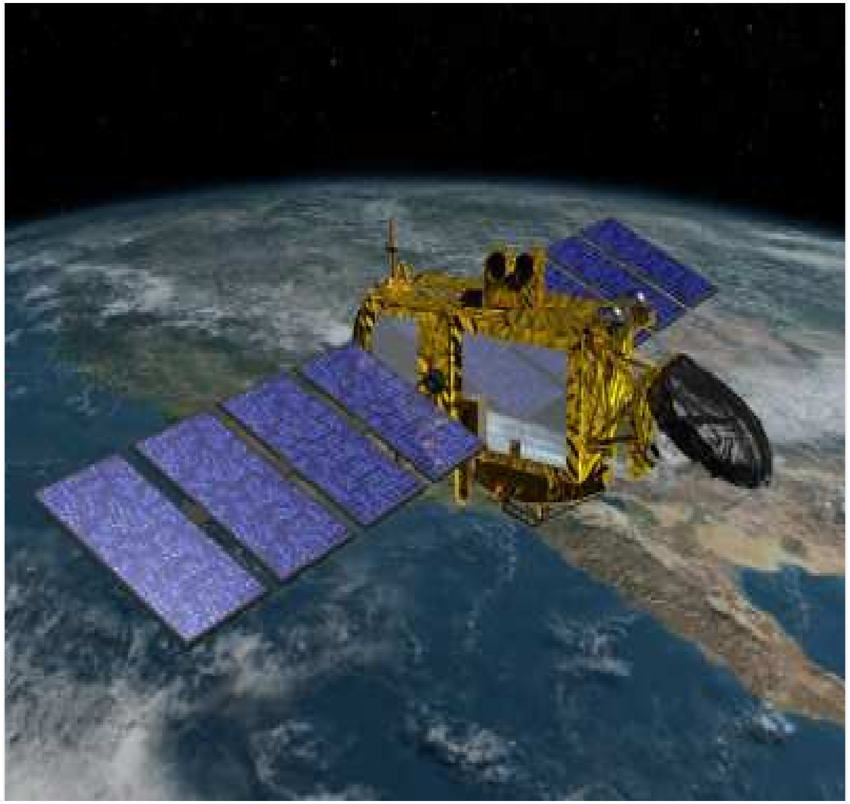
Solar panels are used for electronics and continuous operation.



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Types of rocket engine

- Solid propellant
  - Fuel, oxydator and binding substance
  - Simple construction, reliable, stable
  - Cannot be stopped and restarted
  - Moderate specific impulse, high thrust
- Liquid propellant
  - Monopropellant, or bipropellant (one, or two tanks)
  - Simple construction, reliable, unstable
  - Can be stopped and started
  - Moderate specific impulse, moderate thrust
- Ion propellant
  - Acceleration of positive ions
  - Complicated construction, reliable, stable
  - Can be stopped and restarted
  - High specific impulse, low thrust

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Specific impulse vs. thrust

Specific impulse is defined as thrust over combustion.

It is a measure of the propellant efficiency. Unit in seconds.

$I_{sp} = F/(g_0 \cdot dm/dt)$

F = thrust [N]
g_0 = acceleration of gravity [m/s^2]
dm/dt = propellant mass flow [kg/s]

Definition of total impulse:

$I_{tot} = F \cdot \Delta t = \Delta p$

$I_{sp} = I_{tot} / (\Delta m \cdot g_0)$

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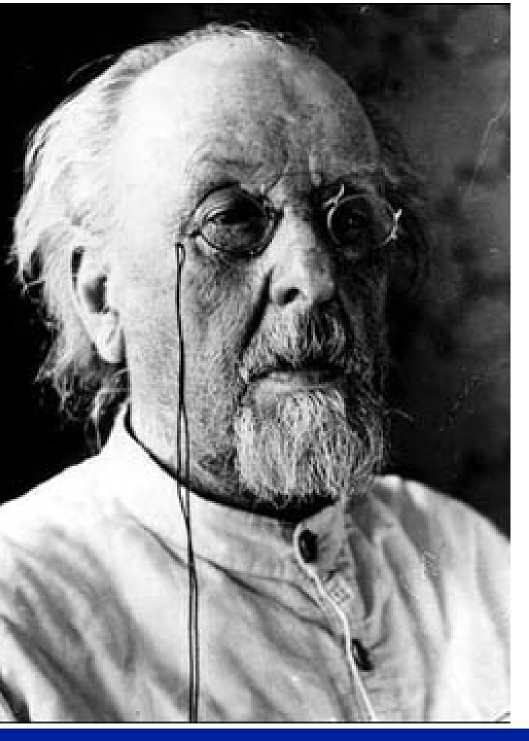
The total impulse vs. the specific impulse, and velocity increment

Total impulse: $I_{tot} = F \cdot \Delta t = v_e \cdot \Delta m = g_0 \cdot I_{sp} \cdot \Delta m$

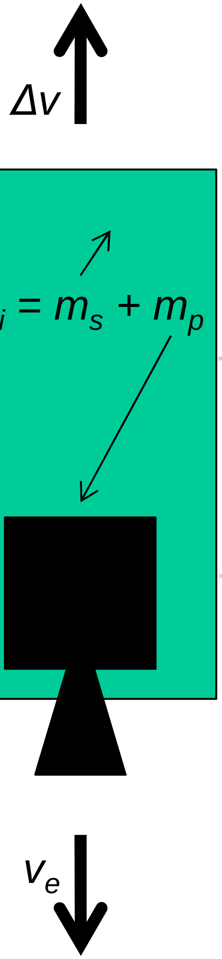
Specific impulse: $I_{sp} = F/(g_0 \cdot \Delta m/\Delta t) = (F \cdot \Delta t)/(g_0 \cdot \Delta m) = v_e /g_0$

v_e = exhaust velocity



The first «rocket scientist»
Konstantin Tsiolkovskij
1857-1935

Constant momentum



Newton's 2nd law: the momentum is constant.

This gives us the Rocket equation: $\Delta v = v_e \cdot ln(m_i/(m_i - m_p))$

m_i = the total initial mass = $m_s + m_p$

m_s = satellite mass (structure + payload)

m_p = propellant mass

The velocity increment when burning a propellant mass $m_p = \Delta m$:

$\Delta v = v_e \cdot ln(m_i/(m_i - m_p)) = v_e \cdot ln(m_i/(m_i - ((F \cdot \Delta t)/(g_0 \cdot I_{sp}))))$

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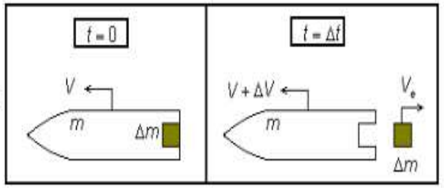
Derivation [edit]

Consider the following system:



In the following derivation, "the rocket" is taken to mean "the rocket and all of its unburned propellant".

Newton's second law of motion relates external forces (F_i) to the change in linear momentum of the whole system (including rocket and exhaust) as follows:

$$\sum F_i = \lim_{\Delta t \to 0} \frac{P_2 - P_1}{\Delta t}$$

where P_1 is the momentum of the rocket at time *t=0*:

$$P_1 = (m + \Delta m)\, V$$

and P_2 is the momentum of the rocket and exhausted mass at time $t = \Delta t$:

$$P_2 = m\,(V + \Delta V) + \Delta m V_e$$

and where, with respect to the observer:

- V is the velocity of the rocket at time *t=0*
- $V + \Delta V$ is the velocity of the rocket at time $t = \Delta t$
- V_e is the velocity of the mass added to the exhaust (and lost by the rocket) during time Δt
- $m + \Delta m$ is the mass of the rocket at time *t=0*
- m is the mass of the rocket at time $t = \Delta t$

The velocity of the exhaust V_e in the observer frame is related to the velocity of the exhaust in the rocket frame v_e by (since exhaust velocity is in the negative direction)

$$V_e = V - v_e$$

Solving yields:

$$P_2 - P_1 = m\Delta V - v_e \Delta m$$

and, using $dm = -\Delta m$, since ejecting a positive Δm results in a decrease in mass,

$$\sum F_i = m\frac{dV}{dt} + v_e\frac{dm}{dt}$$

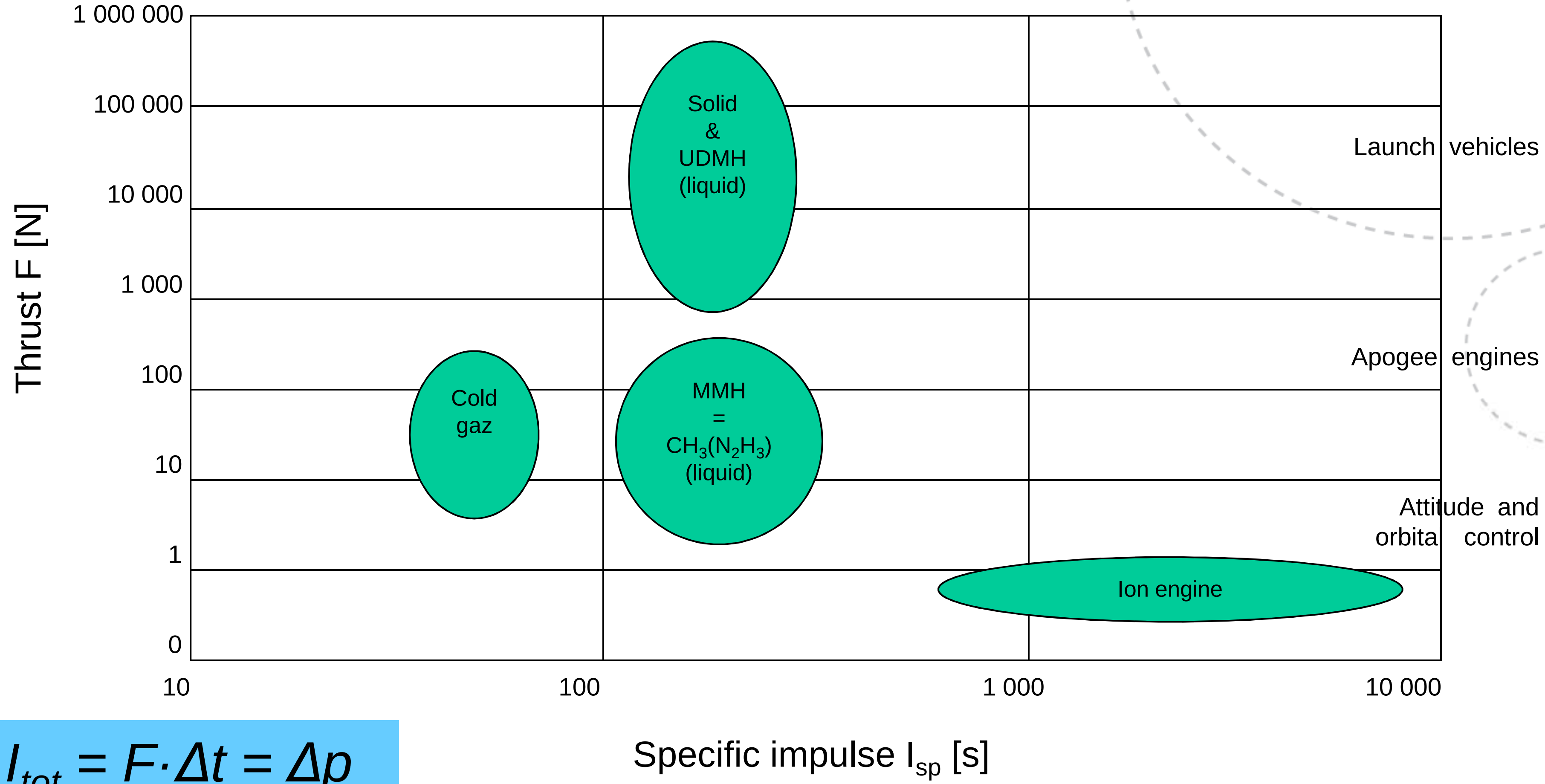
If there are no external forces then $\sum F_i = 0$ (conservation of linear momentum) and

$$m\frac{dV}{dt} = -v_e\frac{dm}{dt}$$

Assuming v_e is constant, this may be integrated to yield:

$$\Delta V\ = v_e \ln \frac{m_0}{m_1}$$

Thrust vs. Specific impulse



$$I_{tot} = F \cdot \Delta t = \Delta p$$

$$I_{sp} = I_{tot} / (\Delta m \cdot g_0)$$

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Liquid propellants

- Hydrazine is an inorganic chemical compound with the formula N_2H_4. It is a colourless flammable liquid with an ammonia-like odor and is derived from the same industrial chemistry processes that manufacture ammonia. However, hydrazine has physical properties that are closer to those of water. Hydrazine is highly toxic and dangerously unstable, and is usually handled while in solution for safety reasons.
- Monomethyl hydrazine (MMH), chemical formula CH_3 (NH)NH_2, where one of the hydrogen atoms on the hydrazine molecule has been replaced with a methyl group (CH_3). This chemical is toxic and carcinogenic in small amounts, but it is easily stored in outer space providing moderate performance for very low fuel tank system weight.
- 1,1-dimethylhydrazine, $C_2H_8N_2$, (unsymmetrical dimethylhydrazine, UDMH) and 1,2-dimethylhydrazine (symmetrical dimethylhydrazine) are hydrazines where two hydrogen atoms are replaced by methyl groups. UDMH is easier to manufacture than symmetrical dimethylhydrazine is, and UDMH is a fairly common rocket fuel. UDMH is a toxic volatile hygroscopic clear liquid, with a sharp, fishy, ammoniacal smell typical for organic amines. It turns yellowish on exposure to air and absorbs oxygen and carbon dioxide. It mixes completely with water, ethanol, and kerosene. In concentration between 2.5% and 95% in air, its vapors are flammable. It is not sensitive to shock.

http://en.wikipedia.org/wiki/Hydrazine

10N Hydrazine Thruster Model CHT 10 (Astrium)

Characteristics:

Propellant:	Hydrazine
Thrust range:	3-10N
I_{sp} vacuum:	220-230s
Total impulse:	517 000 Ns
Accum. burn time:	5 hours
Overall length:	142 mm
Nozzel diameter:	19 mm
Mass:	0.24 kg

Meteosat: 25 units

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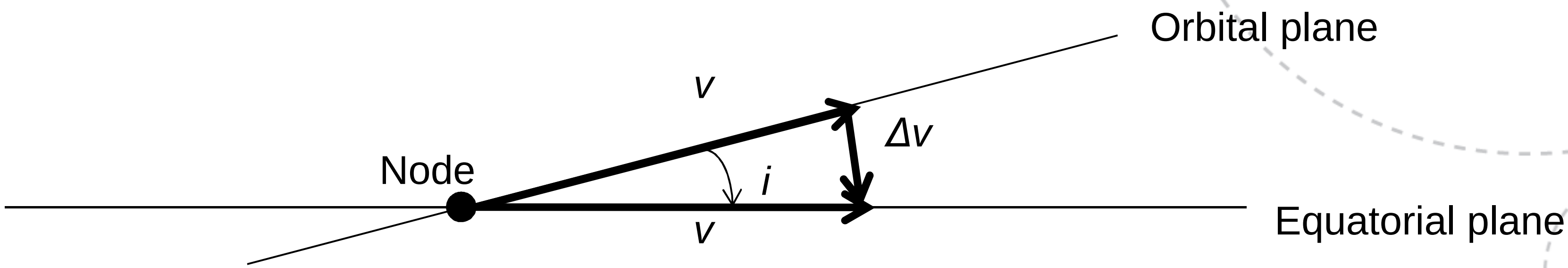
http://cs.astrium.eads.net/sp/spacecraft-propulsion/hydrazine-thrusters/heritage-thrusters.html

Example: correction of inclination drift



Given figures:

$m_s = m_f = 800kg$ $I_{sp} = 230s$ $v = 3.075km/s$ $i = 0.05°$

Needed correction speed: $\Delta v = v \cdot 2 \cdot \sin(i/2) = 3.075 \cdot 10^3 \cdot 2 \cdot \sin(0.025°) = 2.7m/s$

Propellant mass needed for correction: $m_p = m_s \cdot (e^{\Delta v/(I_{sp} \cdot g_0}-1) \sim 1kg$

The drift is *0.8°* per year, and *1kg* is needed to correct *0.05°*. To correct inclination drift *16kg* of propellant is needed per year.

