Introduction to Aerospace Engineering

Lecture slides

Introduction to Aerospace Engineering AE1101

Dept. Space Engineering

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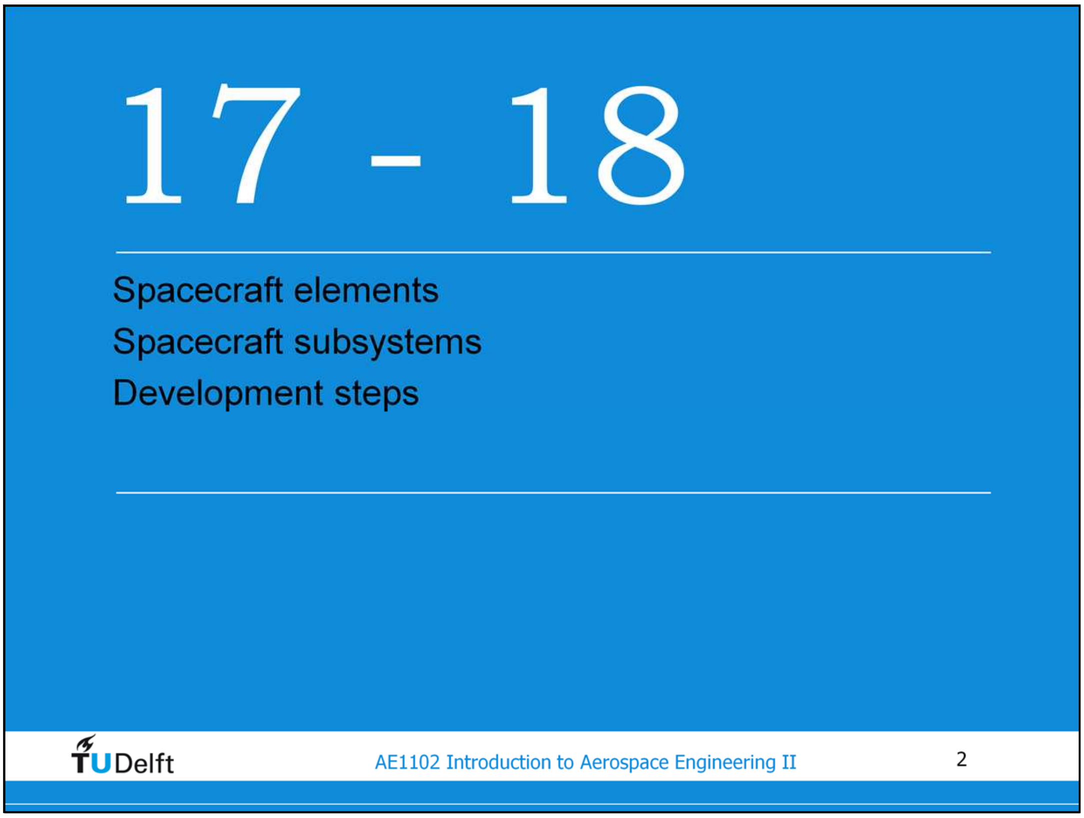
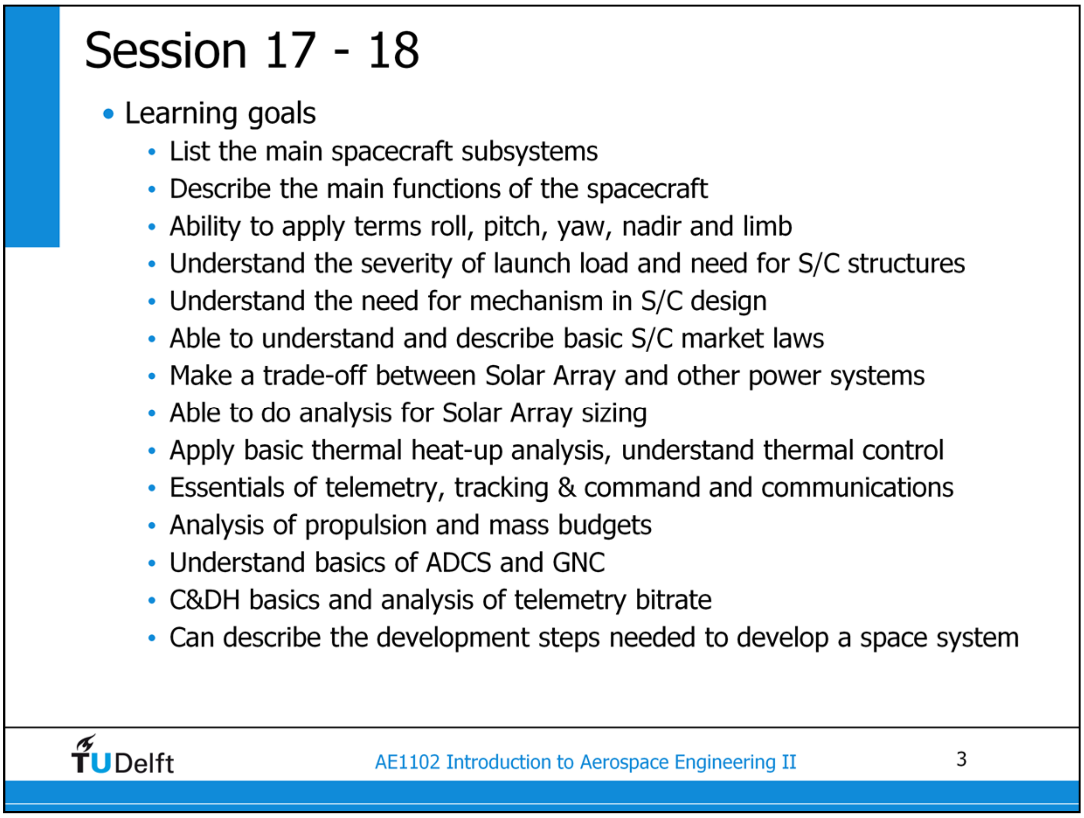


17 - 18

Spacecraft elements
Spacecraft subsystems
Development steps

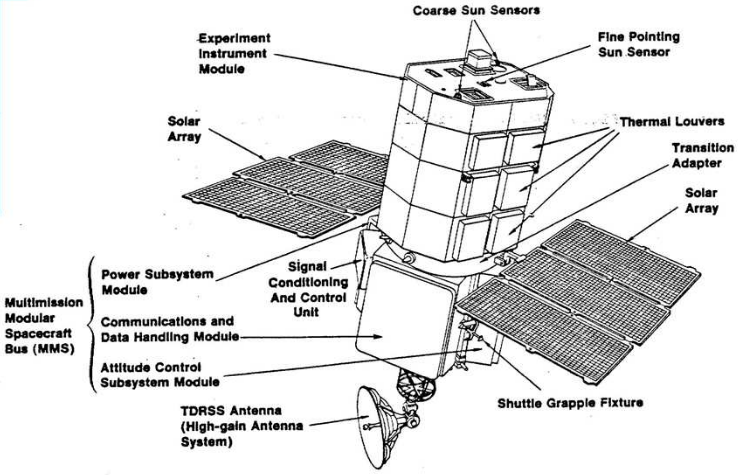
Part of the contents of this presentation originates from the lecture “Space Engineering and Technology I, Part I” (ae1-801/1), by R. Hamann.



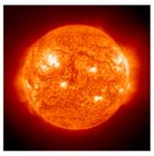
Session 17 - 18

- Learning goals
  - List the main spacecraft subsystems
  - Describe the main functions of the spacecraft
  - Ability to apply terms roll, pitch, yaw, nadir and limb
  - Understand the severity of launch load and need for S/C structures
  - Understand the need for mechanism in S/C design
  - Able to understand and describe basic S/C market laws
  - Make a trade-off between Solar Array and other power systems
  - Able to do analysis for Solar Array sizing
  - Apply basic thermal heat-up analysis, understand thermal control
  - Essentials of telemetry, tracking & command and communications
  - Analysis of propulsion and mass budgets
  - Understand basics of ADCS and GNC
  - C&DH basics and analysis of telemetry bitrate
  - Can describe the development steps needed to develop a space system

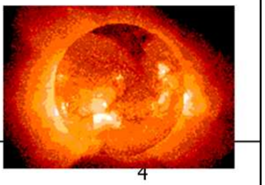
Solar Max



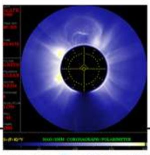
UV image of the sun (SOHO)

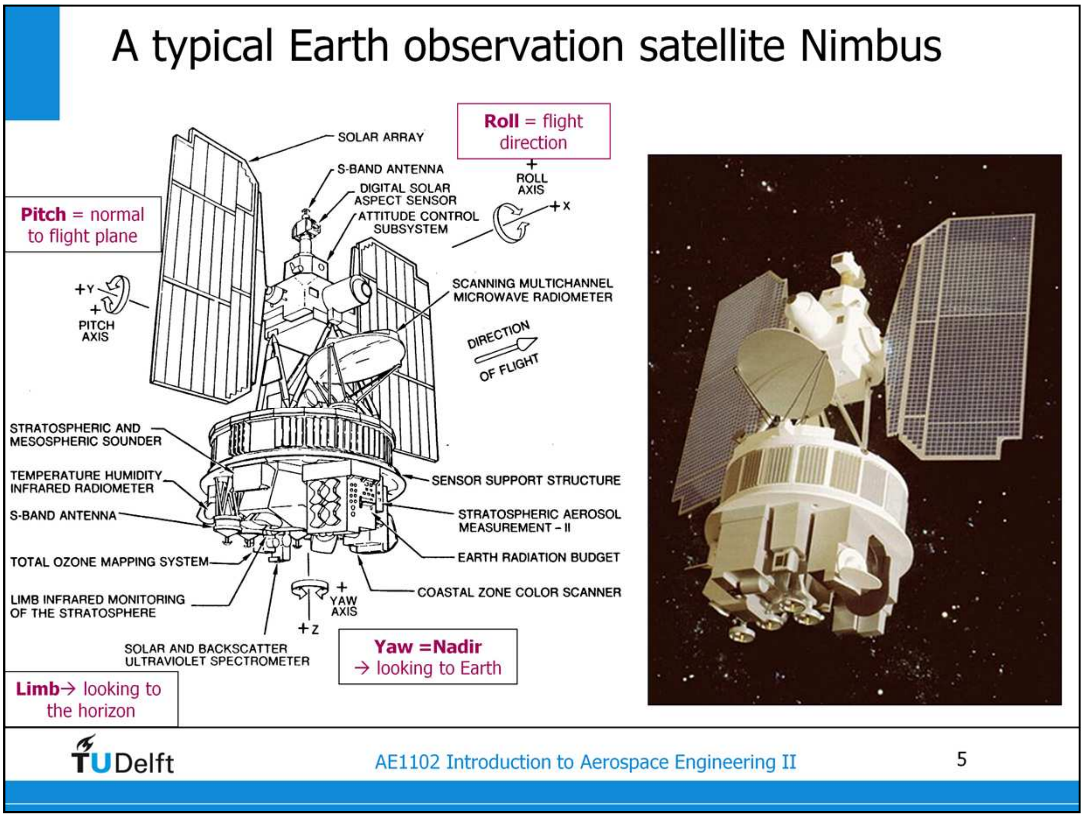


Röntgen image of the Sun (Yohkoh satellite)