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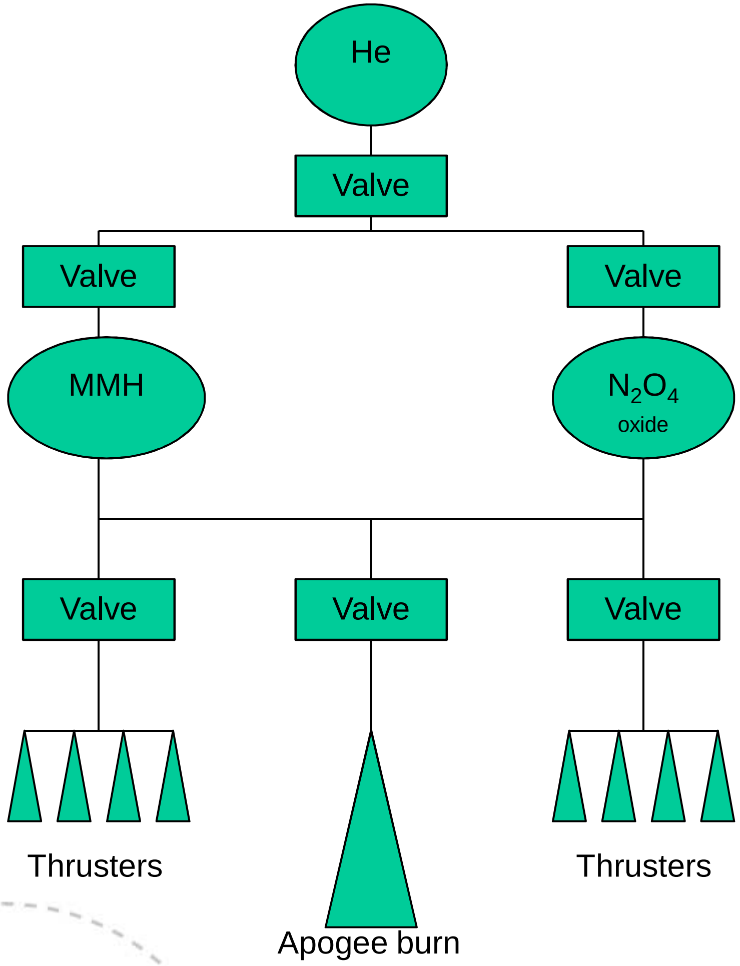
Propulsion system with liquid propellant (Bipropellant)



Unified propulsion system (UPS)

- The same fuel source for apogee burn (transfer orbit, TO) and for station keeping
- Helium used to drive the propellants
- Redundant valves and fuel tanks
- Typically 10 N thrusters for on-orbit operations
- 500 N thrusters for apogee burn
- GTO to GEO via repeated apogee burns saves propellant
- Precise orbiting increases lifetime

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Ion thrusters

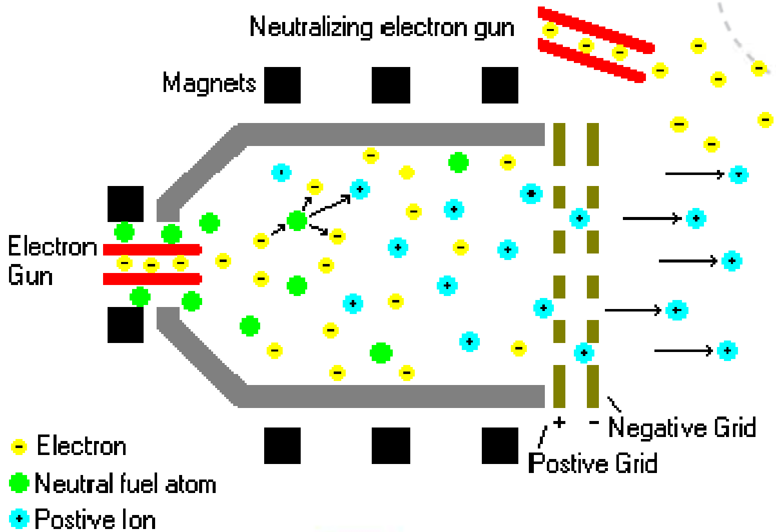
Propellant atoms are injected into the discharge chamber and get ionized by electron bombardment.

The positively charged ions move due to diffusion, and are accelerated by the potential difference between the first (screen) and the second (accelerator) grid of the extraction system.

Electrons are emitted from a separate cathode placed near the ion beam to prevent the spacecraft from gaining a net negative charge.

The negative voltage of the accelerator grid prevents electrons outside the thruster from streaming back to the discharge chamber.

Mercury or cesium atoms were used as propellants during tests in the 1960s and 1970s, but these propellants adhered to, and eroded the grids.
And they are toxic. Xenon atoms are used currently, as Xenon is a safe, inert gaz, possible to store as dense gaz at relatively low pressure.



Performance: Up to I_{sp}= 10000s

Ion thrusters

Propellant atoms are injected into the discharge chamber and get ionized by electron bombardment forming a plasma.

- The electrons are emitted from a hollow cathode and are accelerated on their way to the anode.
- The electrons can be accelerated by the oscillating electric field.
- Microwave heating.

The positively charged ions move towards the extraction system of the chamber due to diffusion, and are accelerated by the potential difference between the first (screen) and the second (accelerator) grid of the extraction system. The final ion energy is determined by the potential of the plasma (the plasma potential is a few volts larger than the screen grid voltage).

The negative voltage of the accelerator grid prevents electrons of the beam plasma outside the thruster from streaming back to the discharge plasma.

Electrons are emitted from a separate cathode placed near the ion beam to prevent the spacecraft from gaining a net negative charge.

Mercury or cesium atoms were used as propellants during tests in the 1960s and 1970s, but these propellants adhered to, and eroded the grids.

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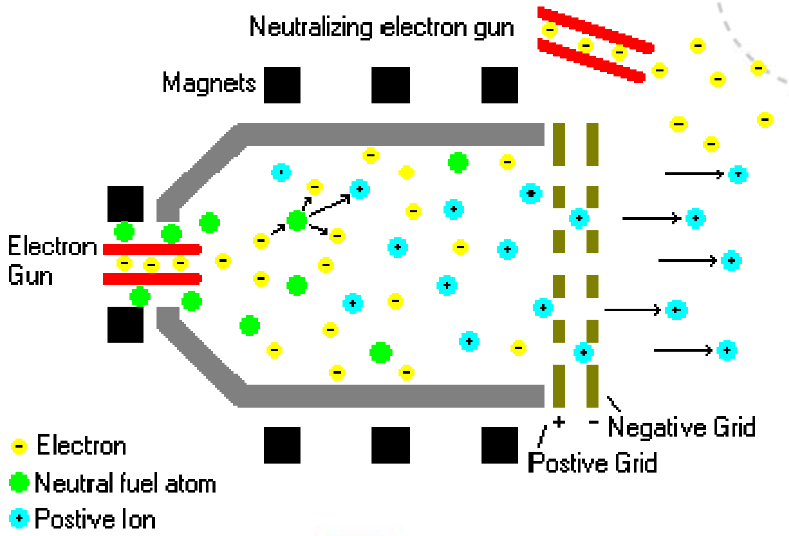

Performance: Up to I_{sp}= 10000s

Solar cells

The main power supply for the electrical energy needed for all subsystems on-board is solar cells and batteries.

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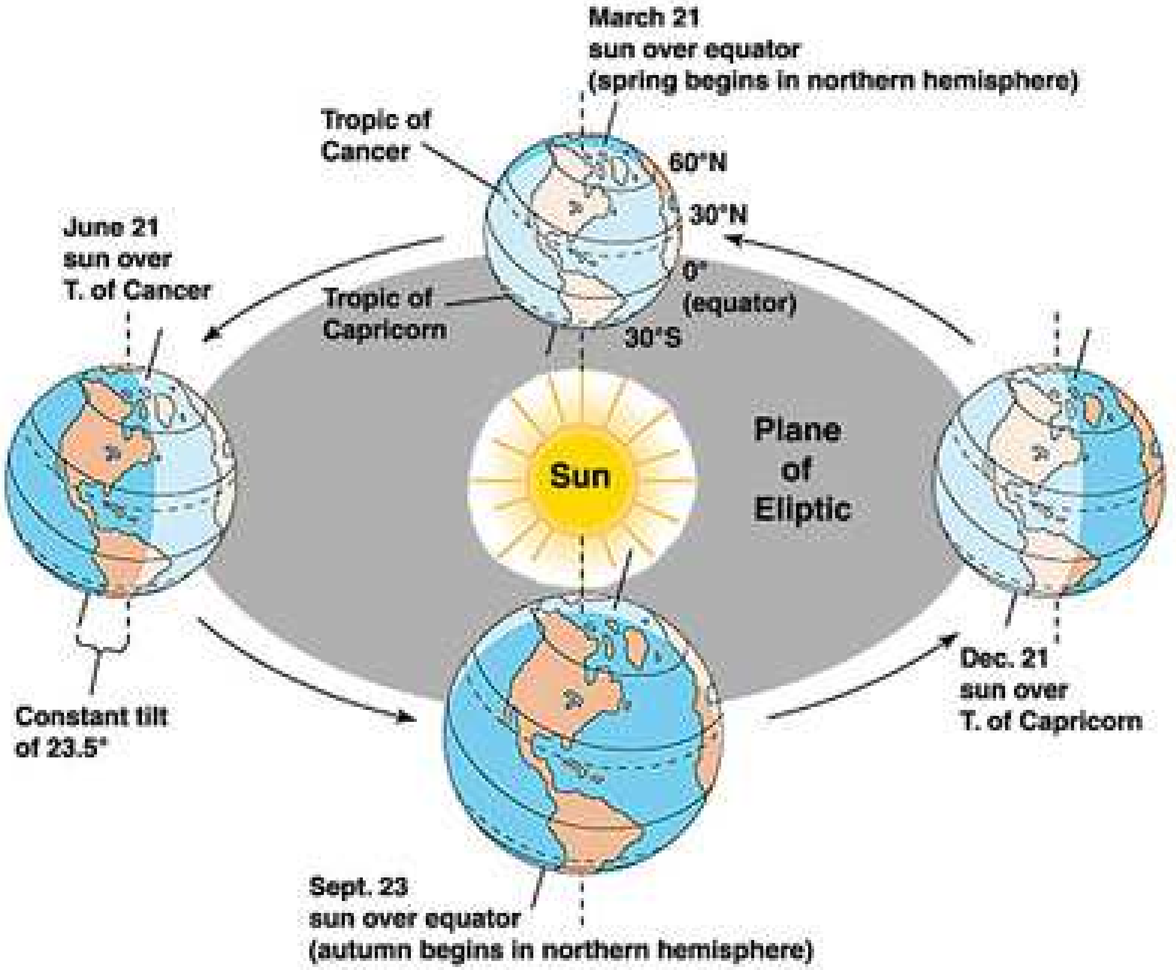
Needs on-board a satellite

Solar panels are needed to create energy in space. The energy is used to run components on-board, and to charge batteries for use during eclipses (secondary batteries).

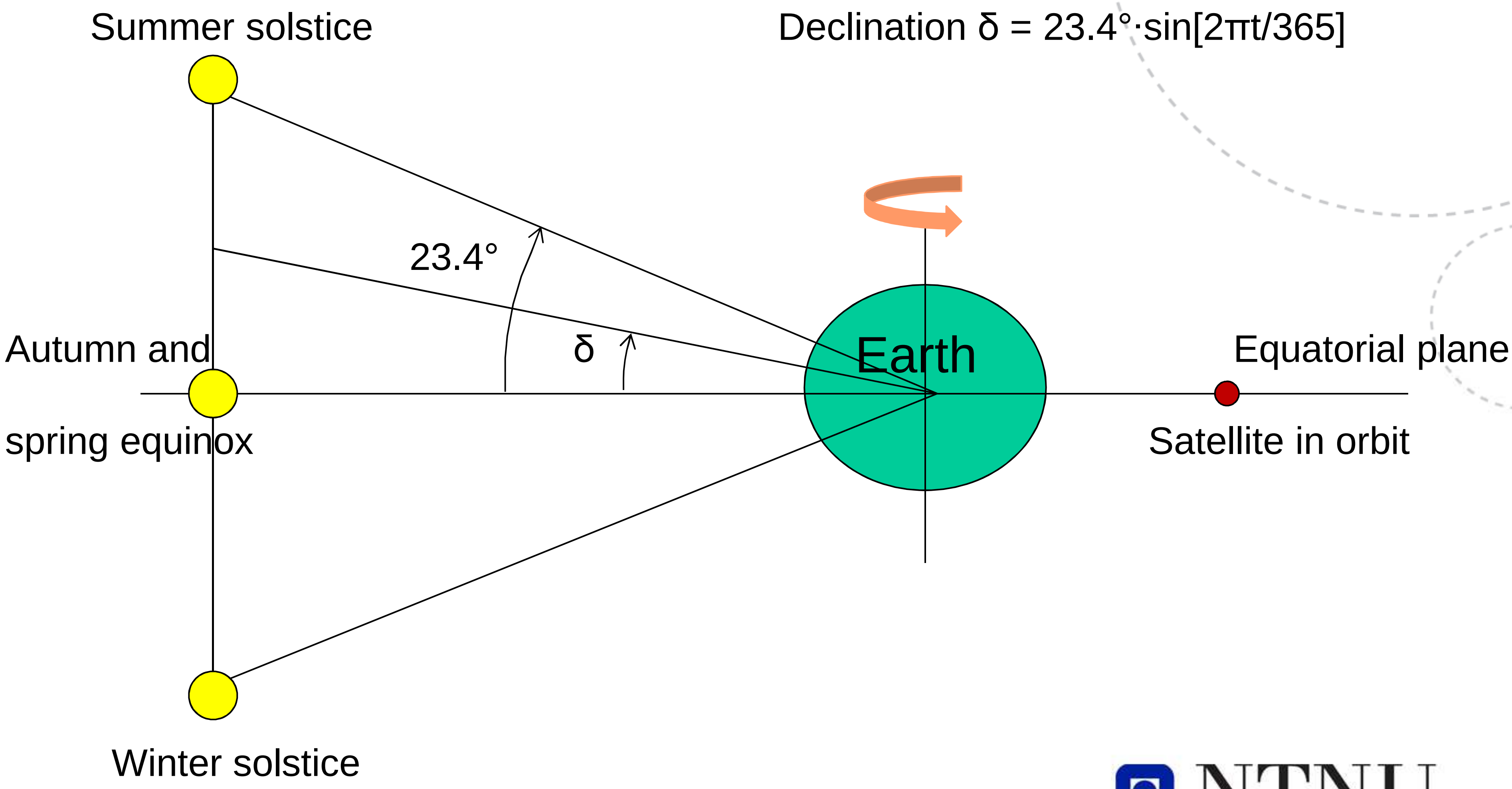
Other possibilities are:

- Use of primary batteries (no charging in space) (short missions)
- Fuel cells (short missions)
- Nuclear power (very long/distant missions)

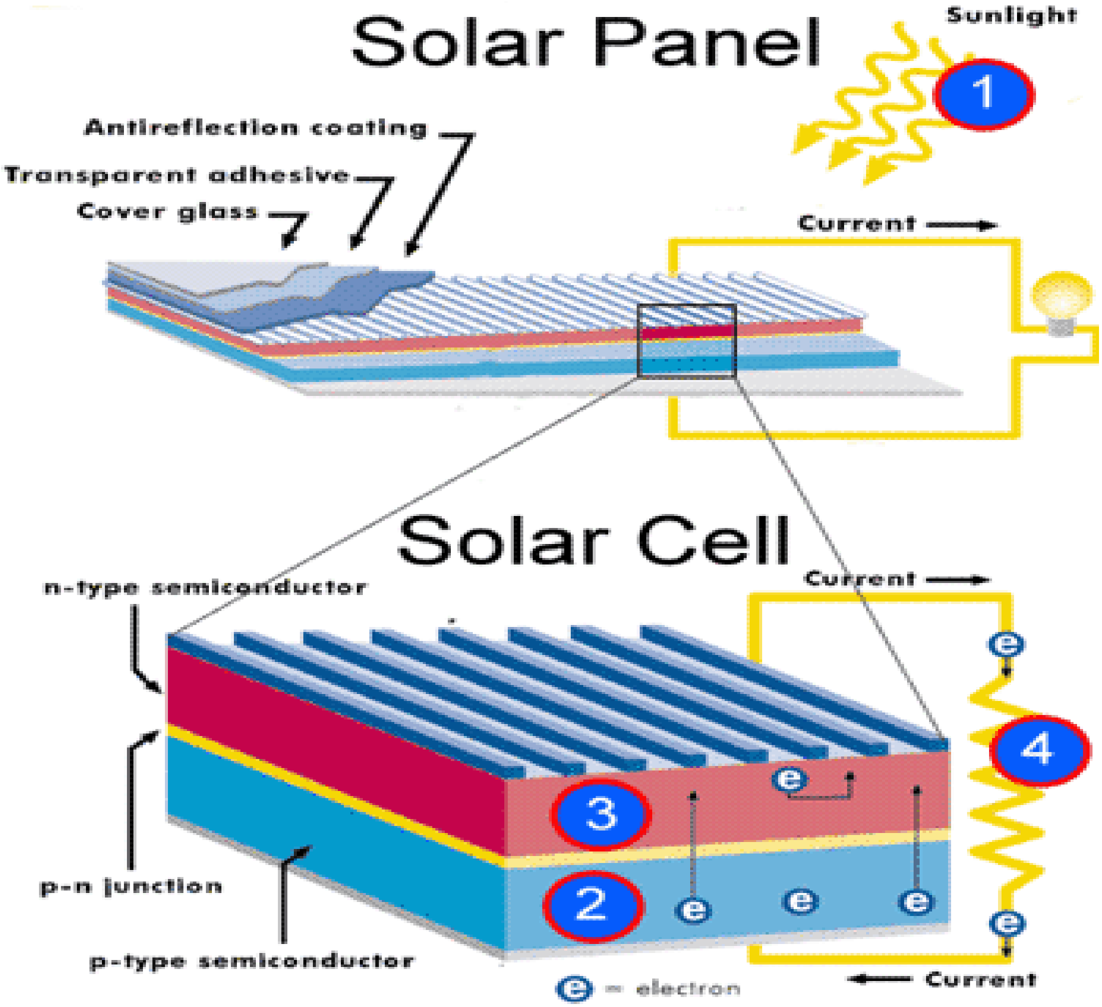
Earth orbiting the sun



The apparent motion of the sun seen from earth



Solar panel cell power production



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Output power vs. inclination

The output power of a solar cell depends on the solar incidence angle:

$$P \propto \varphi_{sol} \cdot \cos(\delta) \cdot A$$

φ_{sol} = solar power flux density at the solar panel.