Coronograph image (Solar Max)





A typical Earth observation satellite Nimbus

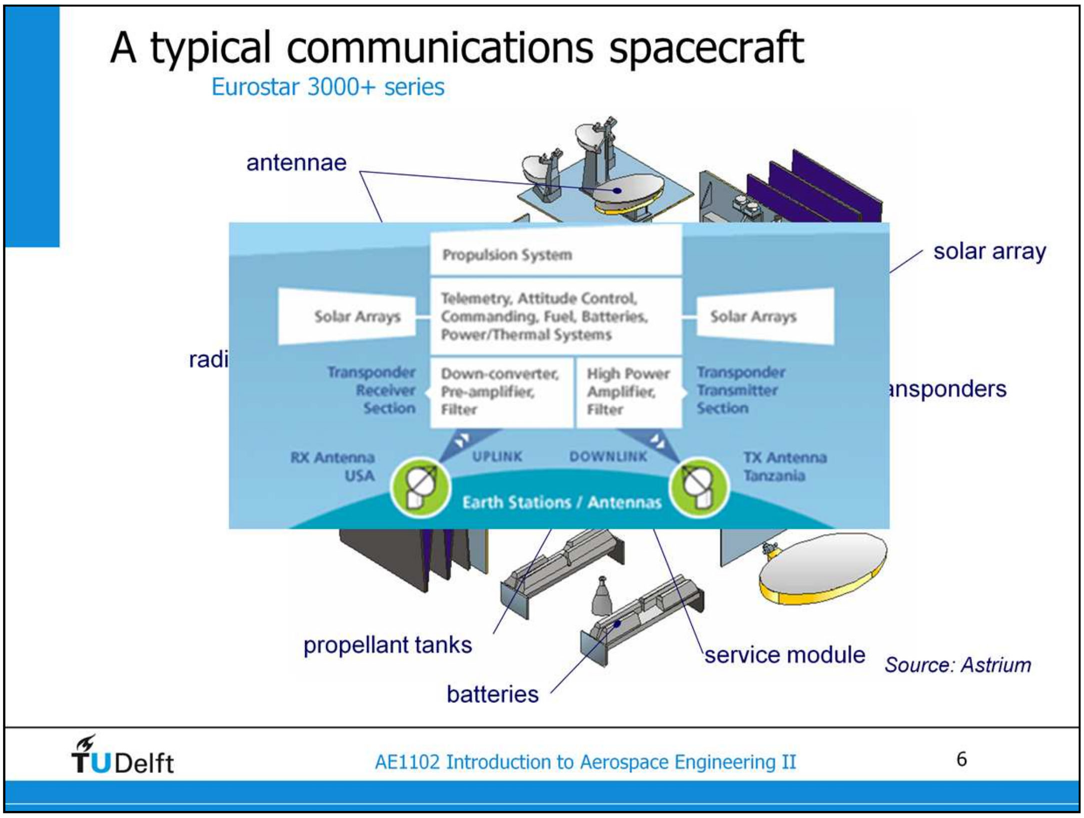
Roll = flight direction

Pitch = normal to flight plane

Yaw =Nadir → looking to Earth

Limb→ looking to the horizon

Nimbus/Landsat satellite - Artist's drawing of the general design of the Nimbus series of satellites. The solar-panel "wings" move throughout the day to track the Sun during the daylight part of the satellite’s orbit. The 10-foot-tall satellite has the attitude control system on top, separated from a 5-foot-diameter "sensory ring" (center) with scaffolding. The sensory ring holds the batteries and electronics for each of the sensors that are mounted underneath the ring (bottom).



A typical communications spacecraft

Eurostar 3000+ series

Source: Astrium

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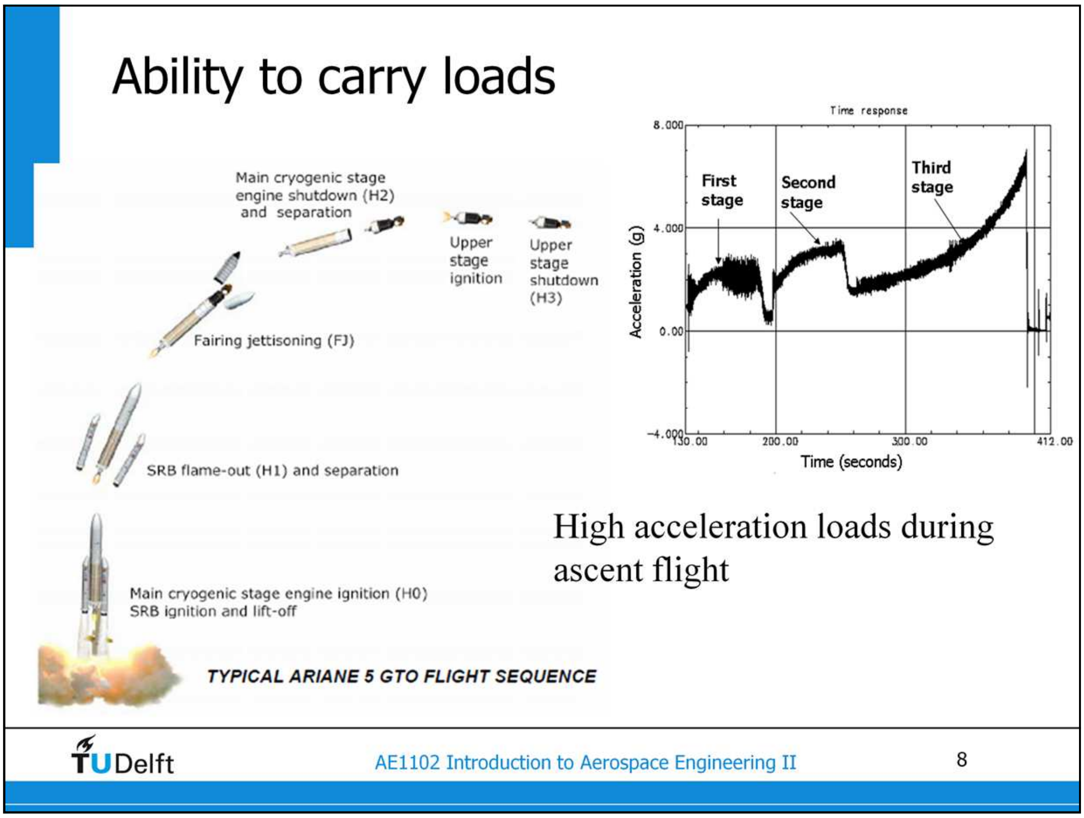
EUROSTAR 3000 Series + Diagrammatic representation of satellite - The Thuraya-1 satellite, part of a turnkey mobile communications system built for Thuraya Satellite Telecommunications Company of the United Arab Emirates - Sea Launch Zenit-3SL rocket from the a the equator in the middle of the Pacific Ocean. The on-board processor constantly changes the beam configuration to match fluctuating usage patterns and make the most efficient use of the bandwidth available. In addition, the digital processor flexibly forms and switches many thousands of user channels simultaneously. The communications P/L design was one of the most powerful ever undertaken by Hughes, now Boeing, and uses an enhanced active phased-array antenna design in combination with a company developed state-of-the-art, digital signal processor for beam forming, channel formation and switching. The spot beams can be redirected on-orbit, wherever needed from big cities to rural areas and even at sea. Thuraya has the capacity for 13,750 simultaneous calls. Thuraya-1 will be positioned at 44 degrees East longitude, with a contracted life of 12 years in geosynchronous orbit. Power is derived from two wings of panels, each with dual-junction gallium arsenide solar cells.

Spacecraft Subsystem functions

Principal functions
Adjust orbit and attitude, dump angular momentum
Determine and control orbit
Determine and control attitude, point and manoeuver the spacecraft, manage angular momentum
Communicate with ground, support spacecraft tracking
Process commands, perform data processing/ formatting, provide computing power
Control equipment temperature
Generate and distribute power
Provide structural integrity, provide motion possibility to structural parts



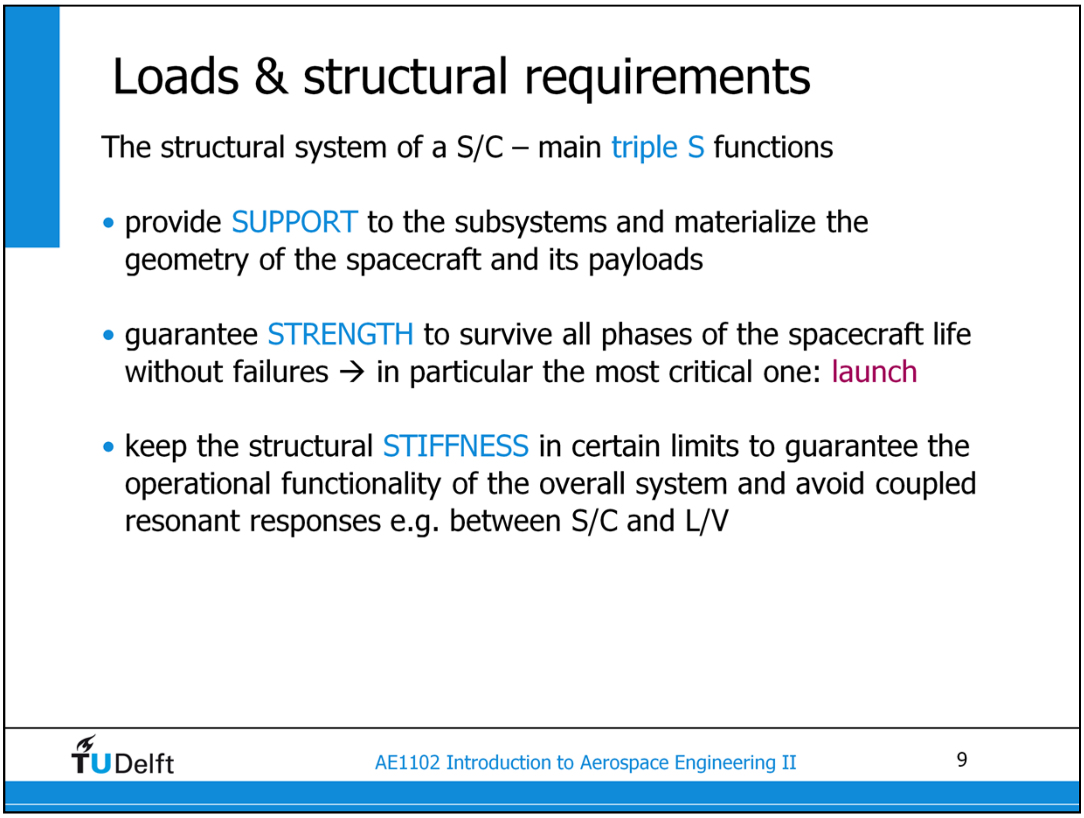
Ability to carry loads

High acceleration loads during ascent flight

The QSL, sinusoidal, shock and acoustic loads are in general specified in the L/V User’s manual. The random loads are derived from acoustic loads, in fact the random responses of the equipment mounted to the structure.