Average value $\varphi_{sol\ av} \approx 1360 W/m^2$

δ = sun angle

For a GEO satellite δ=0° at autumn and spring equinoxes, δ=23.4° at summer solstice and -23.4° at winter solstice.

A = solar panel area.

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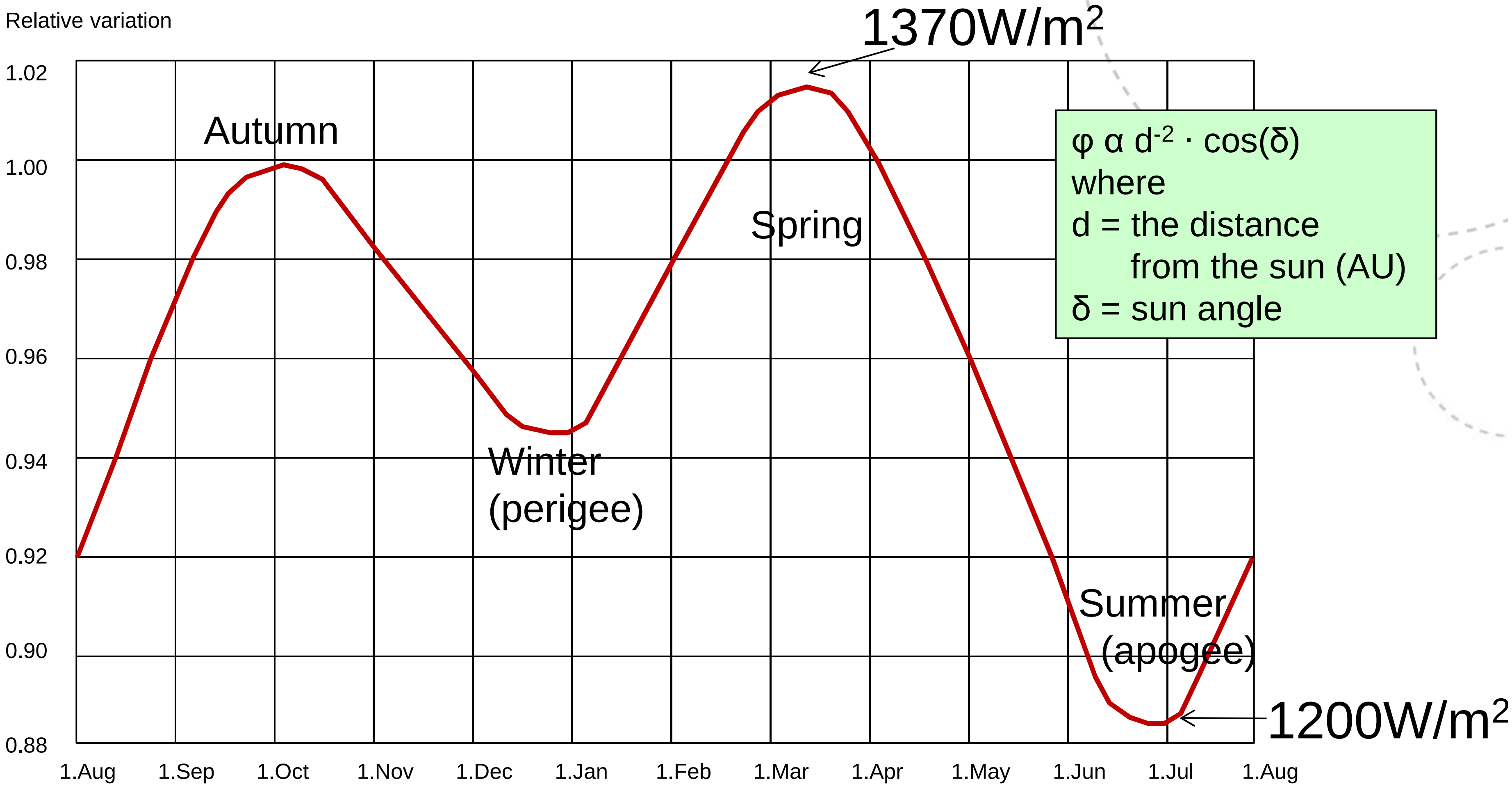
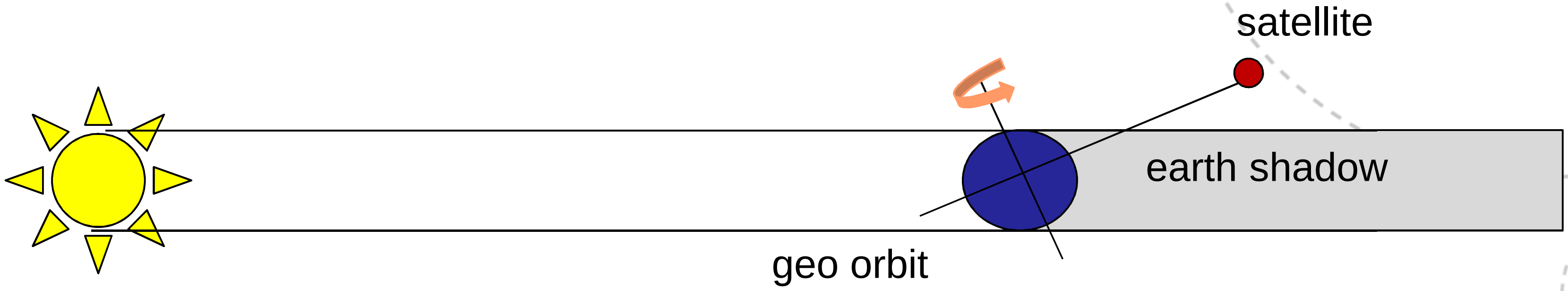
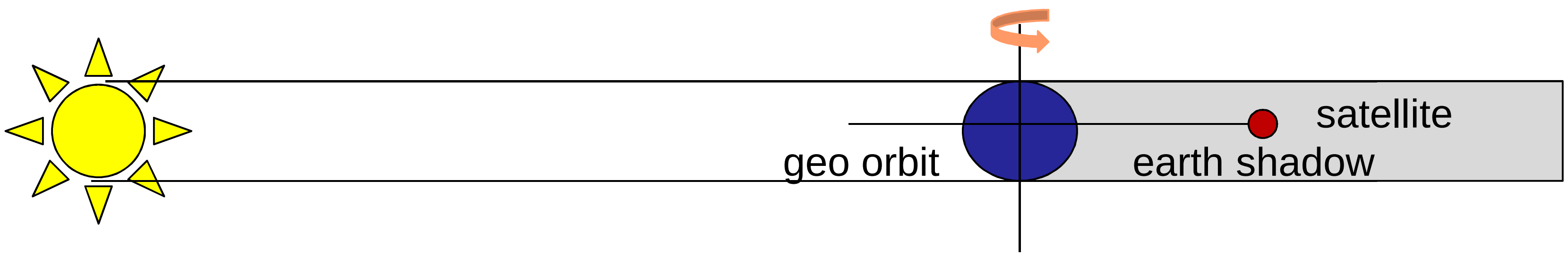
Typical (measured) variations in available solar flux over a year



Sun eclipse provoked by the earth



The satellite will be in sunlight through the entire winter and summer



The satellite will pass through the earth shadow once per day around autumn and spring equinox

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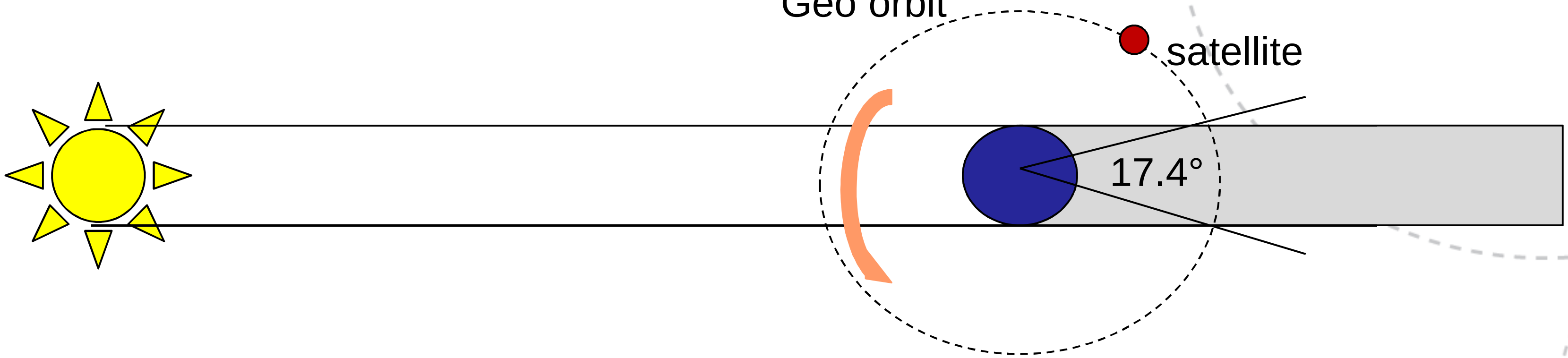
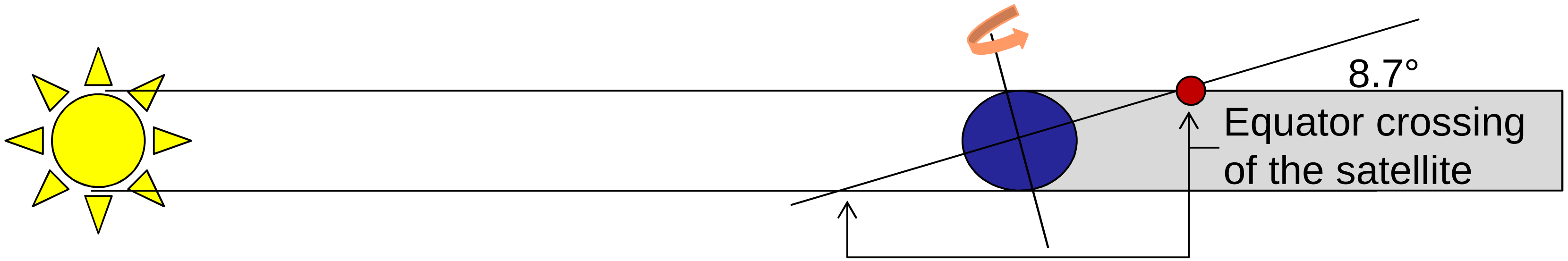
Time of an eclipse



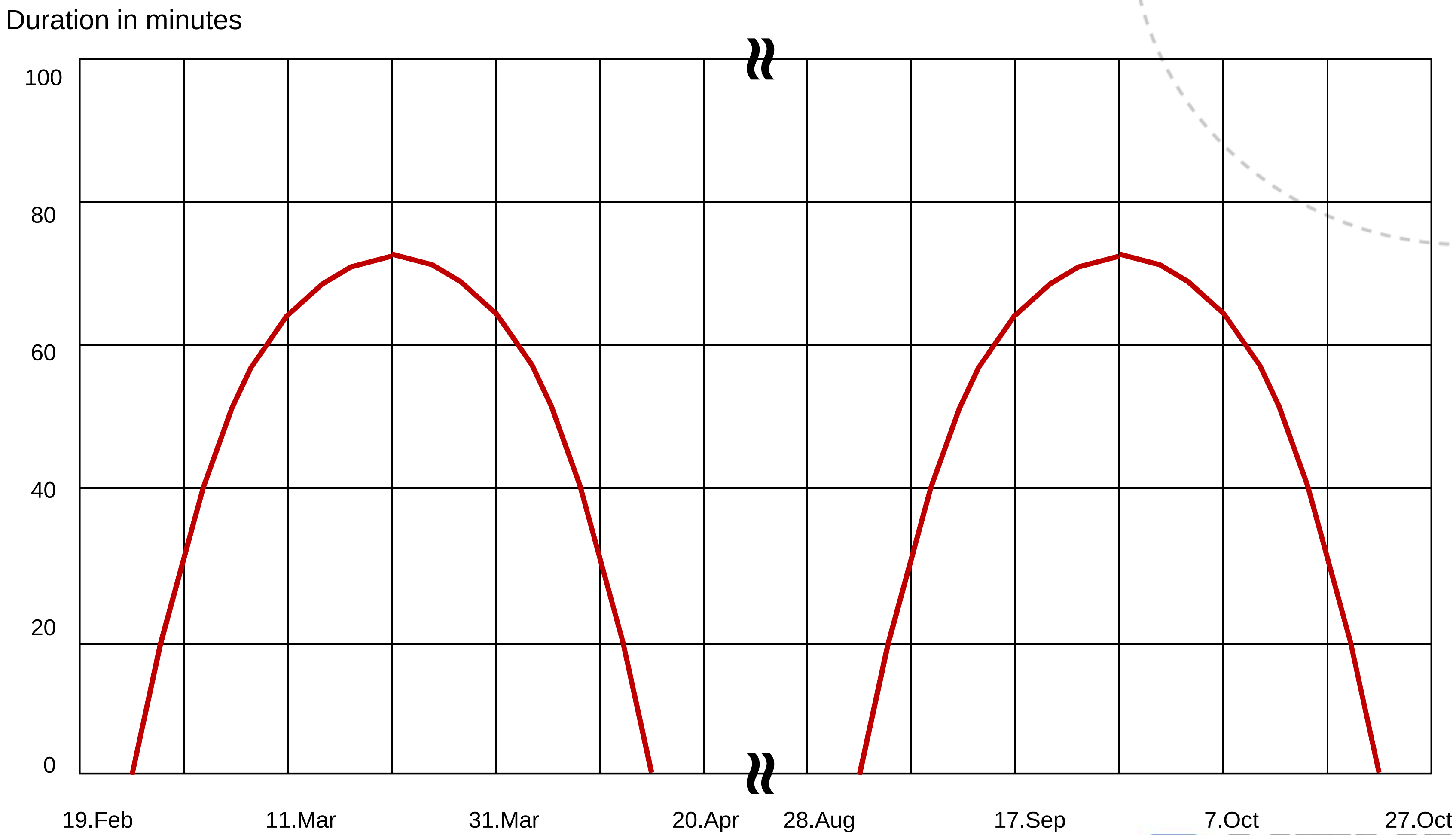
Maximum duration of an eclipse: t_{max} = (17.4°/360°)·24h·60min = 70min



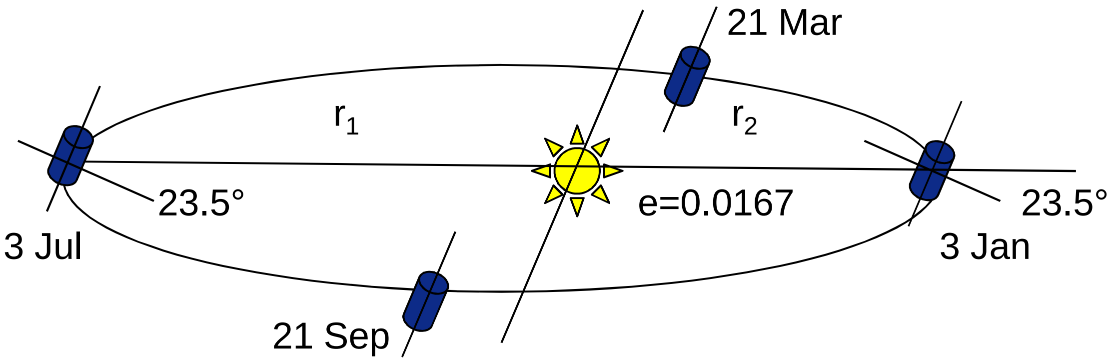
At the first day of an eclipse: δ ≈ 8.7°

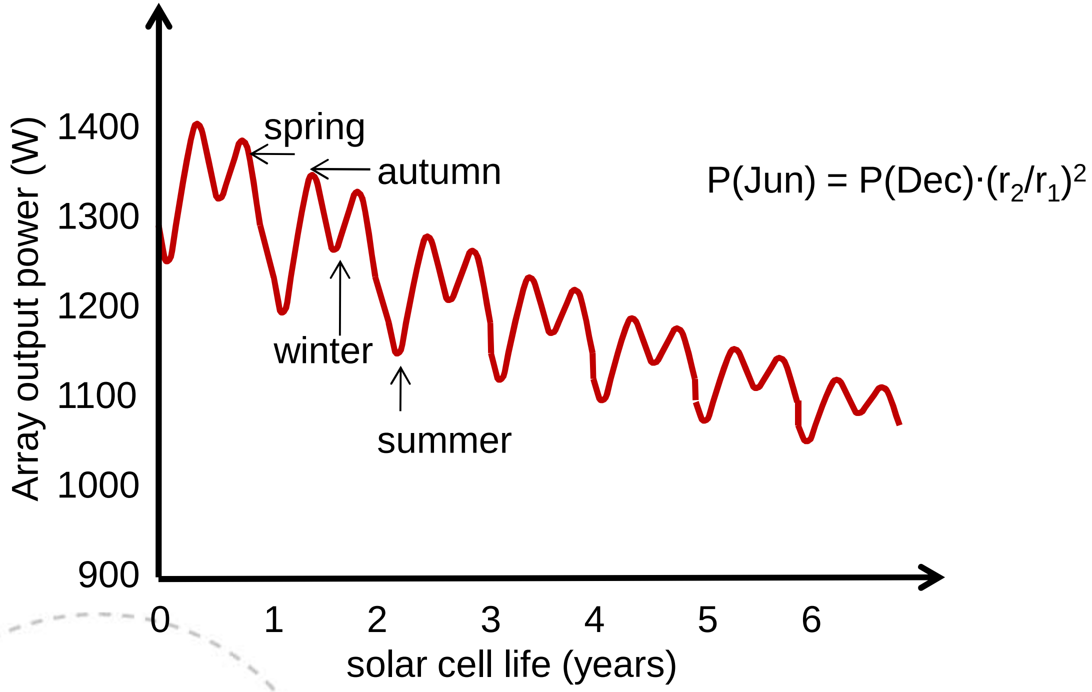
δ = 23.4°·sin[2π·Δt / (365 days)]
=> The time period of daily entering the eclipse 2·Δt ≈ 42 days

Daily duration of eclipse vs. date



Variation in solar cell output





Solar cells degrade due to damage caused by high energy particles and UV radiation.

The solar cell array degradation factor is approximately:

$F_s \approx 0.7 + 0.3\, e^{-t/1000}$

where t is the number of days from launch.

DoD, Depth of Discharge

DoD is the ratio (%) of the used electric charge (Ah) and the battery's total stored electric charge (Ah).

$$DoD = 100 \cdot W_{used}/W_{total\ capacity}$$

The smaller DoD, the more times a battery can be cycled before it dies.

E.g. a NiCd battery can be charged 20000 times at 25% DoD, but only 800 times at 75% DoD.

In the end it will not hold charge, and die.

LEO satellites need more battery than GEO satellites due to more frequent eclipses.

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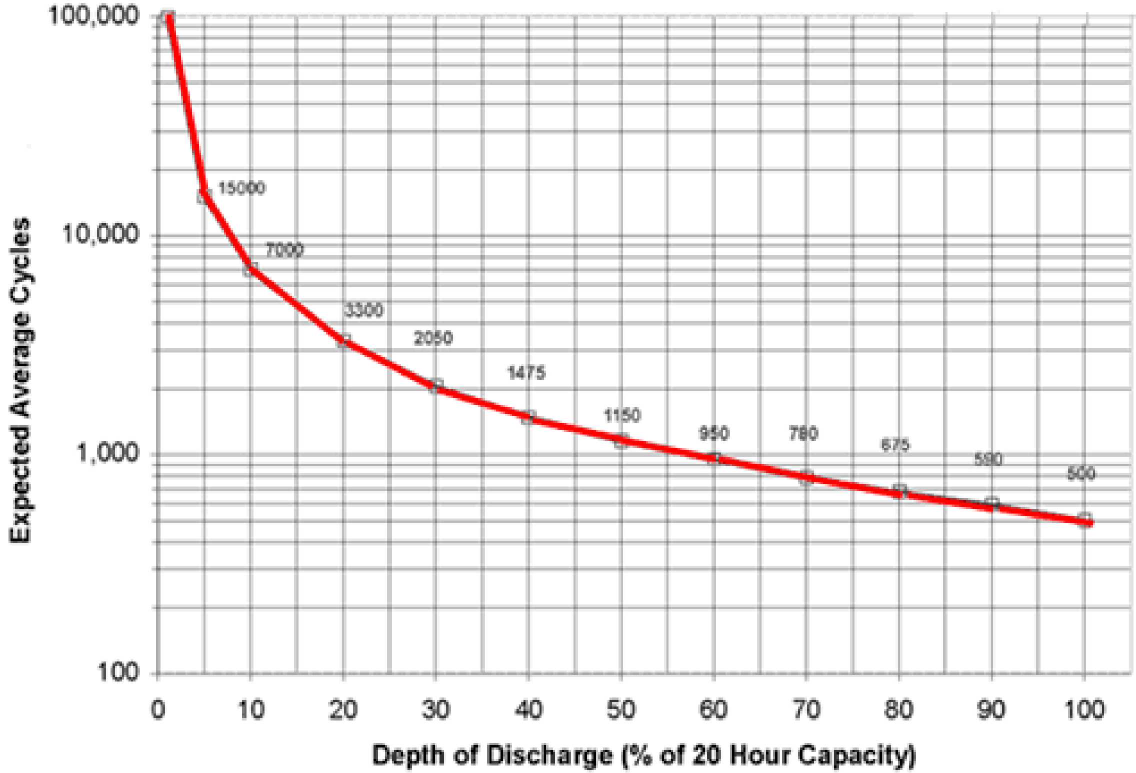
Depth of Discharge vs. Cycle Life



The above graph was constructed for a Lead acid battery, but with different scaling factors, it is typical for all cell chemistries including Lithium-ion. This is because battery life depends on the **total energy throughput** that the active chemicals can tolerate.
Ignoring other ageing effects, the total energy throughput is fixed so that one cycle of 100% DOD is roughly equivalent to 2 cycles at 50% DOD and
10 cycles at 10% DOD and 100 cycles at 1% DOD.

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http://www.mpoweruk.com/life.htm

Power supply requirements

Example: small TV satellite

1) No. of power amplifiers (TWTA): 15
   Output power: 40W — 15•40/0.6= 1kW
   Efficiency: 60%

2) "Housekeeping" (TT&C, heaters, charging etc.): 0.5kW

Total power requirement from solar panels or batteries: 1.5kW

The satellite shall last for 12 years.

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Dimensioning of the power supply

3) Battery 16 cells x 1.25V = 20V

DoD requirement: less than 70% with 1 failed cell

i.e. battery with 15 cells x 1.25V = 18.75V

Duration of eclipse: = 70 min

4) $I = P_{tot}/V_{battery} = 1500W/18.75V = 80A$

$W = I \cdot t_{eclipse} = 80A \cdot t_{eclipse} = 93.33Ah$

$DoD = W/W_{battery\ total} = 0.7$

$\Rightarrow W_{battery\ total} = W/DoD = 93.33/0.7 = 133.3Ah$

=> Chosen battery: <u>141Ah</u> (standard size)

5) The performance of the solar panels shall match 105% at EOL:

$1.5kW \cdot 1.05 \approx 1.6kW$

BOL: $1.6kW/F_s = 1.6/(0.7+0.3e^{-4380/1000}) = \underline{2.3kW}$

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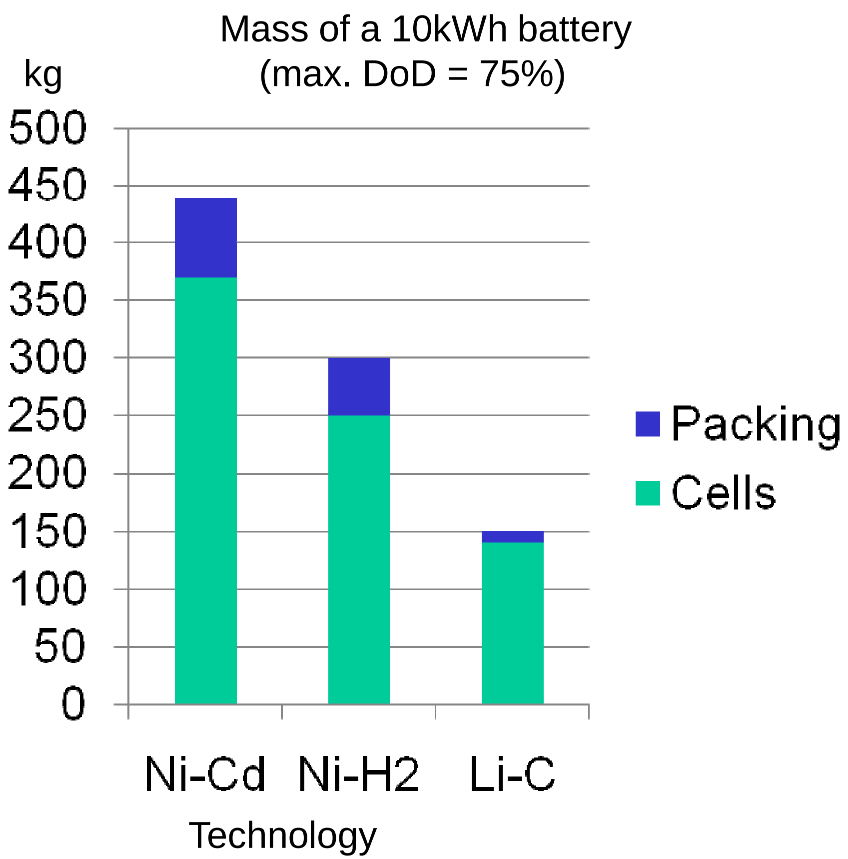
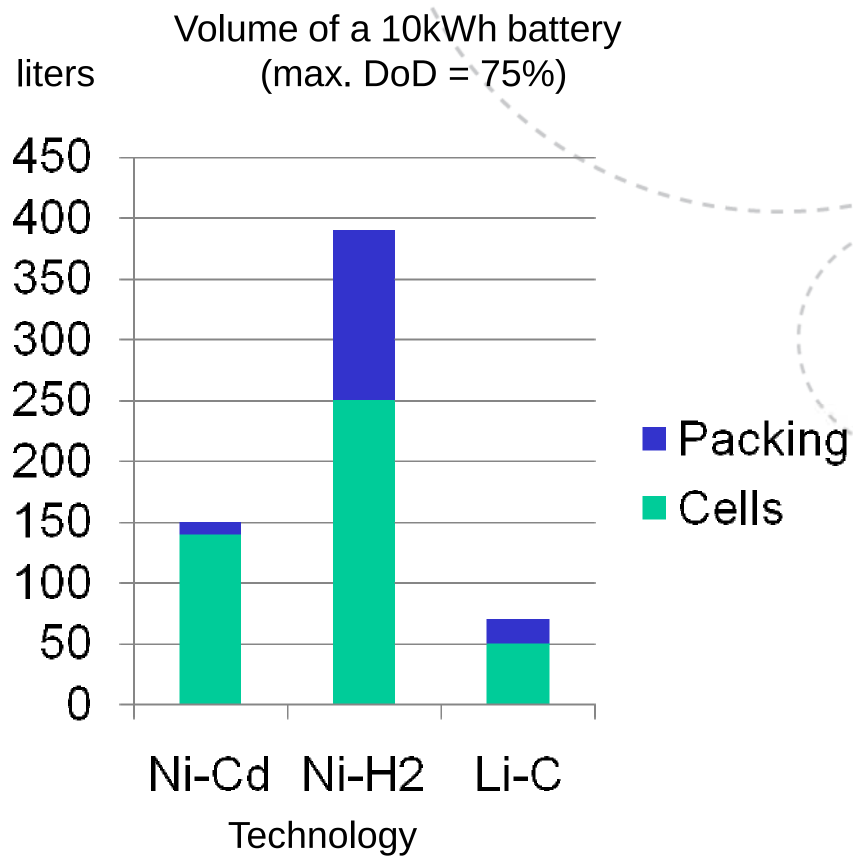
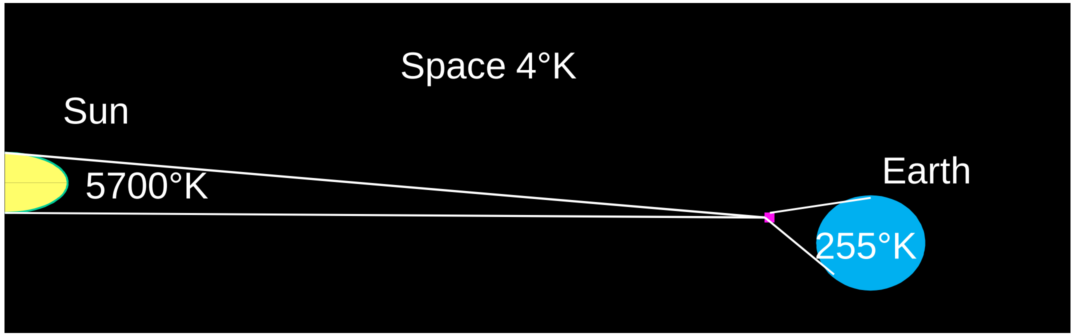
Mass and volume of different battery technologies





4. Temperature control



Operating temperature ranges:

Batteries:	0 to 20°C
Solar panels:	-100 to 50°C
Electronics:	-10 to 60°C
Fuel:	10 to 50°C
IR detectors:	-190 to -150°C
TWT:	-10 to 40°C

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Thermal sub-systems

- Passive protection (blankets, heatpipes)
- Active components (heaters)

Purpose: Keep a stable temperature in a hostile environment with temperatures ranging from about -200°C to +150°C.