Homework:

Students shall collect and list design loads and natural frequency requirements of specific space launchers in Excel file.



Loads & structural requirements

The structural system of a S/C – main triple S functions

- provide SUPPORT to the subsystems and materialize the geometry of the spacecraft and its payloads
- guarantee STRENGTH to survive all phases of the spacecraft life without failures → in particular the most critical one: launch
- keep the structural STIFFNESS in certain limits to guarantee the operational functionality of the overall system and avoid coupled resonant responses e.g. between S/C and L/V

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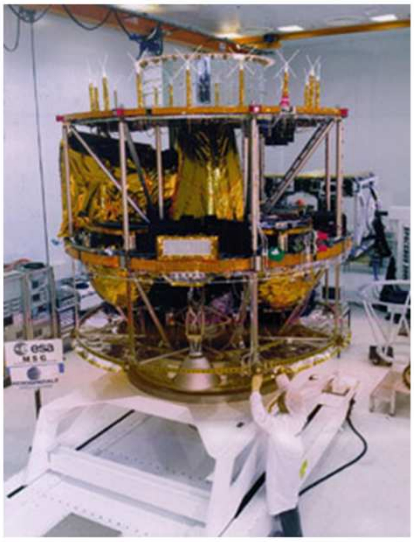
The Quasi-Staic-Load (QSL), sinusoidal, shock and acoustic loads are in general specified in the L/V (Launch Vehicle) User’s manual. The random loads are derived from acoustic loads, in fact the random responses of the equipment mounted to the structure.

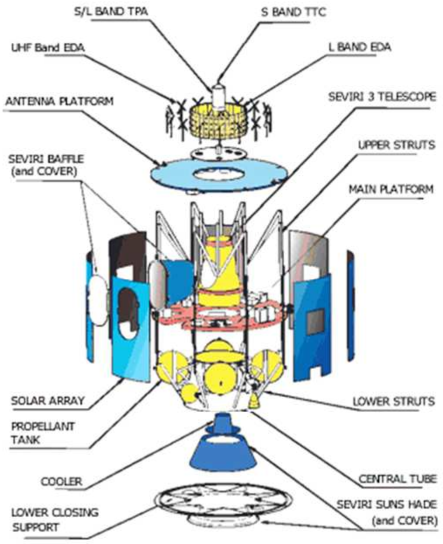
Homework (optional)

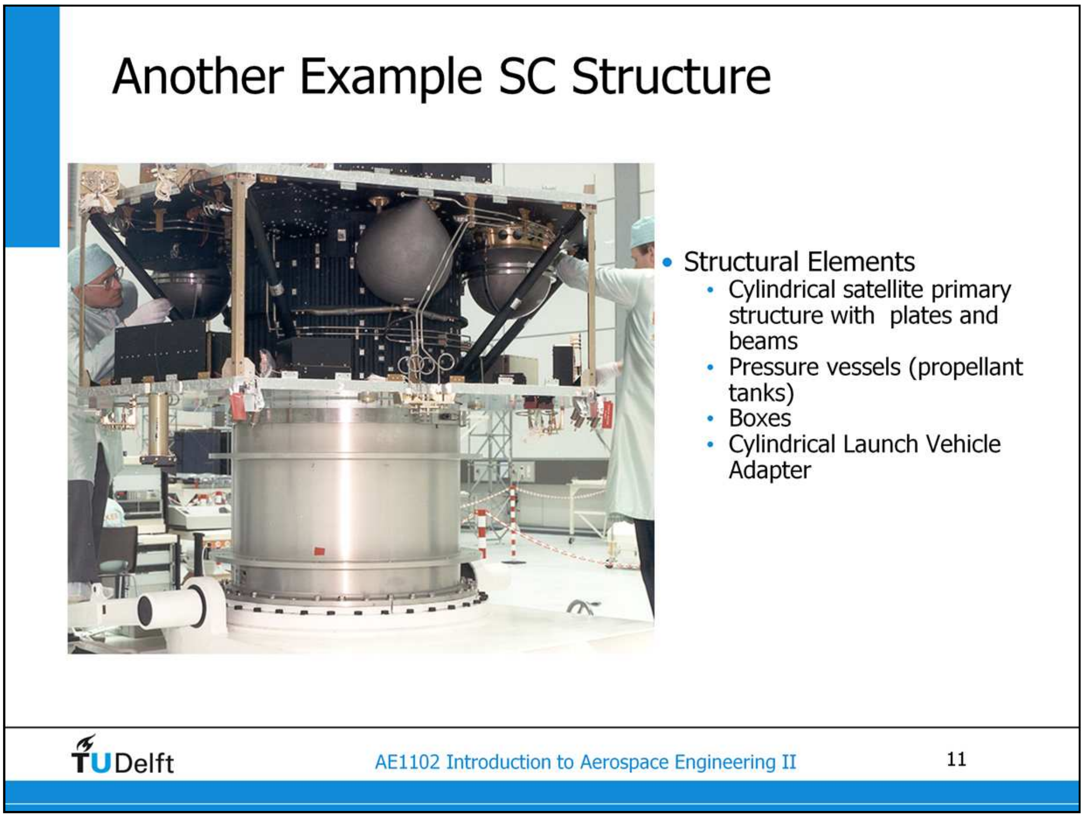
Students may collect and list design loads and natural frequency requirements of specific space launchers in Excel file.

Example S/C Structure



Typical meteorological satellite



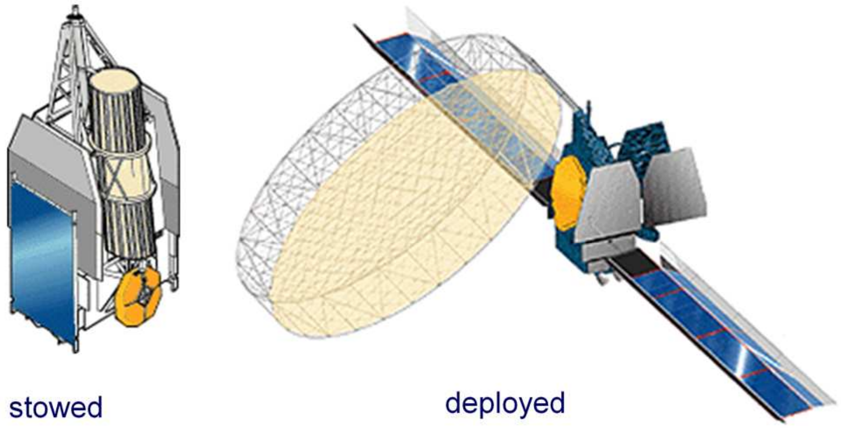


Another Example SC Structure

- Structural Elements
  - Cylindrical satellite primary structure with plates and beams
  - Pressure vessels (propellant tanks)
  - Boxes
  - Cylindrical Launch Vehicle Adapter

Primary structure of Hipparcos satellite

Need for mechanisms: Thuraya S/C

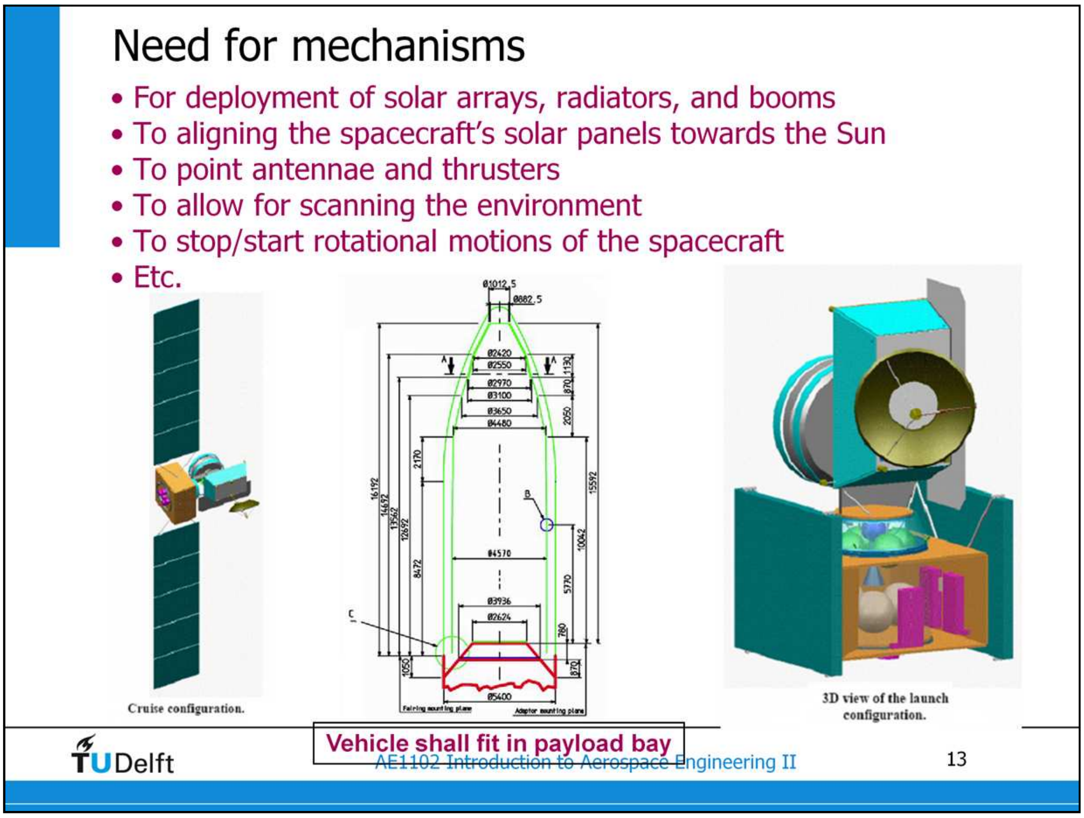


stowed

deployed

Vehicle shall fit in payload bay

http://www.hughespace.com/factsheets/geomobile/thuraya/



Need for mechanisms

- For deployment of solar arrays, radiators, and booms
- To aligning the spacecraft’s solar panels towards the Sun
- To point antennae and thrusters
- To allow for scanning the environment
- To stop/start rotational motions of the spacecraft
- Etc.