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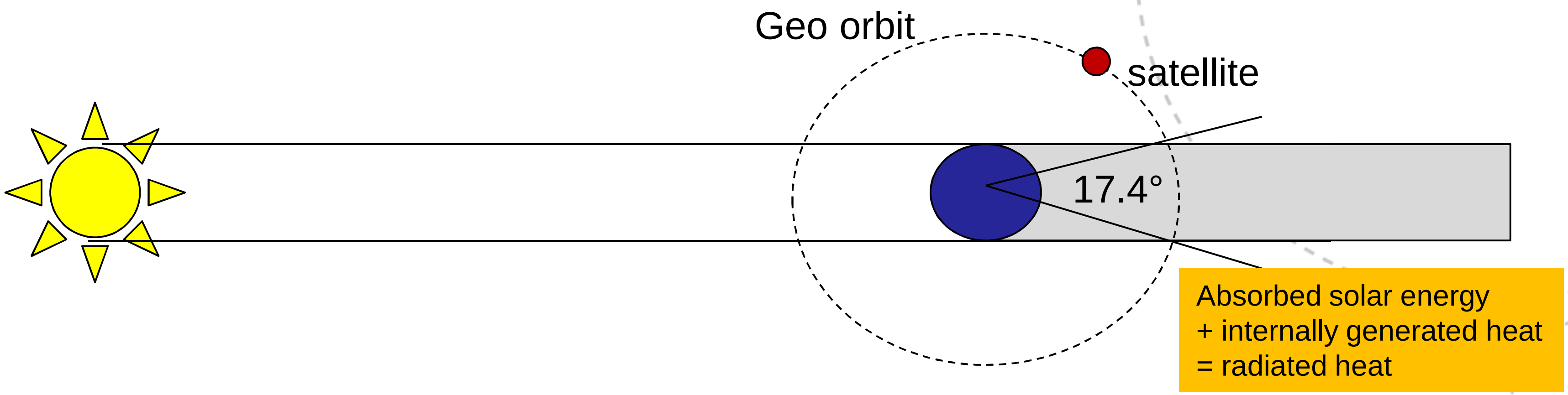
Thermal control principles



- Heat pipes, based on evaporation and condensation of a fluid
- Blinds and curtains, used to change the effective areas of radiating and absorbing surfaces
- Electrical heating elements, to maintain a minimum temperature, controlled by thermostat or by remote control

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Measures to reduce thermal effect impact in a satellite/space craft

Absorbed thermal power: $P_{abs} = S \cdot \alpha \cdot A_a$

S = solar constant ≈1355W/m^2,

α = absorption coefficient,

A_a= the area of the absorbing surface

Radiated termal power: $P_{rad} = \varepsilon \cdot A_r \cdot \sigma \cdot T^4$

ε = emission coefficient,

σ = Stefan-Botzmann constant = $5.67 \cdot 10^{-8}$W/m^{2}K^4,

T = temperature

A_r= the area of the radiating surface

Thermal equilibrium when $P_{abs} = P_{rad,}$ with no internal heat source.

$$\boxed{T = \left(\frac{\alpha A_a S}{\varepsilon A_r \sigma}\right)^{1/4}}$$

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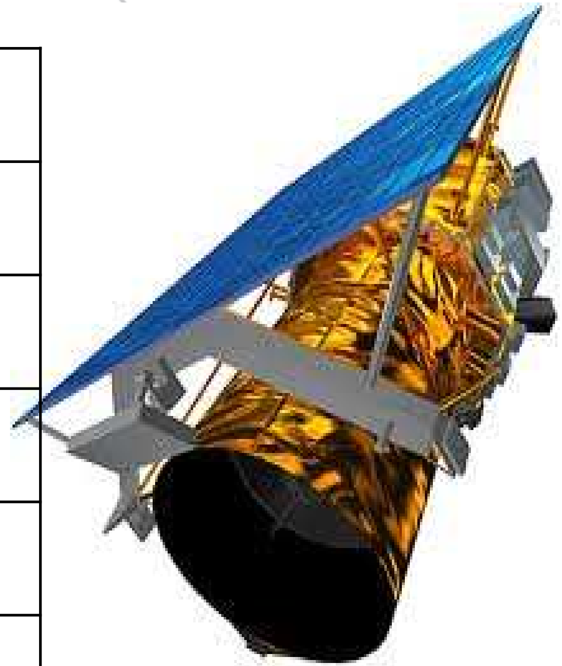
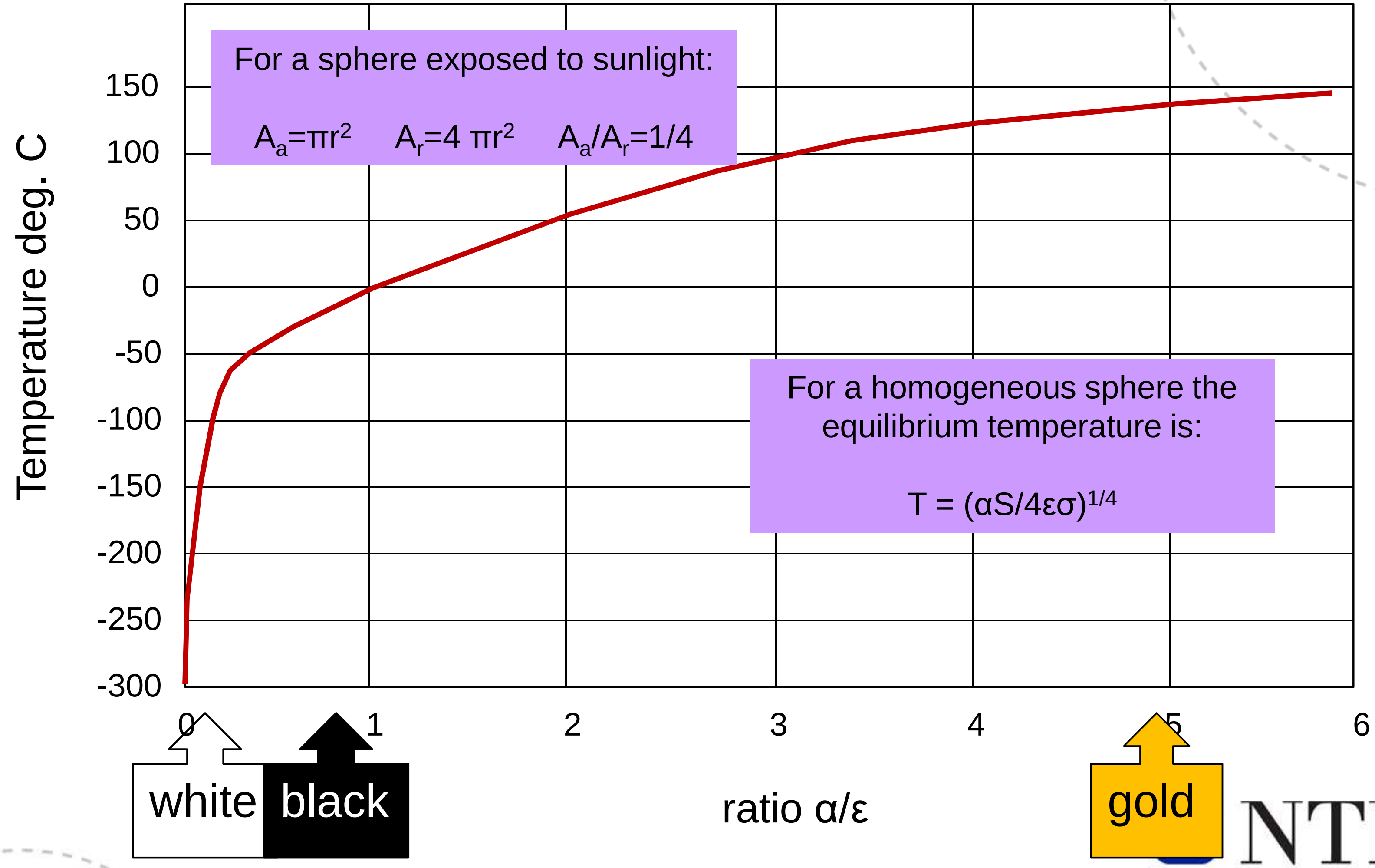
Examples of coating material coefficients

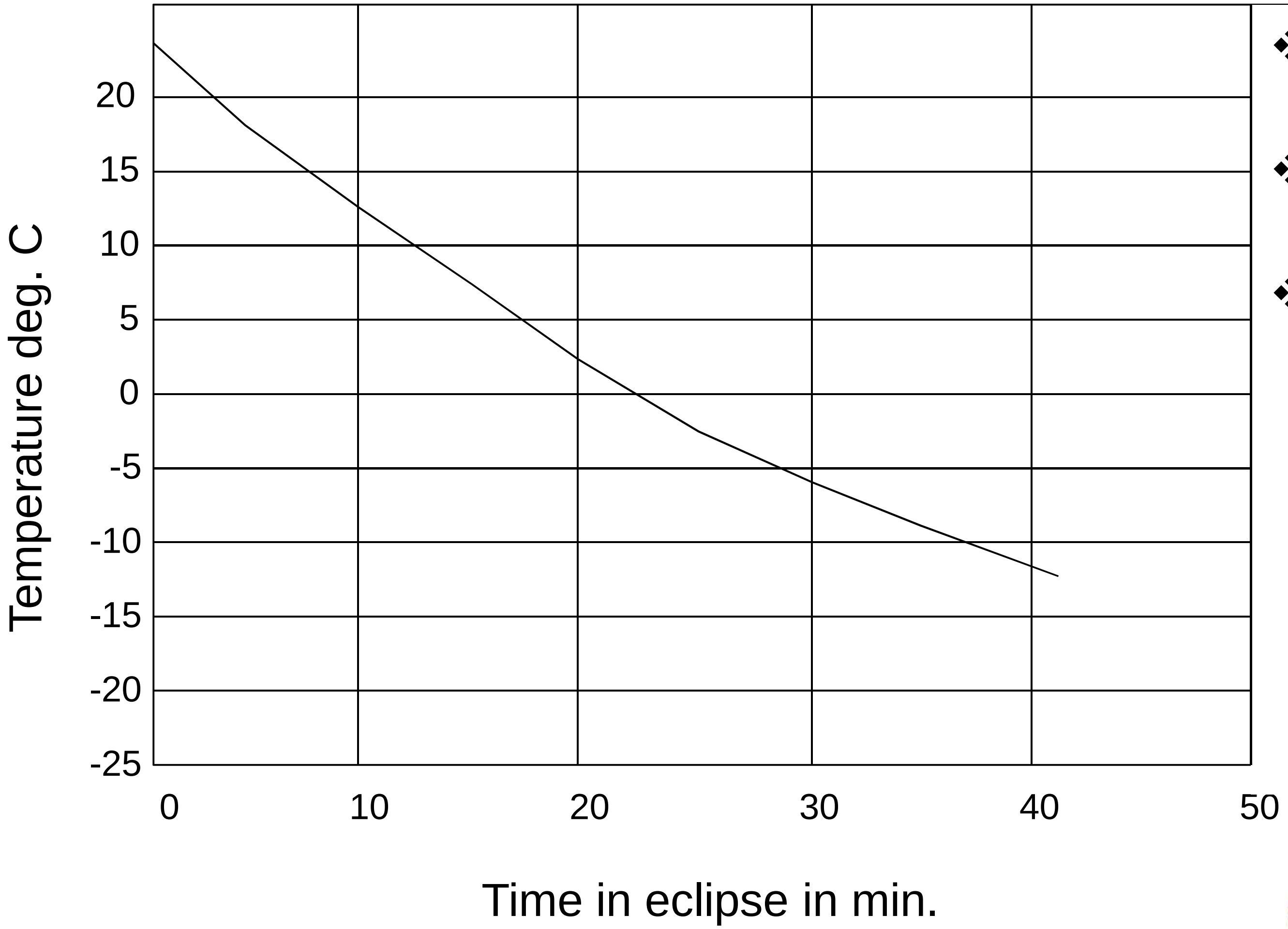
Coating material	α	ε	α/ε
White paint	0.20	0.90	0.22
Black paint	0.95	1.05	0.90
Gold	0.25	0.05	5.00
Optical sun reflector	0.15	0.10	1.50
Solar panels	0.80	0.90	0.89



Equilibrium temperature of a sphere

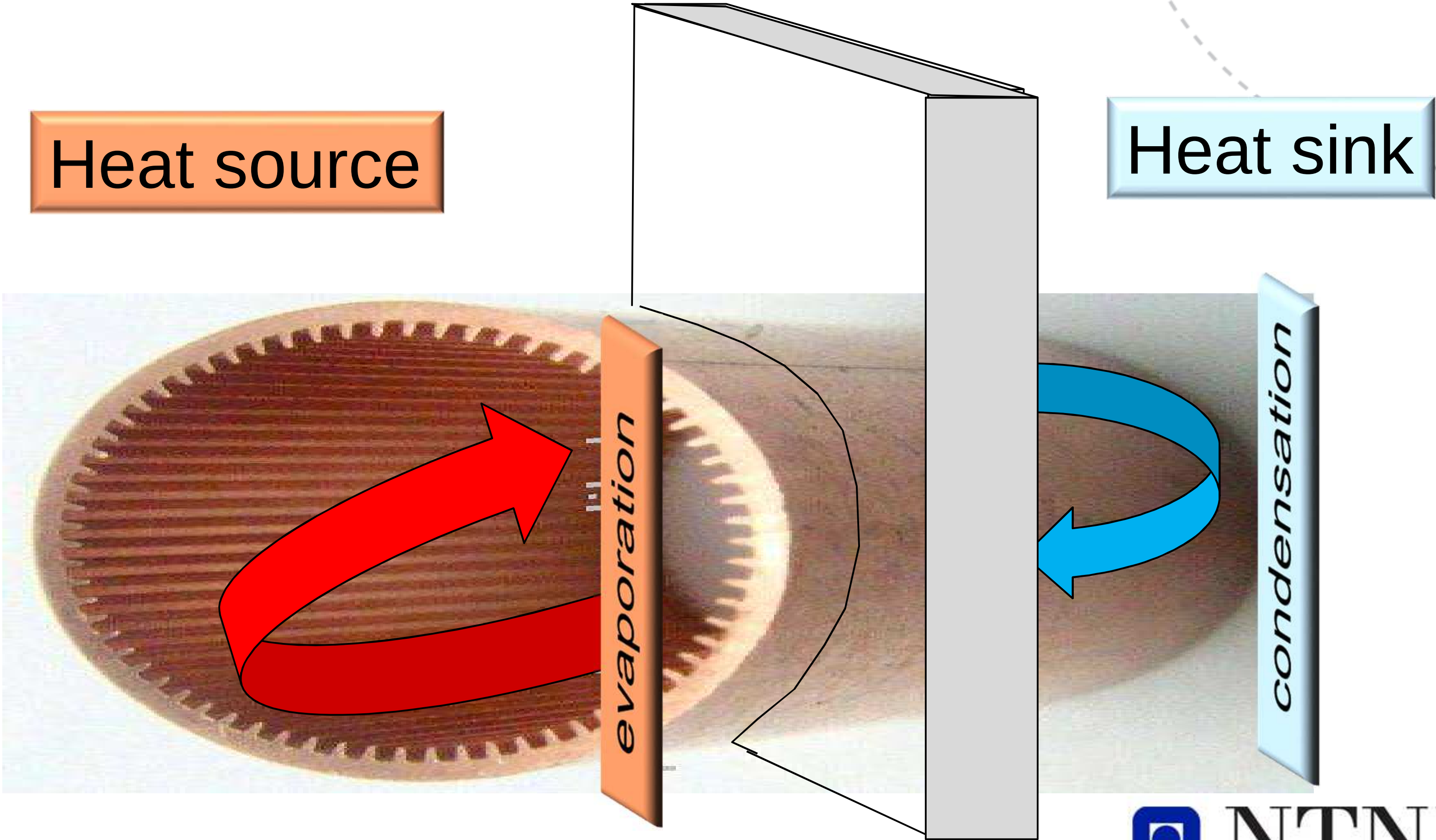


Example (CubeSat)



- 4W Internal heat source
- 40 minutes in eclipse
- Equilibrium at 290K ≈ 20°C

Heat pipes



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Earth station solar interference problems

5. Telemetry

The communication channel between the control centre on Earth and the satellite.

A satellite operator needs to monitor and operate all the control systems; the solar panels, the temperature, the orbital corrections, the deployment and orientation of antennas, control of transmit power, and maneuvers to avoid impacts from debris collisions and sun storms.

The application itself may also need the telemetry link, e.g. for upload of new software, correction of bugs, traffic monitoring and control (OBP).

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Telemetry, Tracking and Command (TT&C)

- Telemetry — measuring of the satellite's physical parameters
- Tracking — orbital information
- Command — correction of orbit/attitude, switching of antennas, transponders etc.