Cruise configuration.

3D view of the launch configuration.

Vehicle shall fit in payload bay

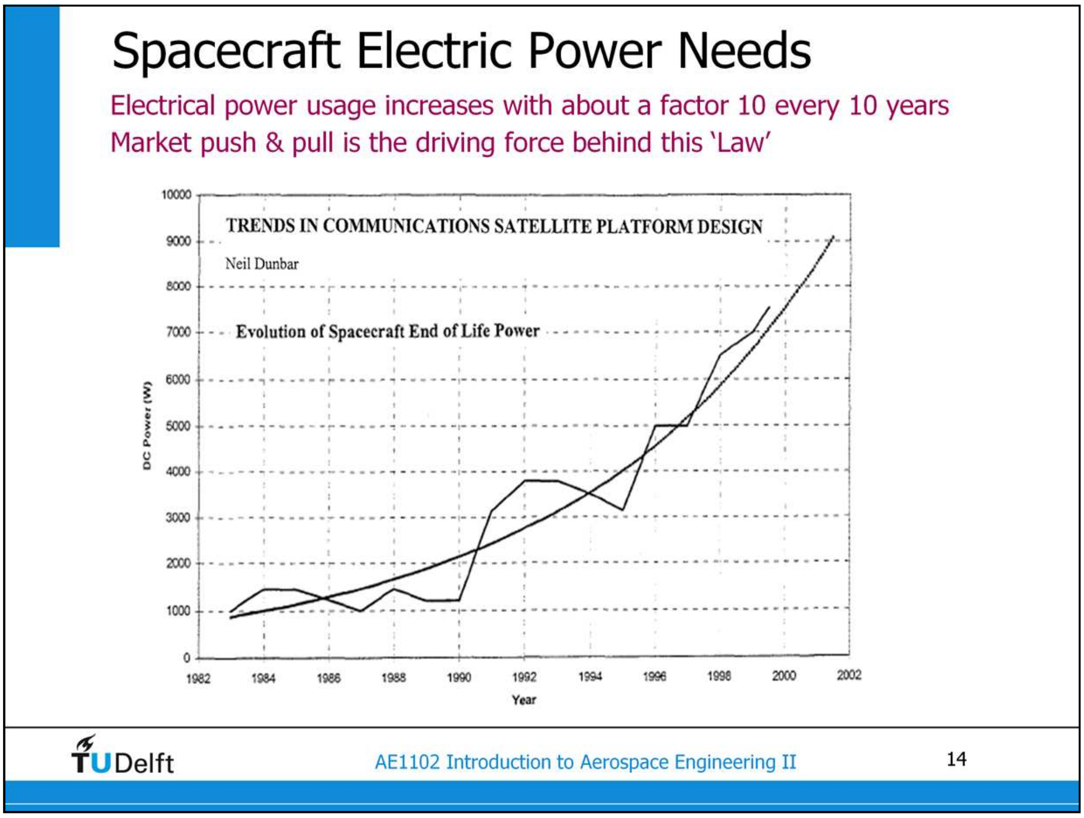
Slide shows ISS (included in the Table 1 in the hand out).

Bepi Colombo candidate configuration.

Slide to be used later on to show limitations to size as well as need for mechanisms. For the moment used her to remind me of incorporating some text on requirements from the other elements of a space system.

Homework:

Students shall collect and list payload bay dimensions and volume of specific space launchers in Excel file.



Spacecraft Electric Power Needs

Electrical power usage increases with about a factor 10 every 10 years
Market push & pull is the driving force behind this 'Law'

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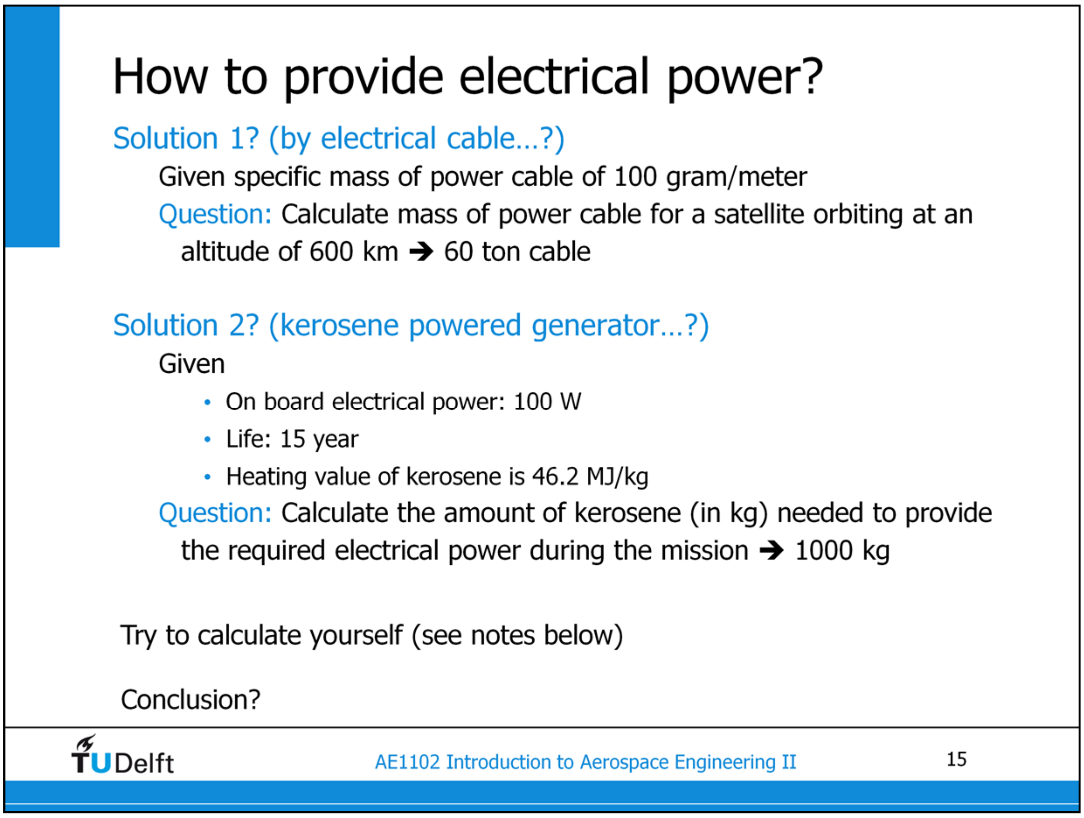
Typical installed powers for various kinds of spacecraft are compared in the next Table (from ESA bulletin 87)

Mission	Orbit Attitude	Installed Power (W) min. max.
Science: - astronomy	HEO, Fixed point Sun pointing (mostly)	
- deep space	Various transfer orbits Sun or planet pointing	200-1500
Telecommunication	GEO Earth pointing	500-5000
Earth Observation	LEO Earth pointing	500-5000
Meteorology	GEO Earth pointing	200-1500
Manned Vehicles	Transfer +LEO Various	1000-10000
Manned Stations	LEO Sun pointing	3000-30000



How to provide electrical power?

Solution 1? (by electrical cable…?)

Given specific mass of power cable of 100 gram/meter

Question: Calculate mass of power cable for a satellite orbiting at an altitude of 600 km ➔ 60 ton cable

Solution 2? (kerosene powered generator…?)

Given

- On board electrical power: 100 W
- Life: 15 year
- Heating value of kerosene is 46.2 MJ/kg

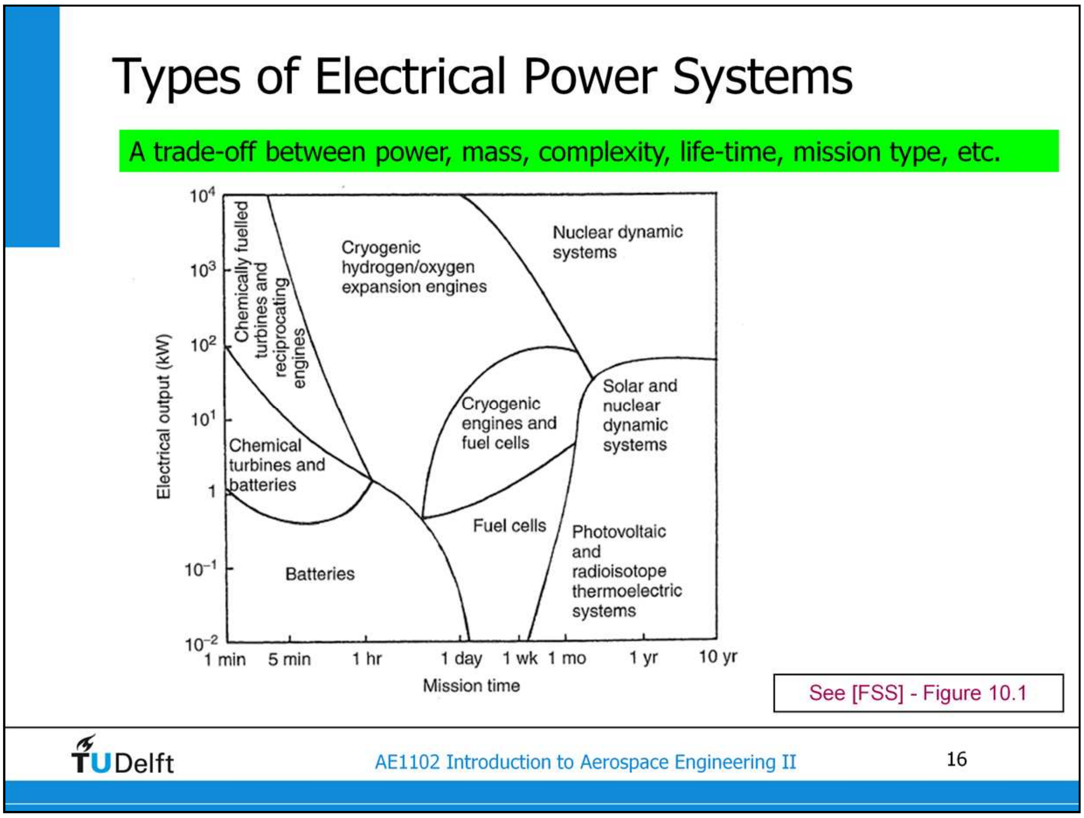
Question: Calculate the amount of kerosene (in kg) needed to provide the required electrical power during the mission ➔ 1000 kg

Try to calculate yourself (see notes below)

Conclusion?