Purpose: It is the lifeline between earth and satellite

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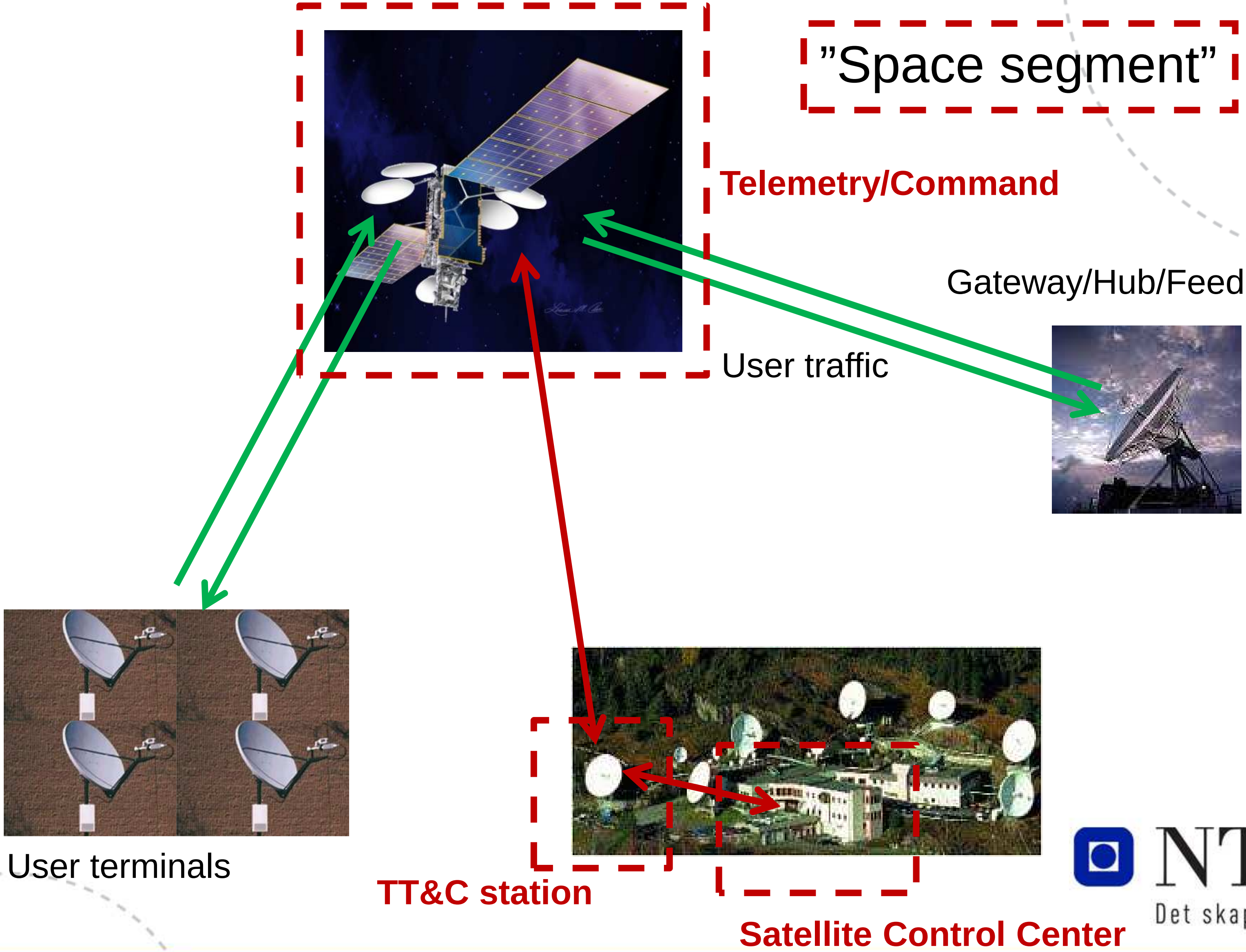
Components in a satellite communication system



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Satellite control

- Satellite Control Center (SCC)
  - Process telemetry data from the satellite
  - Calculate orbital data from satellite position
  - Plan and execute commands, like orbital corrections
  - Control and supervise the TT&C stations
- Telemetry, Tracking and Command station (TT&C)
  - Transmit the commands to the satellite
  - Receive telemetry data from the satellite
  - Measure the distance and angles to the satellite

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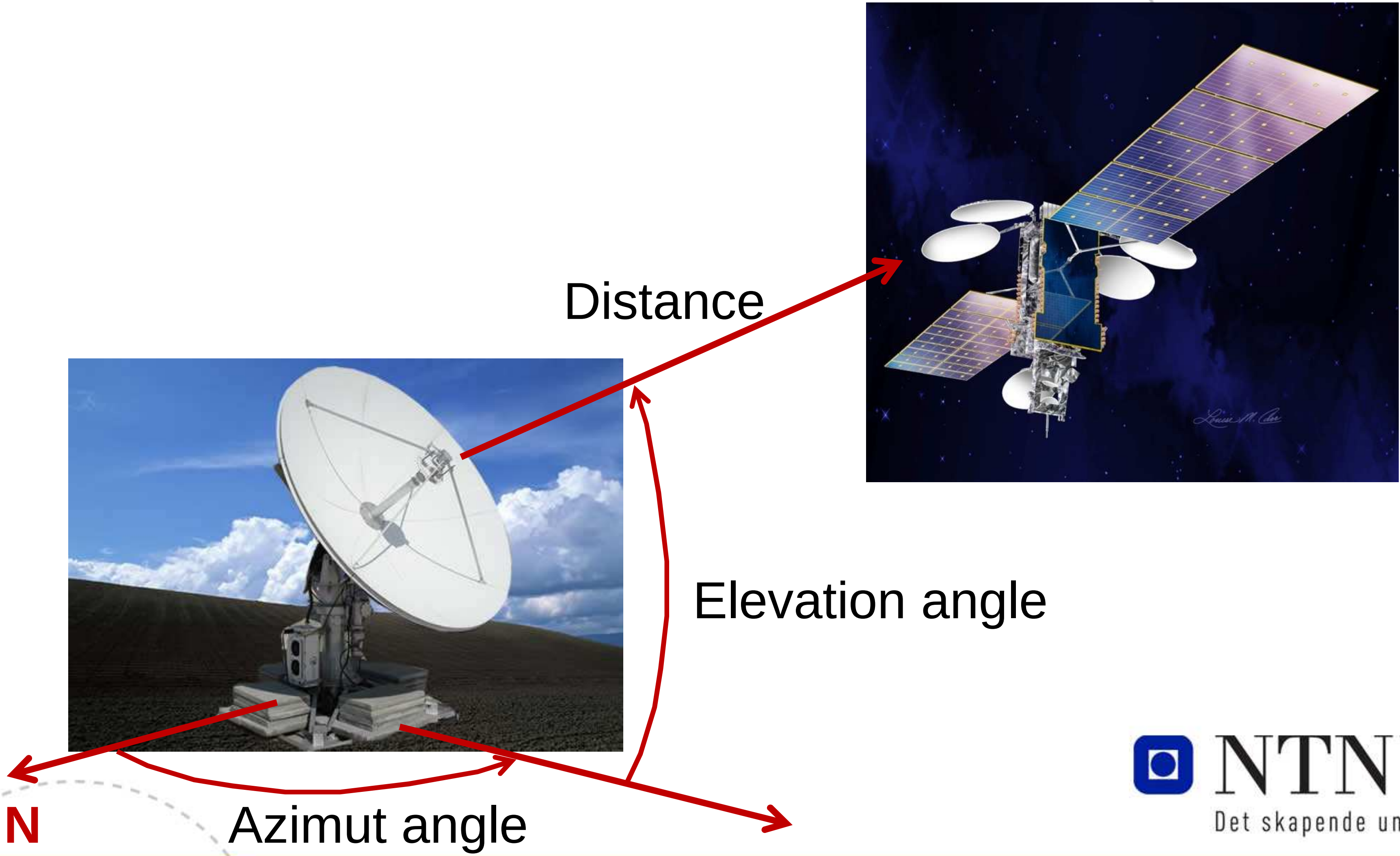
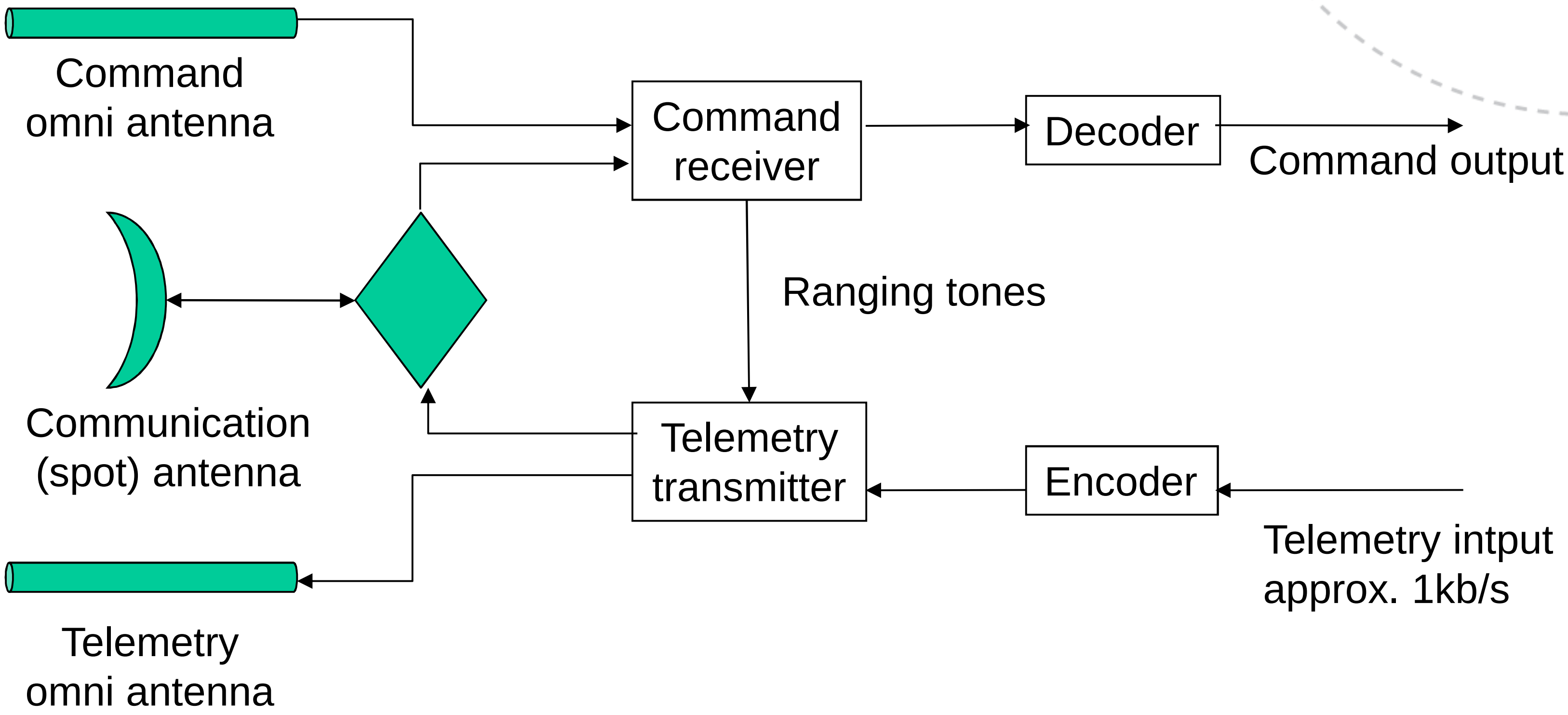
Control room at Nittedal earth station

Determination of the satellite position



Satellite TT&C system



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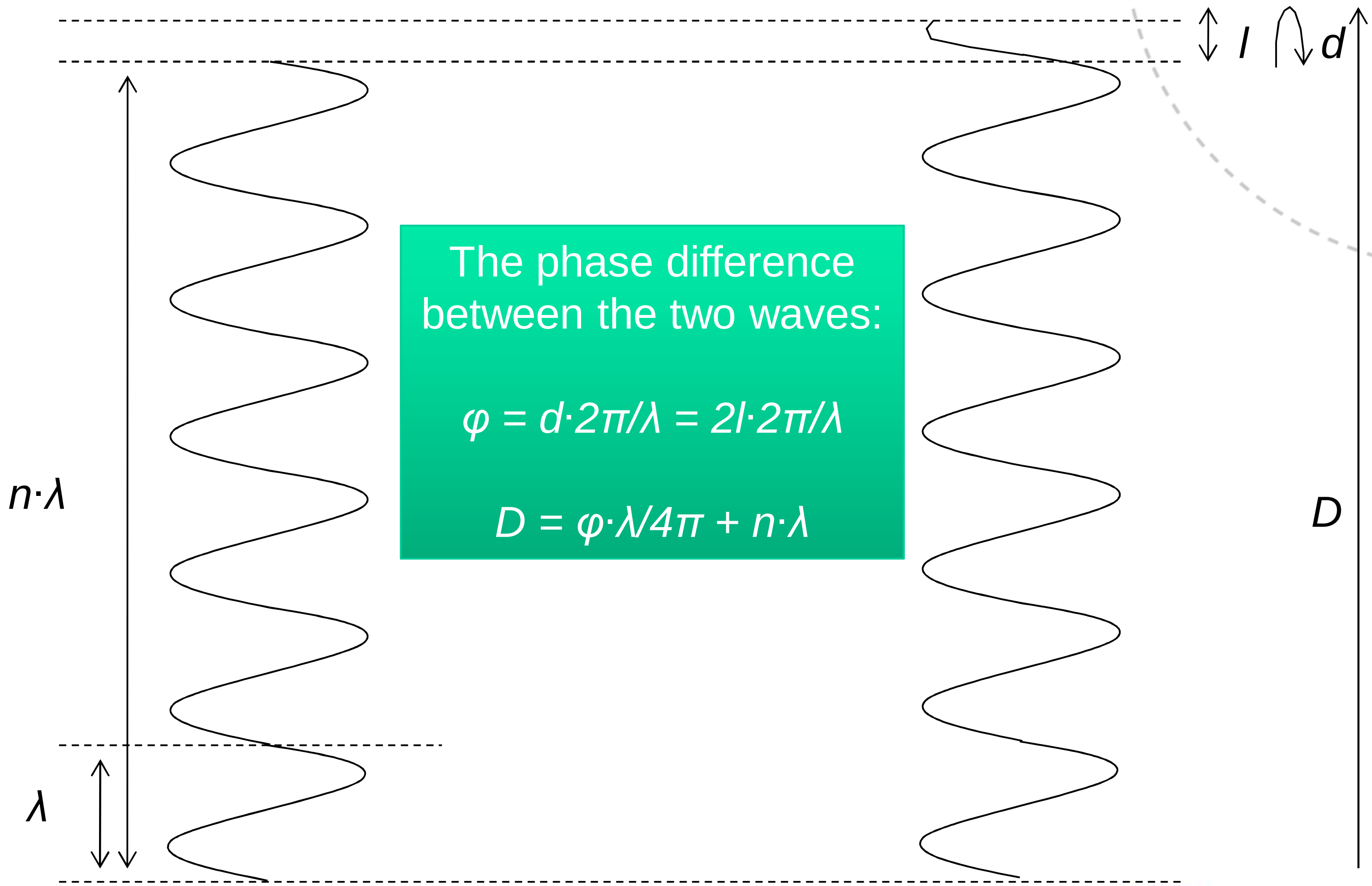
Measuring the distance with ranging tones



The phase difference between the two waves:

$$\varphi = d \cdot 2\pi/\lambda = 2l \cdot 2\pi/\lambda$$

$$D = \varphi \cdot \lambda/4\pi + n \cdot \lambda$$

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Tone ranging example

$f = 28\text{kHz}$, $\lambda = 10700\text{m}$

$$D = \varphi \cdot \lambda / 4\pi + n \cdot \lambda = (\lambda/2) \cdot (\varphi/2\pi) + n \cdot \lambda$$

If the phase is measured with an accuracy of 1°, the accuracy in the distance will be (10700/2) ·(1/360)=15m.

But there is the $n \cdot \lambda$ ambiguity, so several tones are needed in order to solve the phase ambiguity.

The lowest tone used for ranging GEO satellites is 4Hz, i.e. λ=75000km. Then $\lambda/2$ = 37500km, and there is no ambiguity for GEO. The accuracy is about 100 km.

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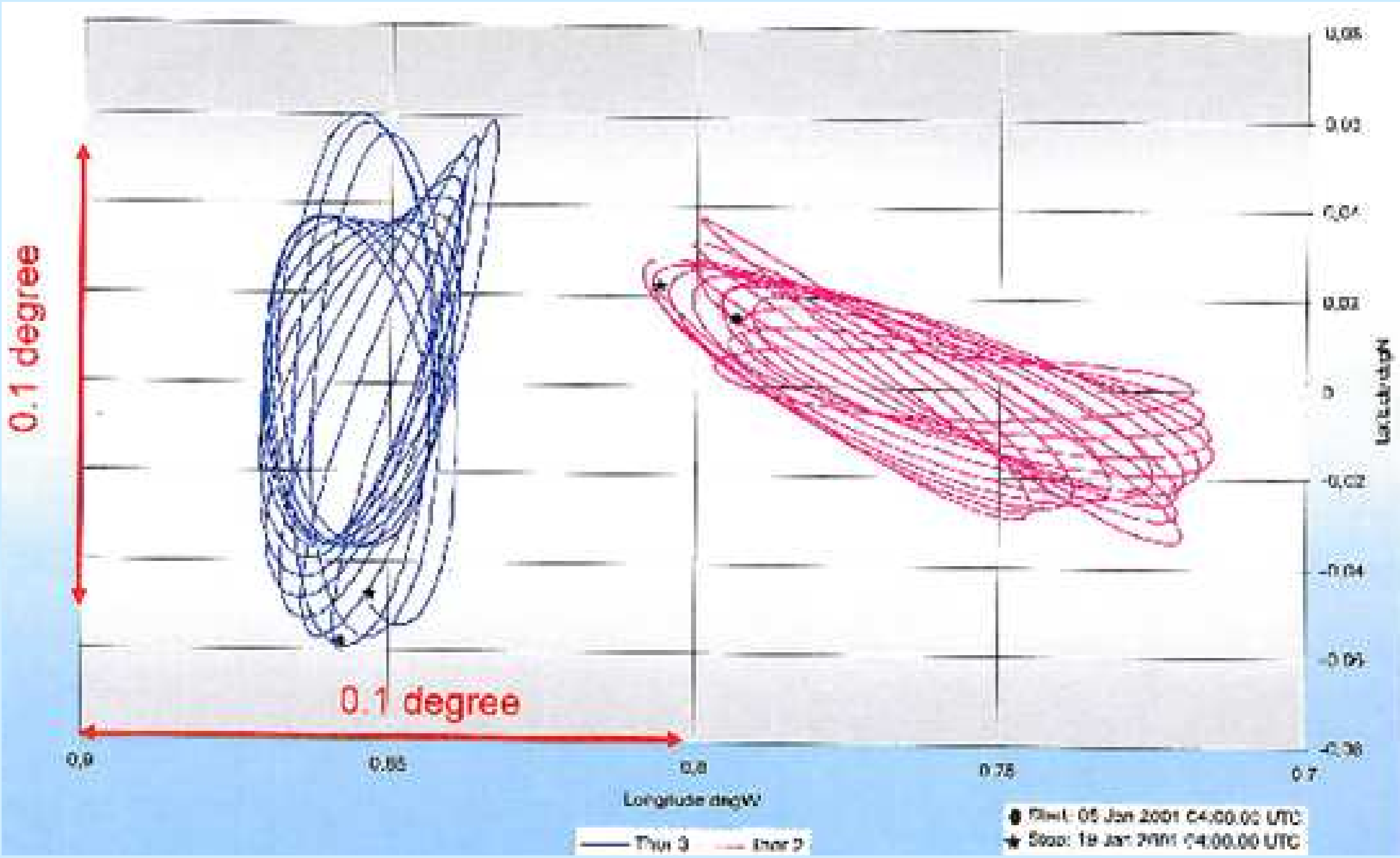
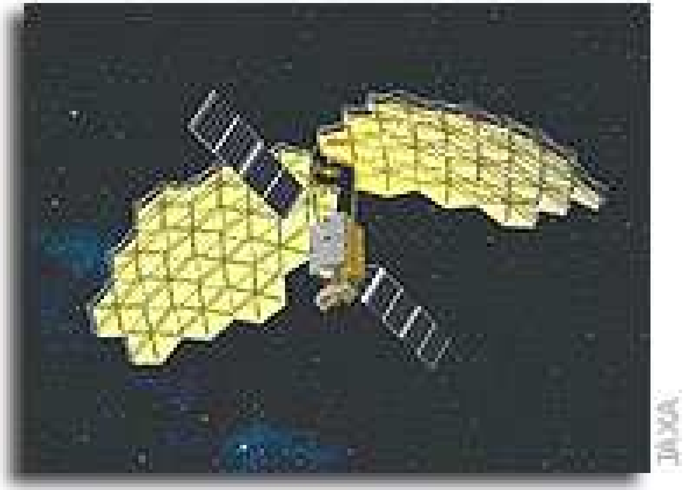
Thor II and III – 14 days of movement



6. Mechanical

- Mechanical structures are important for launch. The interface between the satellite and the rocket is challenging.
- The structures for mounting the components are important.
- Protection to radiation and debris may save a satellite.
- Mechanical deployment of antennas and solar panels is vital for power and telemetry.