600x1000x0.1=60 ton

100x15x365x24x3600=4.7E10 Joule → 4.7E10/46.2E6 =4.7E10/ 4.62E7 ~ 1000 kg but only 30% effective …..



Types of Electrical Power Systems

A trade-off between power, mass, complexity, life-time, mission type, etc.

See [FSS] - Figure 10.1

Important distinction is in systems that work independent from the Sun and those that depend on the distance to the Sun

Short life

- Batteries or fuel cells (for instance on launchers)

Long life:

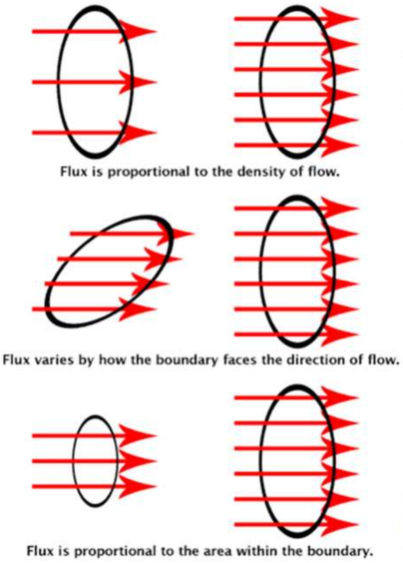
- Deep space: RTG's (Radioisotope Thermoelectric Generators)
- Inner planets: Photovoltaic (Solar-electric) power
- Earth orbit: Solar-electric power
  - Low power, low attitude accuracy: Body mounted
  - Low power, 3-axis control: Body mounted of deployable planar panels
  - High power: Deployable planar arrays

Cryogenic / Heat engines – see explanation earlier

Sigma =σ = Stefan-Boltzman constant = 5.67e-08 W/M2/K4

Epsilon = ε = effective emissivity of the radiating surface depends on the material

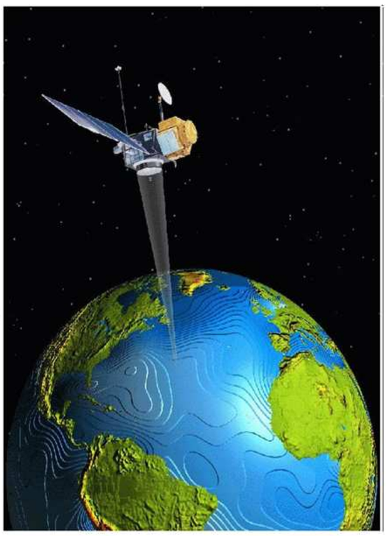
Energy flux, mass flux,



Flux is proportional to the density of flow.

Flux varies by how the boundary faces the direction of flow.

Flux is proportional to the area within the boundary.



Solar Panel Size Estimation

Photo-Voltaics-Assembly (PVA) solar panel → required size A_a

$$A_a = \frac{P_{req}}{P_\delta}$$

With

- A_a =solar array area = array length x array height
  - array height depends on spacecraft height, length may be distributed over multiple wings

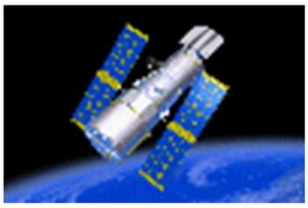
- P_δ is power density yield of the solar array
  - $P_\delta = \eta S$ ~ 100 - 400 W/m^2 @ 1 AU [FSS Table-10.2]
  - $\eta = \eta_1 \eta_2$ is solar panel efficiency (~10-35%)
  - η_1 = PVA efficiency 12-38% (classical monocrystalline Silicon → triple junction solar cells)
  - η_2 = PVA packing density on the panel 90—96%
  - S is solar intensity
  - Astronomical Unit → 1 AU ~ average circular Earth-Sun distance = 149.6×10 km^2

Use of solar cells

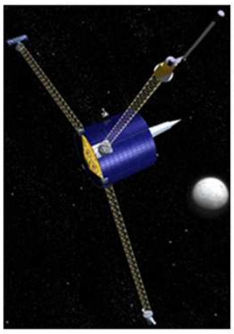
- Body-mounted arrays
  - Planar panels
  - Cylindrical panels (spinning satellites)
- Deployable arrays
  - Rigid panels
  - Flexible panels (blankets)

TUBSat

Orbcomm sat



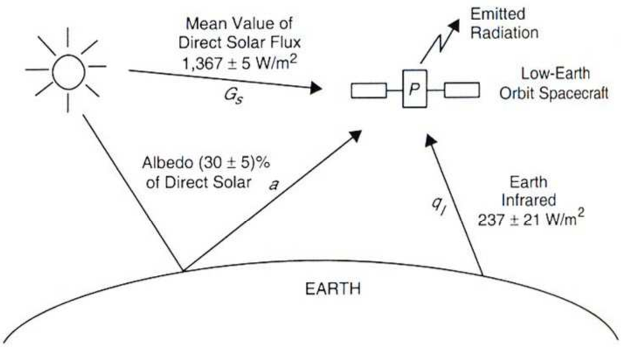
Hubble



Lunarsat

Type of Electrical Power System

Thermal S/S defines the S/C net heat absorption and emitted radiation. As a result eg the temperature of the solar panels and solar cells is defined.



ΔT = temperature change [K]
C = heat capacity
M = mass [kg]
Q = heat exchange [Joule]
T_0 = S/C temperature
T_s = deep space temperature = 4 [K]
α = effective absorption
ε = effective emittance
σ = Stefan-Boltzman constant

$5.67x10^{-8}$ [W/m^2/K^4]

$$\frac{dQ}{dt} = S \cdot \alpha A_\alpha$$

$$\frac{dQ}{dt}\Delta t = CM\Delta T \rightarrow \Delta T = \frac{dQ}{dt}\frac{\Delta t}{CM} = \frac{\alpha S A_\alpha}{CM}\Delta t$$

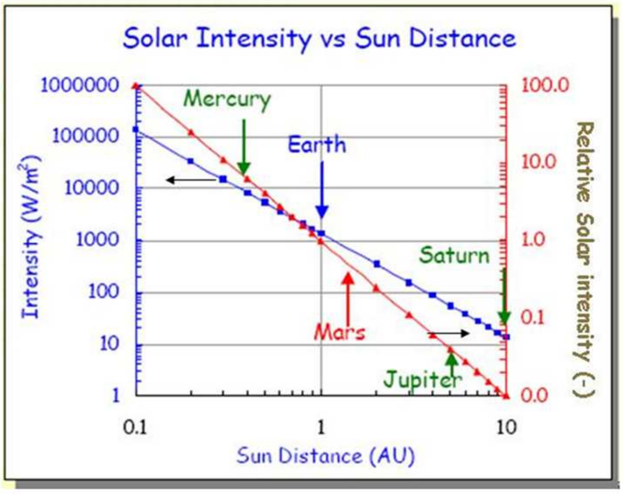
$$Q_{emitted} = \varepsilon A_\varepsilon \sigma T_0^4$$

$$Q_{deep\_space} = \varepsilon A_\varepsilon \sigma (T_0^4 - T_s^4)$$

$$T_s = 4K$$

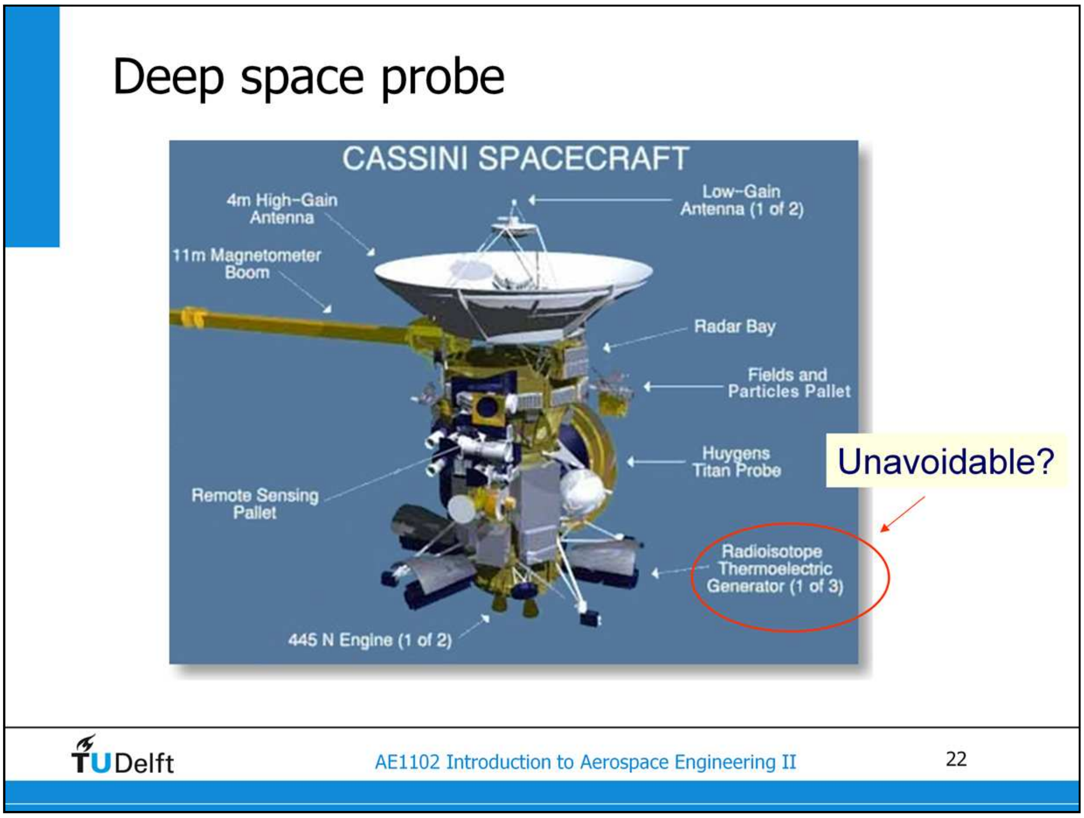
$$\sigma = 5.67x10^{-8}[W/m^2/K^4]$$

Spacecraft environment



Solar intensity varies with distance to Sun (d) squared

$J_s = S = P/(4\pi d^2)$
with $P = 3.856 \times 10^{26}$ W

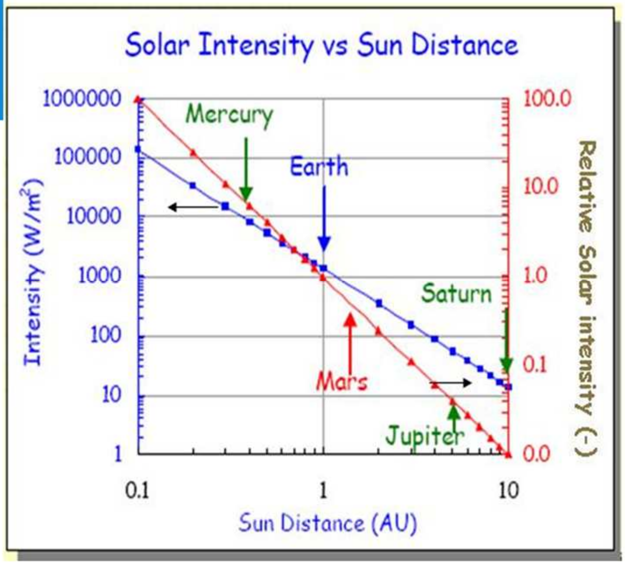


Deep space probe

Unavoidable?