Japanese KIKU 8 satellite launched in 2006

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Mechanisms

- Motors for driving the solar panels
- De-spun motor (dual spin satellite)
- Momentum/reaction wheels
- Hinges for antennas
- Electro-mechanical valves
- Gyros
- etc

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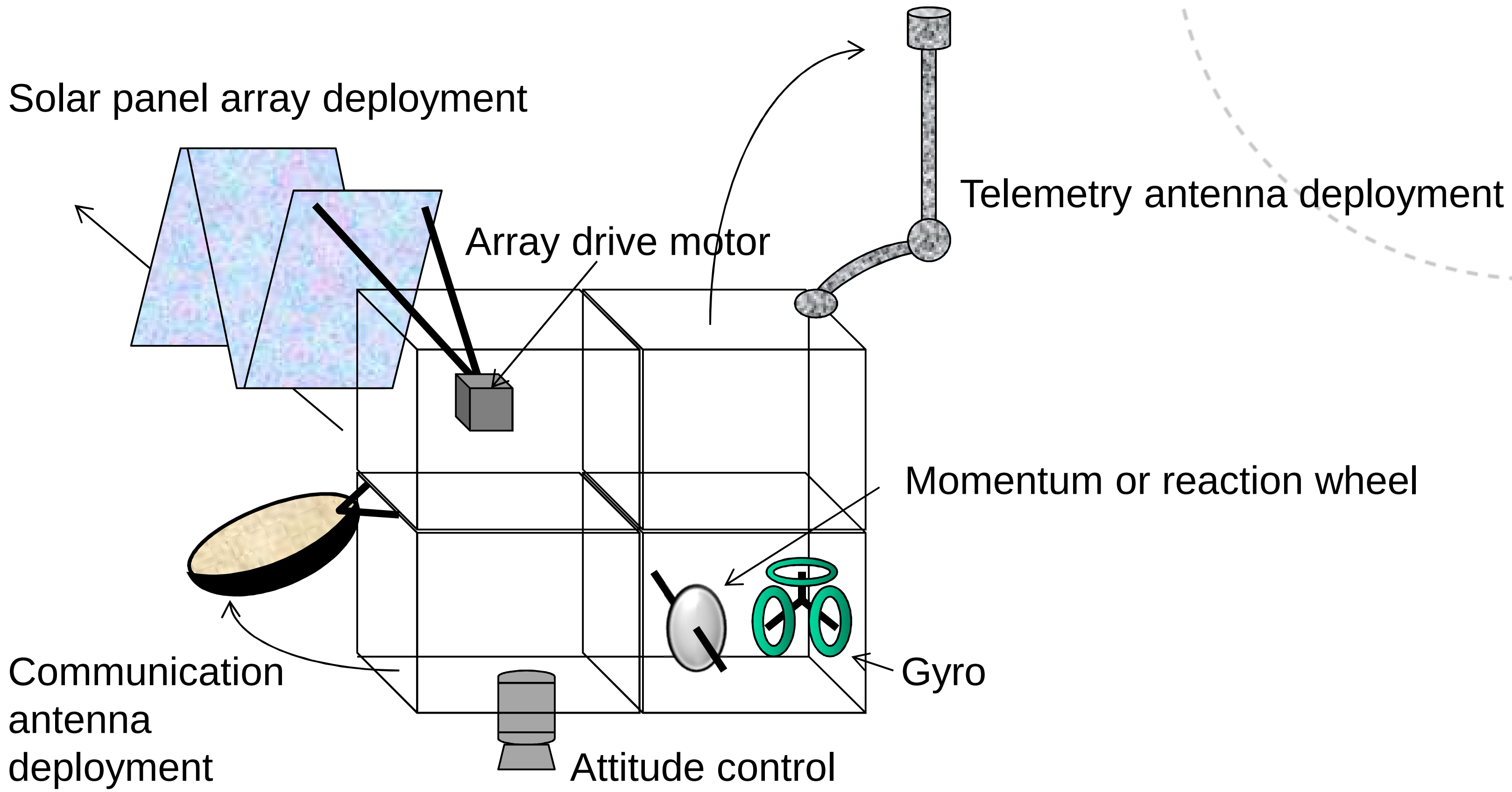
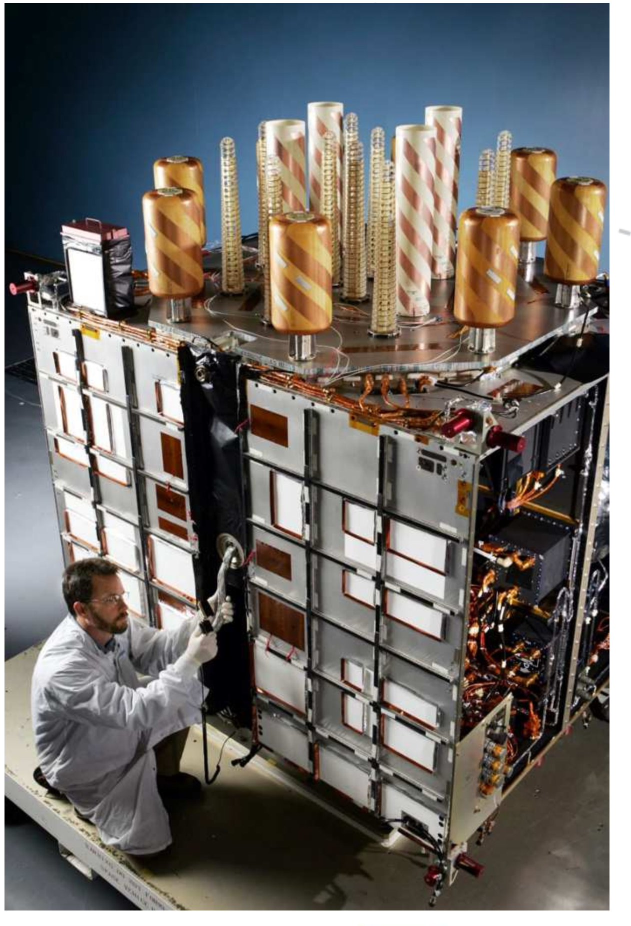
http://www.youtube.com/watch?v=PeaHbgU0ExQ

Mechanisms


Solar panel array deployment
Array drive motor
Telemetry antenna deployment
Momentum or reaction wheel
Communication antenna deployment
Gyro
Attitude control

7. Payload

- Navigation
- Remote sensing
- Communication



Purpose: Generally to create revenue

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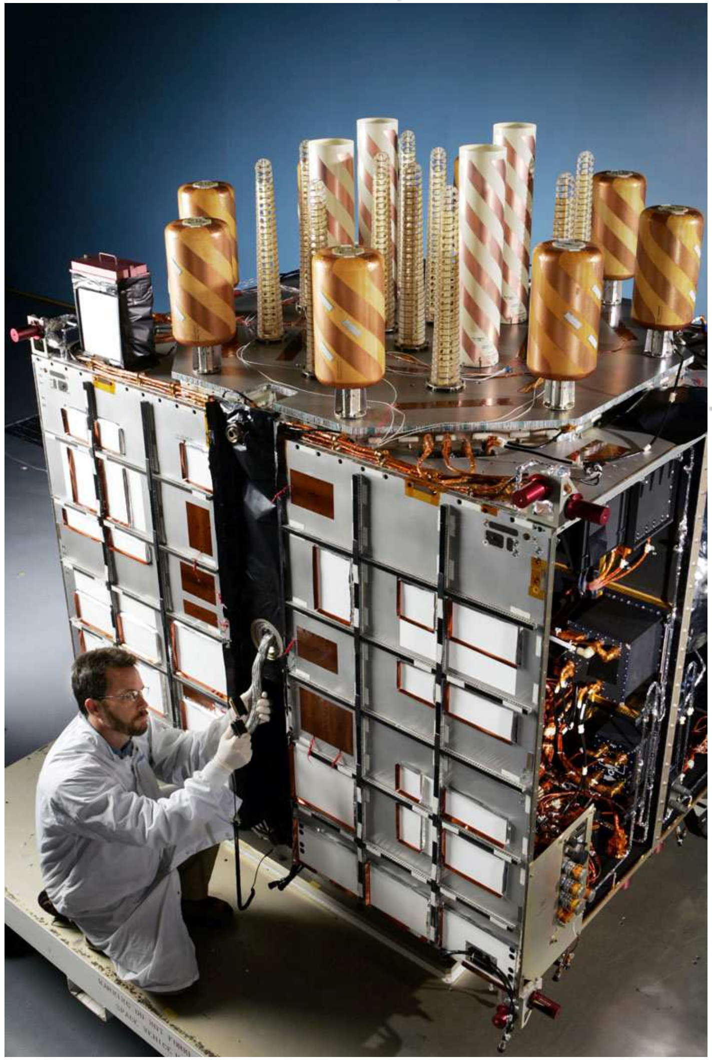
GPS payload



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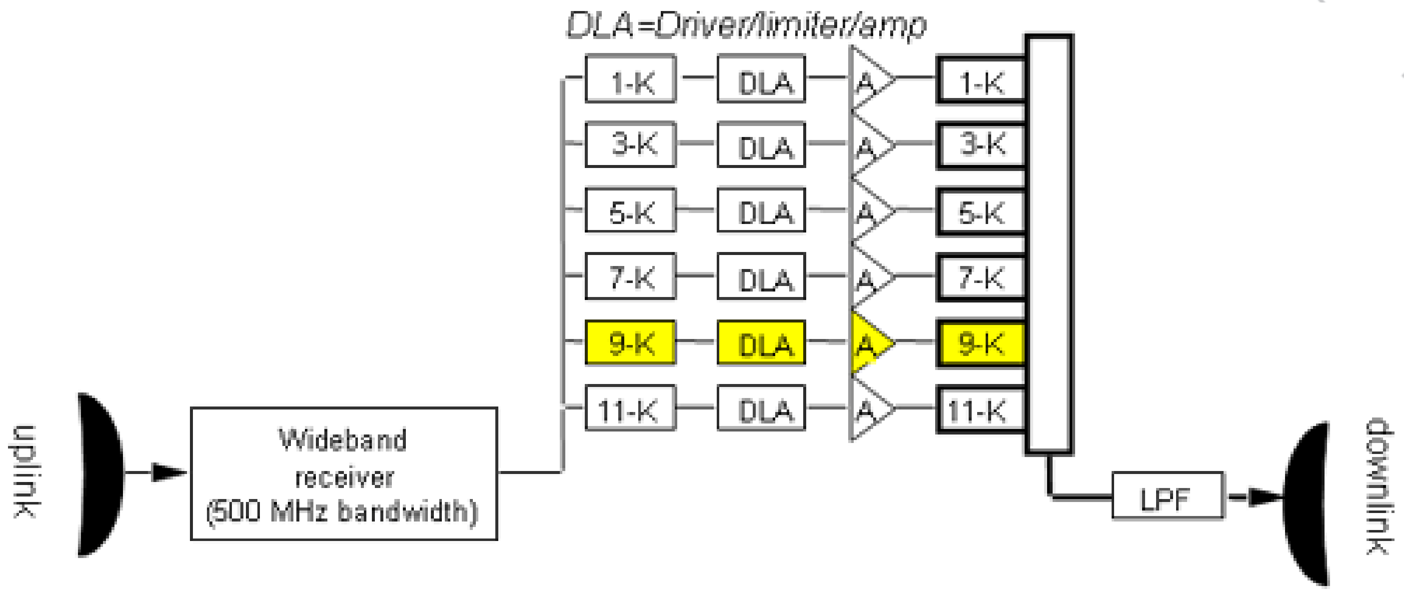
http://www.gpsreview.net/next-gps-satellite-to-launch-monday/

Communication payload



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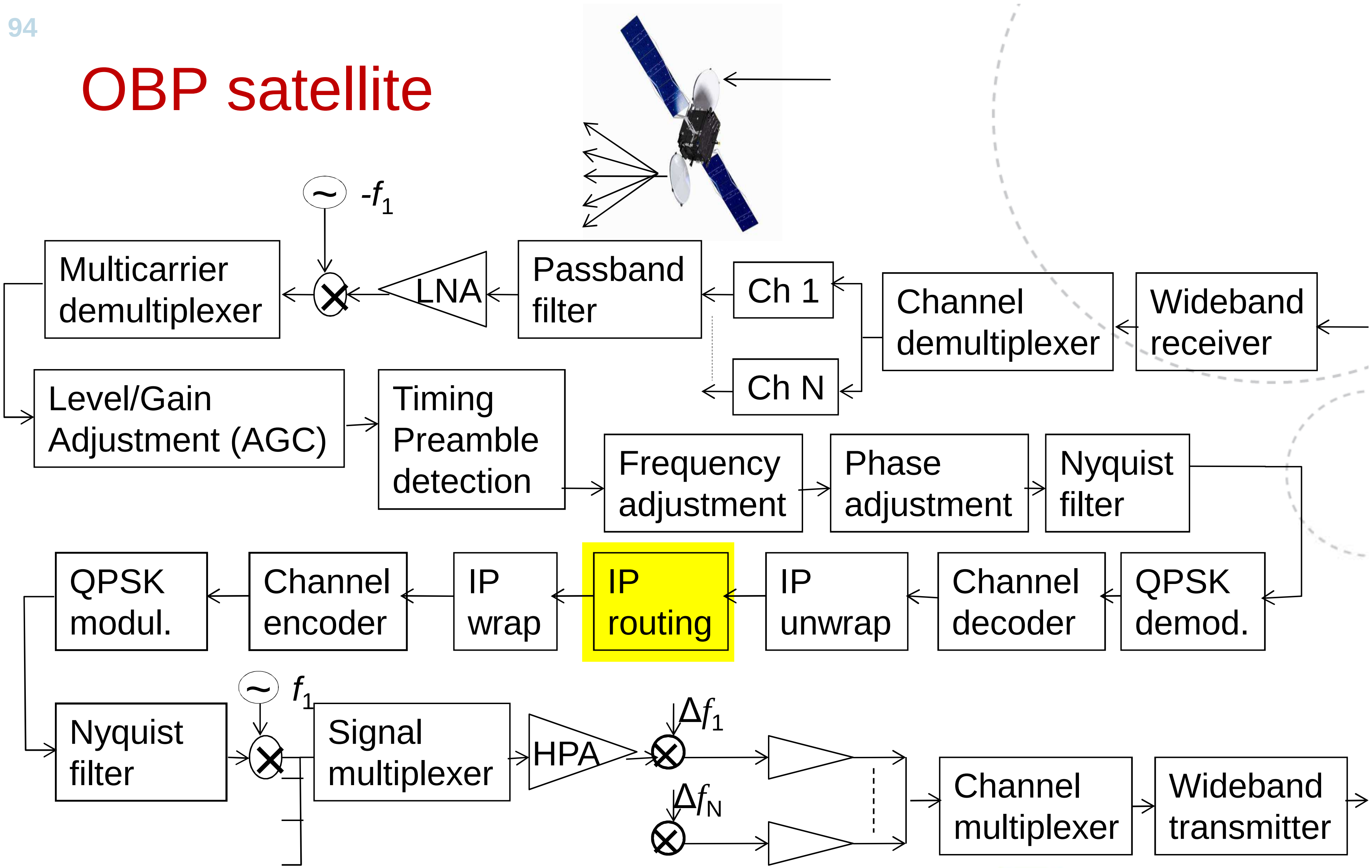
OBP satellite



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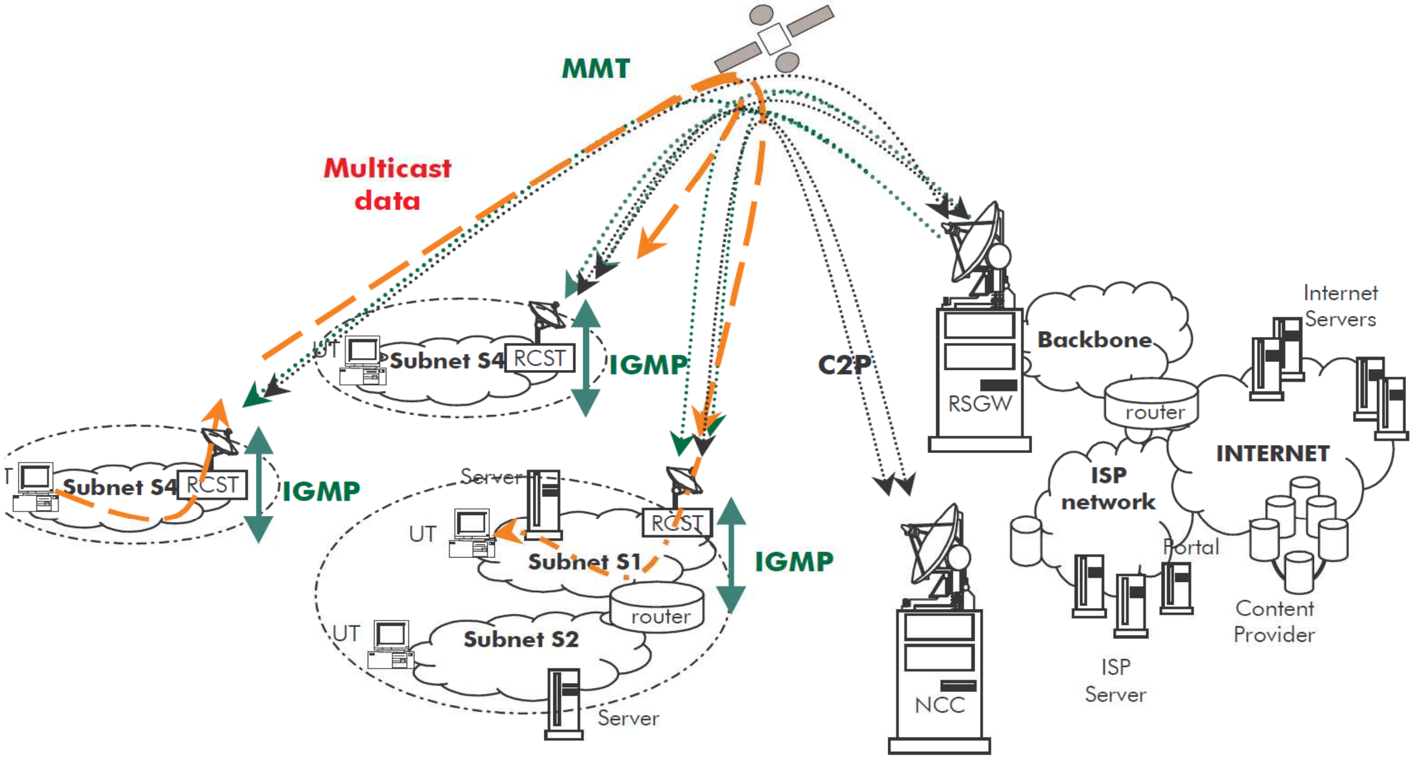
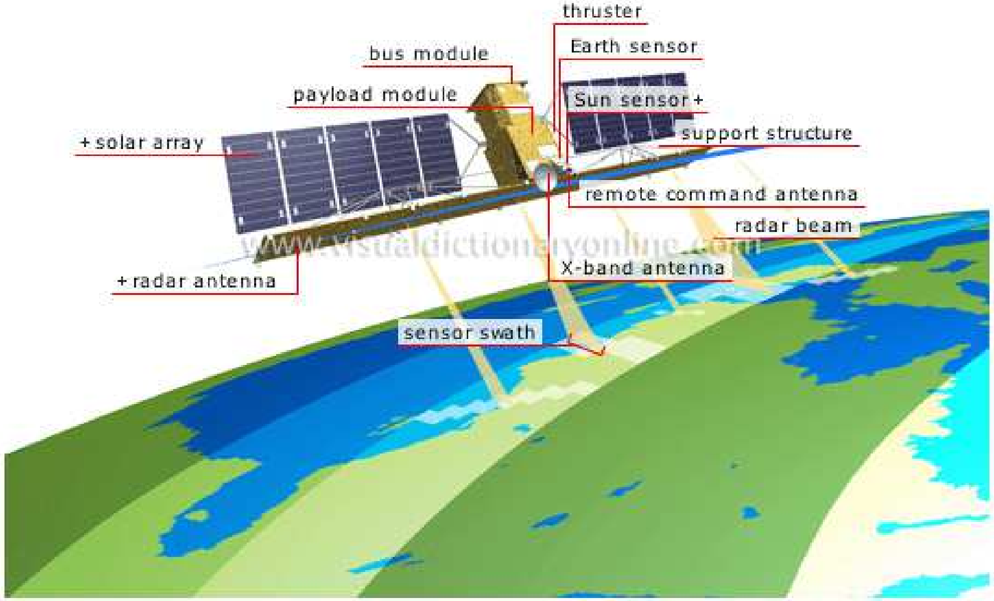
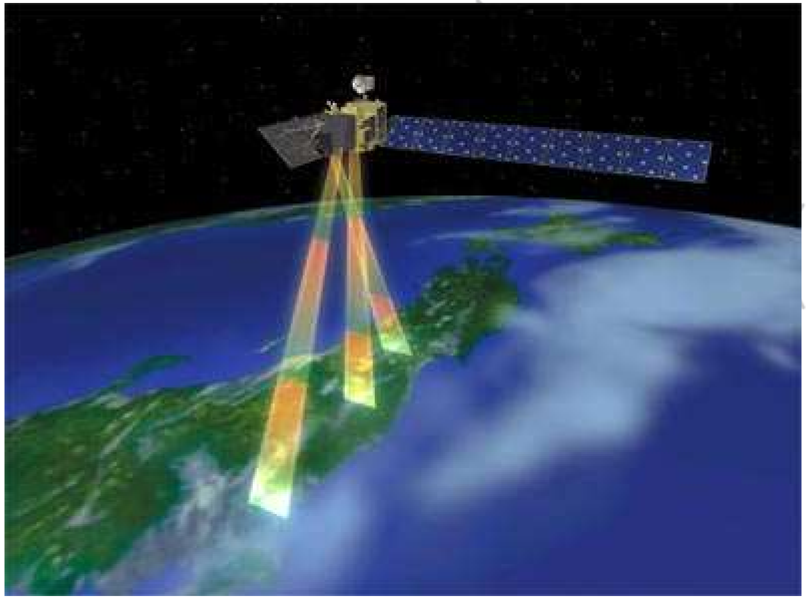
Example – Amerhis satellite



Mesh IP Multicast over AmerHis Network

Earth observation payload





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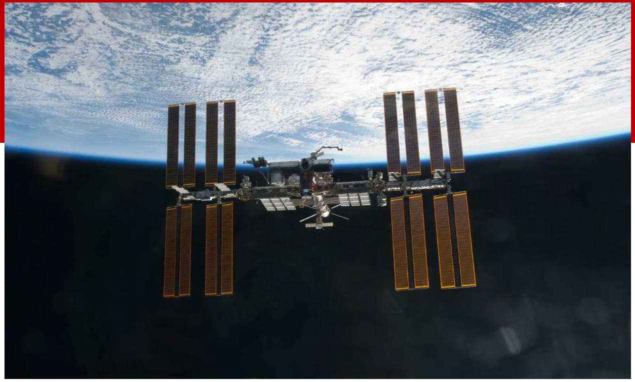
8. Debris



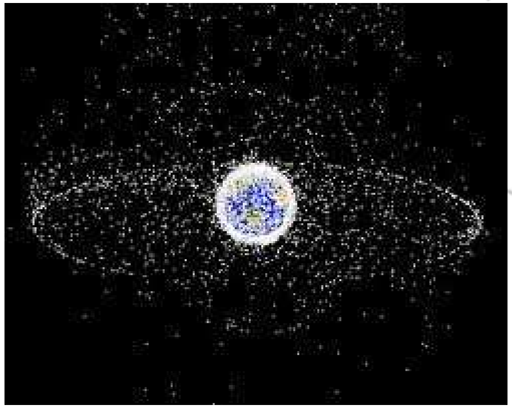
(Dagbladet) 06.02.2011: Mannskapet på Den internasjonale romstasjonen (ISS) har de siste timene vært klare til å evakuere inn i den tilliggende russiske Sojuz-romkapselen på grunn av faren for kollisjon med kinesisk romskrot - en kollisjon som kunne fått katastrofale følger.

Det eventuelle sammenstøtet ville ifølge beregningene ha funnet sted klokka 22.21 norsk tid - men Nasa kan allerede nå avkrefte at det blir noen kollisjon; romsøppelet vil ikke engang komme innenfor den svære sikkerhetssonen på 25 x 25 x 0,75 km.

Nasas kontrollsenter i Houston har holdt øye med skrotet - restene av en kinesisk satellitt Kina selv skjøt ned i 2007 - siden i dag tidlig, og informerte kommandør Dmitrij Kondratjev om at han og de to andre om bord burde stå klare til å søke tilflukt i Sojuz TMA-20-fartøyet.

Tre andre astronauter er for tida på vei til ISS, og skal etter planen legge til i morgen. Dermed vil romstasjonen ha fullt mannskap på seks, bestående av personer fra USA, Russland og Italia.

Romsøppel er et stadig voksende problem; rundt 12 500 kjente større biter skrot befinner seg i bane rundt jorda, det antas at tallet på biter store nok til å gjøre alvorlig skade er 600 000. Skrotet stammer stort sett fra utrangerte satellitter, og problemet blir ikke mindre i ei tid med mer og mer aktivitet i jordas umiddelbare nærhet.



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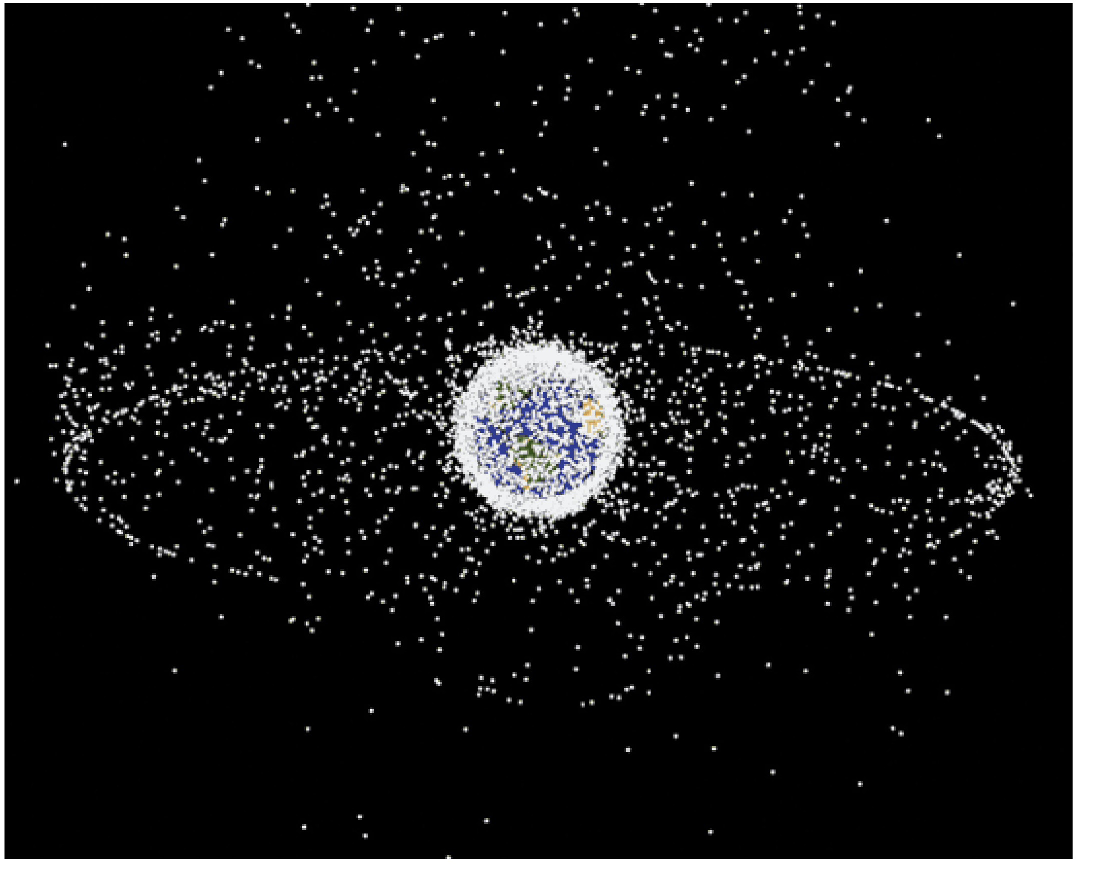
Space debris

- A graveyard orbit is significantly above the GEO orbit
- satellites left in a graveyard orbit will slowly break apart as micrometeorites hit them, and the smaller fragments may filter back down to lower altitudes
- De-orbiting a geostationary satellite requires a speed change of about 1500 m/s
- Re-orbiting it to a graveyard orbit requires about 11 m/s
- For GEO satellites, the graveyard orbit is a few hundred kilometers above the operational orbit
- Transfer to graveyard orbit above geostationary orbit however requires the same amount of fuel that a satellite needs for approximately three months of station keeping
- It also requires a reliable attitude control during the transfer maneuver
- While most satellite operators try to perform such a maneuver at the end of the operational life, only one-third succeed in doing so



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Tracking problem

- Only space debris with a size above 10 cm in diameter is tracked
- 20000 tons of natural material hit earth every year (dust, meteoroids, asteroids, comets....)
- Earth's space debris amounts to about 3000 tons
- In low earth orbit, a piece of debris may move at 7 km/s or more
- Never ending accumulation......

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Debris hole in Challenger 1983
caused by a flake of paint 0.2mm in diameter

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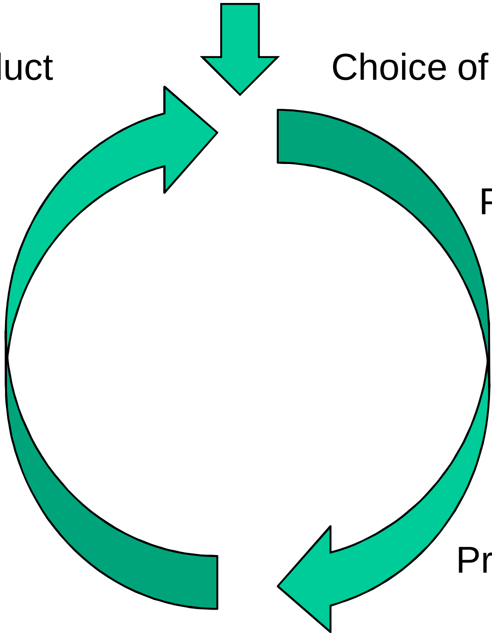
Debris may also return to earth

9. Summary

Mission design:

Payload definition



Choice of orbit

Power generation

Attitude control

Orbital control

Propulsion system

TT&C

On-board data systems

Structure and mechanisms

Thermal systems

Choice of launch vehicle

Space and launch environment

Test and product assurance

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The 7 phases of a Satellite project

1. Wild enthusiasm
2. Desillusionment
3. Confusion
4. Panic
5. Search for the guilty
6. Punishment of the innocent
7. Promotion of non-participants

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Acknowledgement

This presentation is to a large extent built on the presentation made by Professor Odd Gutteberg on the subject.