Cassini - Huygens, mission to Saturn and Titan

Heat input

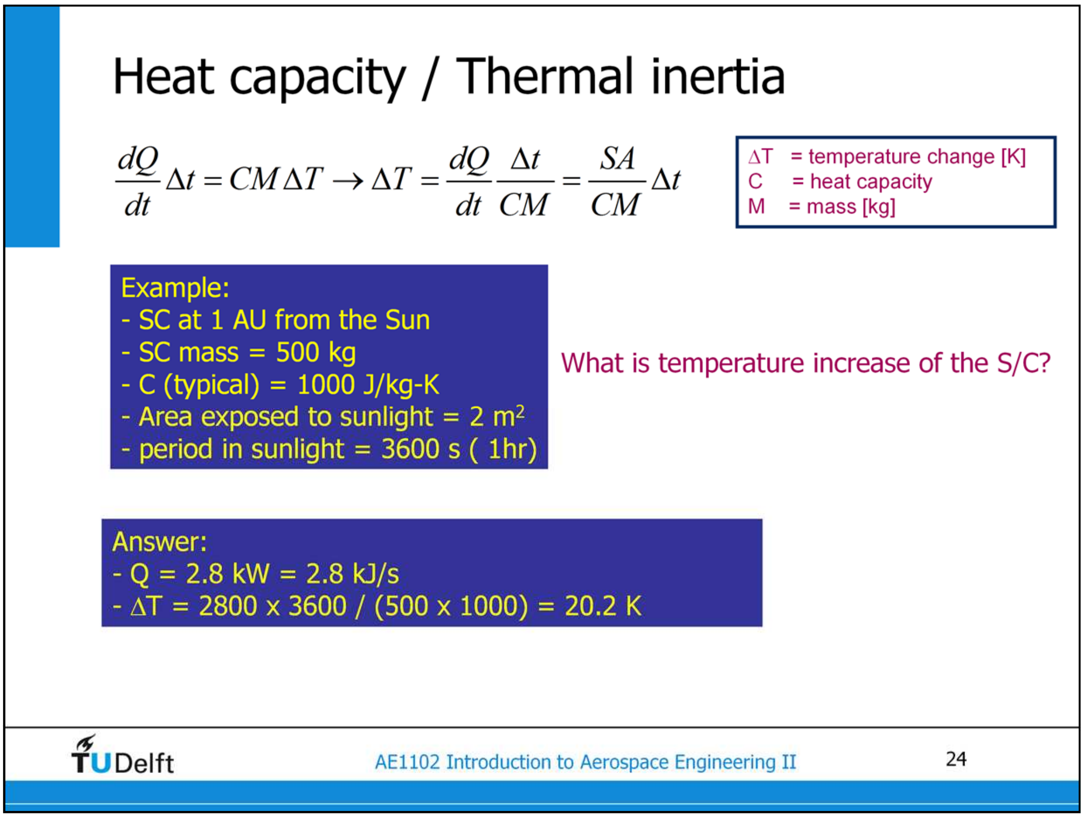


Solar intensity varies with distance to Sun (d) squared

$J_s = S = P/(4\pi d^2)$
with $P = 3.856 \times 10^{26}$ W

$$\frac{dQ}{dt} = S \cdot A$$

Q = solar heat input [Joule]
S = solar intensity [Watt/m^2]
A = sunlit spacecraft surface [m^2]



Heat capacity / Thermal inertia

$$\frac{dQ}{dt}\Delta t = CM\Delta T \rightarrow \Delta T = \frac{dQ}{dt}\frac{\Delta t}{CM} = \frac{SA}{CM}\Delta t$$

ΔT = temperature change [K]
C = heat capacity
M = mass [kg]

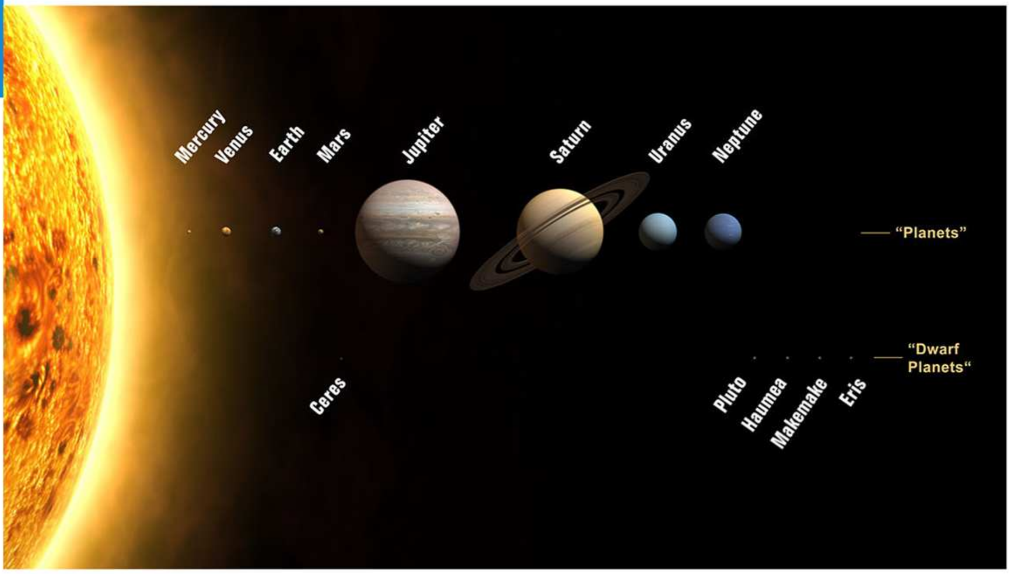
Example:
- SC at 1 AU from the Sun
- SC mass = 500 kg
- C (typical) = 1000 J/kg-K
- Area exposed to sunlight = 2 m^2
- period in sunlight = 3600 s (1hr)

What is temperature increase of the S/C?

Answer:
- Q = 2.8 kW = 2.8 kJ/s
- ΔT = 2800 x 3600 / (500 x 1000) = 20.2 K

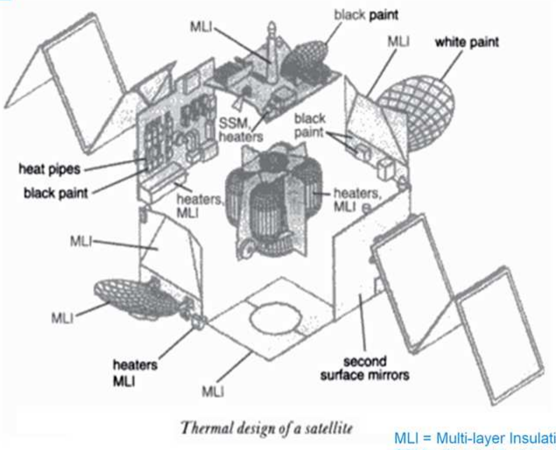
The problem on slide is included in the question pool on Blackboard, be it in a slightly modified way.

How will this value be a Mercury mission?
and for a Jupiter mission?

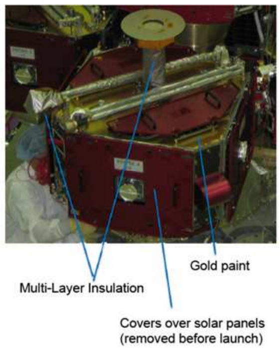


Thermal Control System

- How is S/C thermal control achieved?
  - by means of heaters, cooling devices, insulating materials, coatings, etc.

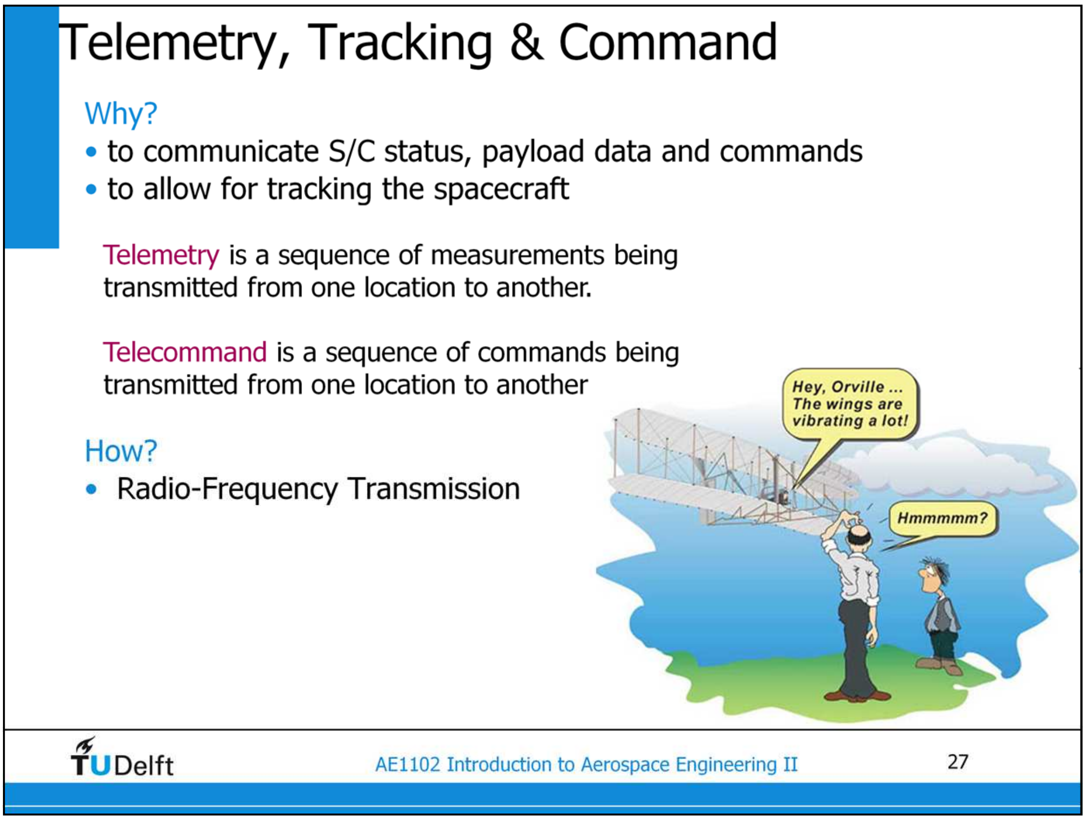


Thermal design of a satellite



Thermal control determines large parts of the spacecraft exterior

MLI = Multi-layer Insulation
SSM = Second Surface Mirrors



Telemetry, Tracking & Command

Why?

- to communicate S/C status, payload data and commands
- to allow for tracking the spacecraft

Telemetry is a sequence of measurements being transmitted from one location to another.

Telecommand is a sequence of commands being transmitted from one location to another

How?

- Radio-Frequency Transmission

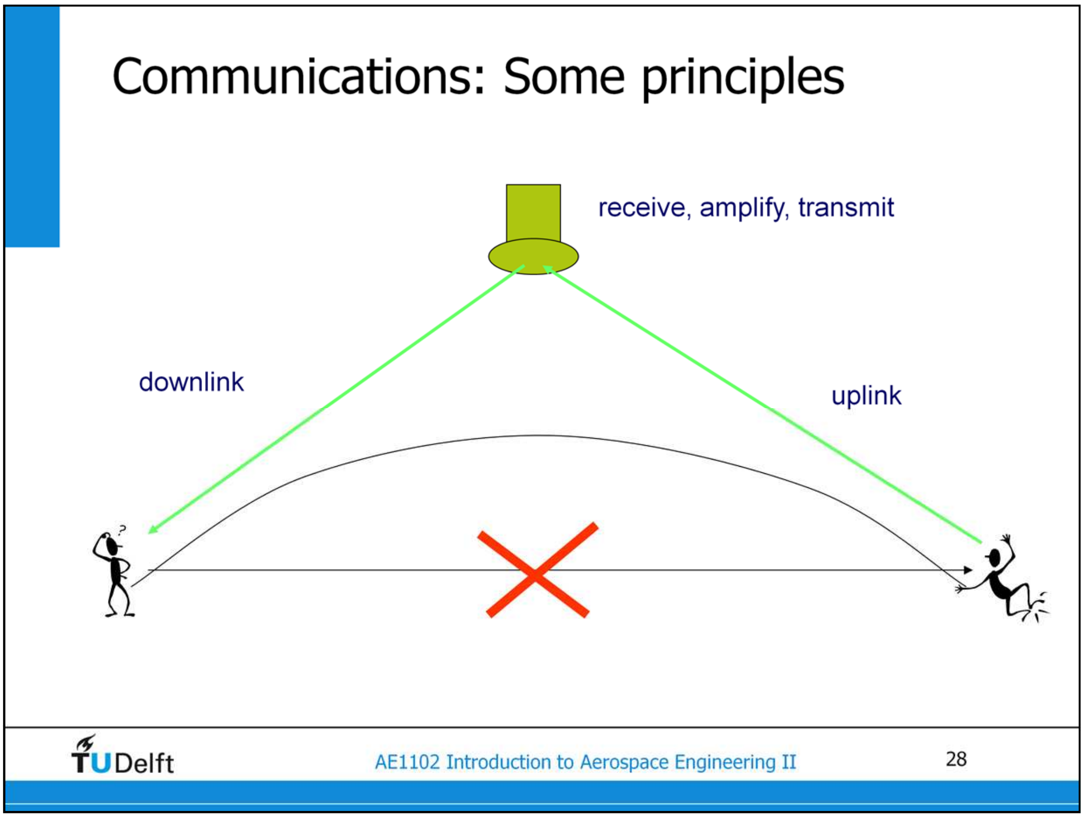
It is essential that a reliable communication link between the ground station and the spacecraft is maintained throughout the satellite's different phases of operation.

During the Launch and Early Orbit Phase (LEOP), ground control sends the required mission commands, such as to fire the booster rockets for orbital correction, to deploy the antenna or solar array, or to fire the apogee boost motors. Some of these operations must happen at precise times, while others can take place during a window of time.

During the lifetime of the mission, which is generally four to ten years, the satellite receives daily the commands required to reconfigure functions according to requirements at the time. Earth observation satellites, such as SPOT or ERS, receive instructions for their next orbits, such as the region of interest of the Earth to observe, the direction of view, or the spectral band to use. A data-relay satellite, such as Artemis or DRS, receives daily commands to inform it of its low Earth orbiting clients; it receives the necessary data for pointing one or more of its antennas towards that satellite and following its path while data relay communication is required.

During launch and early orbit, status data allows ground technicians to check that commands are being carried out correctly, e.g. that boosters are being fired or that the antennas or solar panels are being deployed. Throughout the mission, it enables the mission control centre to survey the 'insides' of the satellite, its configuration, its status, and in the case of failure, it provides the basis for the decisions that have to be made.

Knowing where the spacecraft is (through tracking) allows for timing of commands and to point antennas so that the communication link is (near) optimal.



Communications: Some principles

receive, amplify, transmit

downlink

uplink

LOS-FOV-distance

Formule geven → pointsource

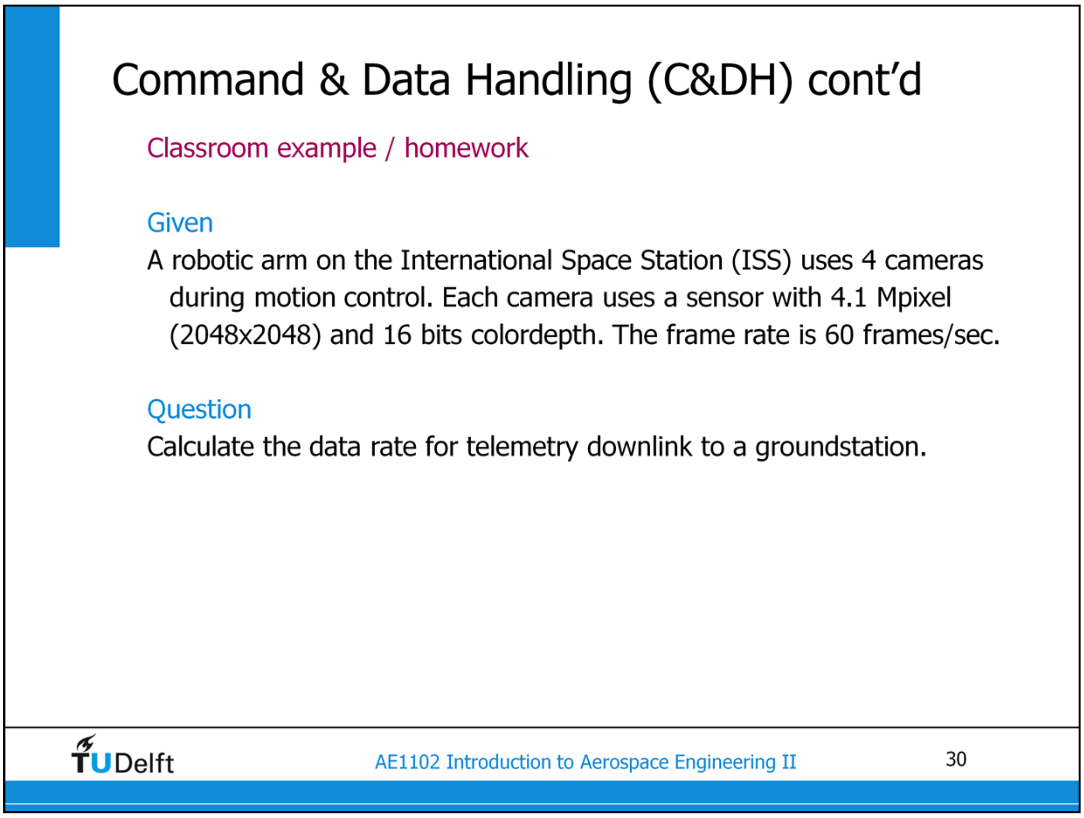
Command & Data Handling (C&DH)

Essentially the brains/intellect and the nerve system of the spacecraft

- Handles sensory information
- Performs decision making
  - authorizes or generates commands
- Commands action,
  - like switching units on/off
- Tracks time
- Acts as memory
- Data Rate (DR) & downlink (kbps, Gbps, kBps, etc)
  - N_{signal} =Number of (video) signals
  - N_{pixels} =Number of picture elements
  - B_{depth}=Color 'depth' per pixel → 8 upto 16 bits = 2^{16}
  - f_{frame} = Number of frames per second (fps) → typically 50-60

Simple system: < 50 cmds & < 200 channels
Complex system: > 50 cmds & > 500 channels

$$DR = N_{signals} N_{pixels} B_{depth} f_{rate}$$



Command & Data Handling (C&DH) cont’d

Classroom example / homework

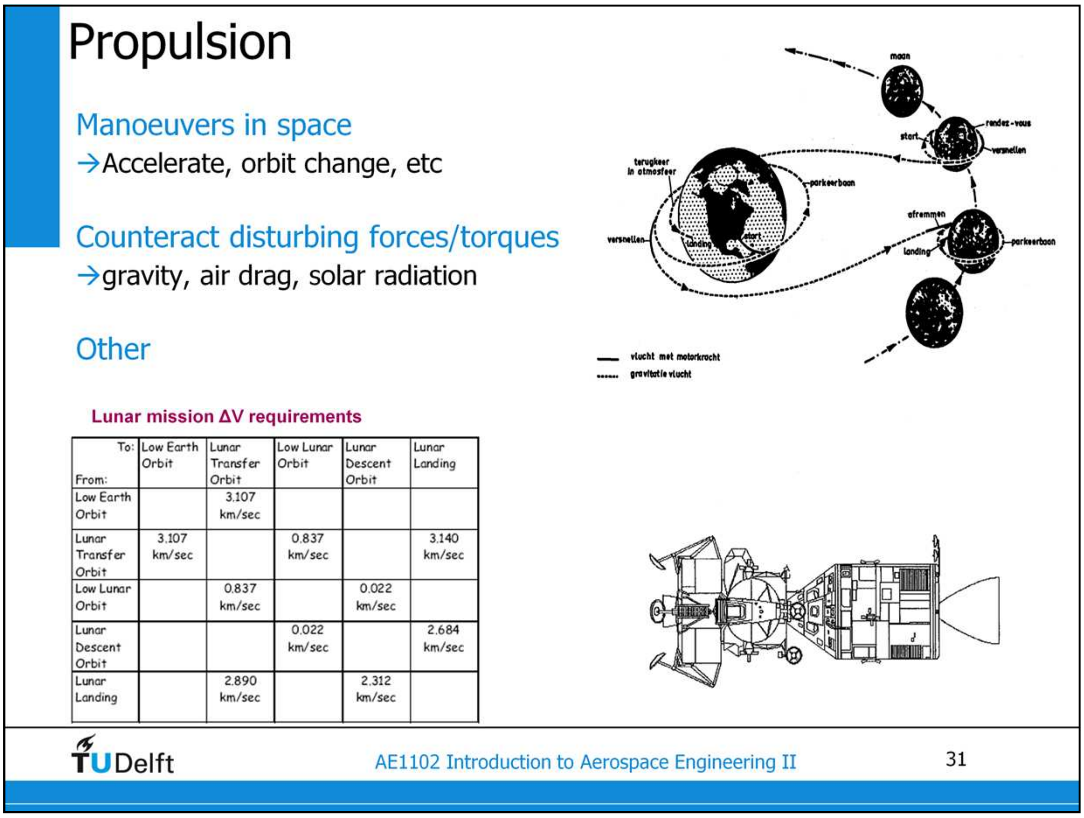
Given

A robotic arm on the International Space Station (ISS) uses 4 cameras during motion control. Each camera uses a sensor with 4.1 Mpixel (2048x2048) and 16 bits colordepth. The frame rate is 60 frames/sec.

Question

Calculate the data rate for telemetry downlink to a groundstation.

4x16x2048x2048x60=4 Gbit/sec



Propulsion

Manoeuvers in space

→Accelerate, orbit change, etc

Counteract disturbing forces/torques

→gravity, air drag, solar radiation

Other

Lunar mission ΔV requirements

To: From:	Low Earth Orbit	Lunar Transfer Orbit	Low Lunar Orbit	Lunar Descent Orbit	Lunar Landing
Low Earth Orbit		3.107 km/sec			
Lunar Transfer Orbit	3.107 km/sec		0.837 km/sec		3.140 km/sec
Low Lunar Orbit		0.837 km/sec		0.022 km/sec	
Lunar Descent Orbit			0.022 km/sec		2.684 km/sec
Lunar Landing		2.890 km/sec		2.312 km/sec	

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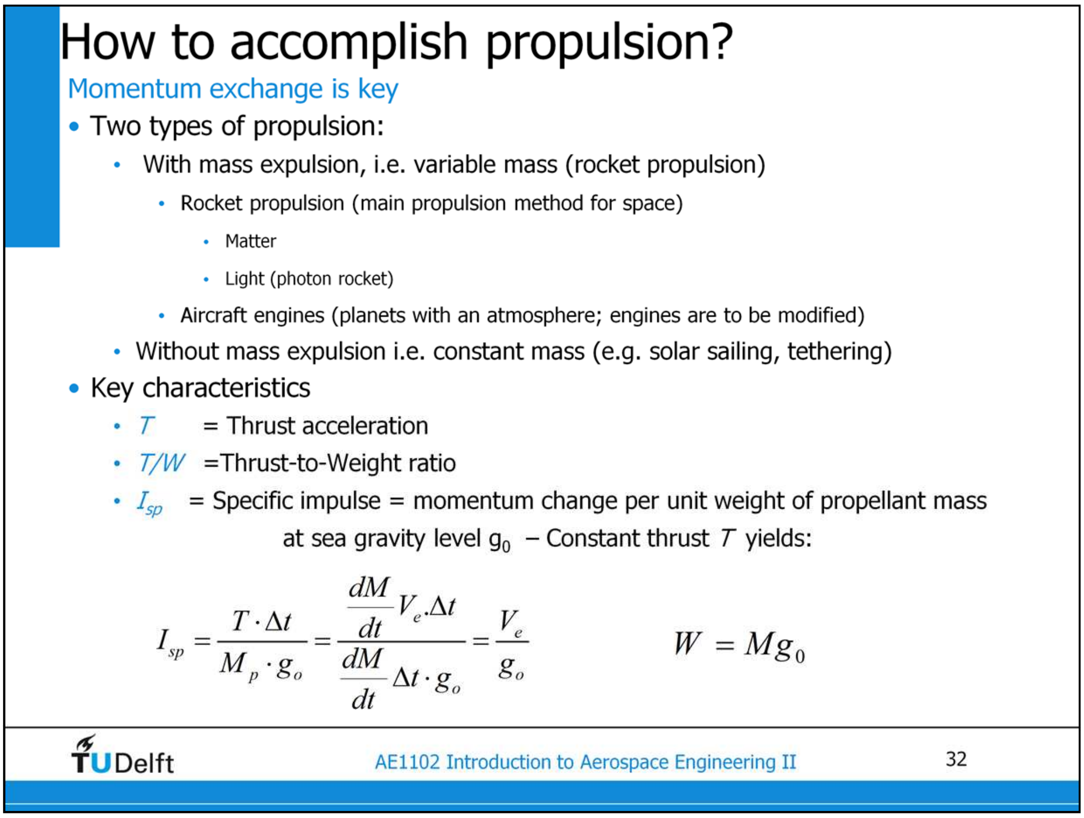
Lunar mission Dv requirements taken from Univ. Of Maryland presentation.

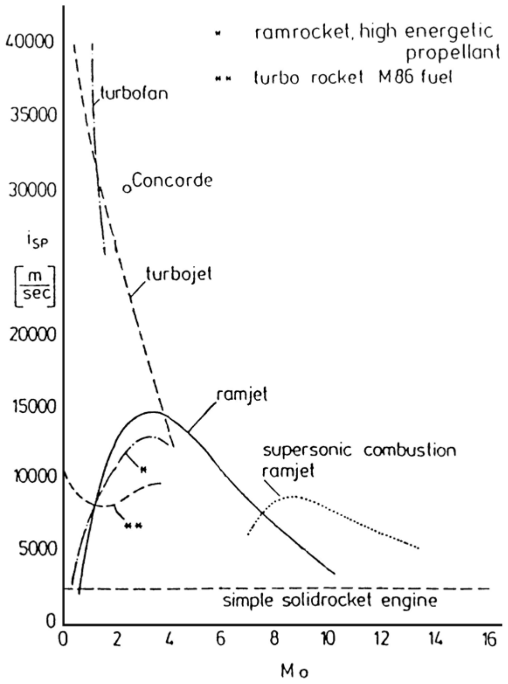


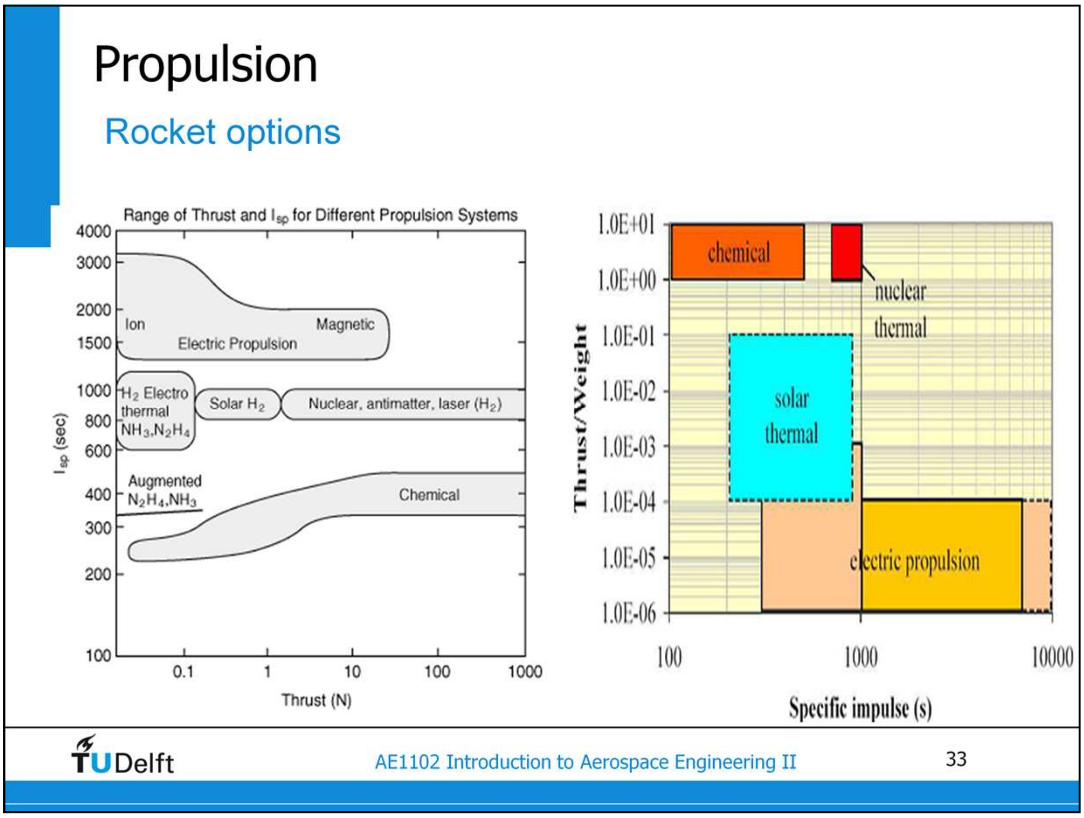
How to accomplish propulsion?

Momentum exchange is key

- Two types of propulsion:
  - With mass expulsion, i.e. variable mass (rocket propulsion)
    - Rocket propulsion (main propulsion method for space)
      - Matter
      - Light (photon rocket)
    - Aircraft engines (planets with an atmosphere; engines are to be modified)
  - Without mass expulsion i.e. constant mass (e.g. solar sailing, tethering)
- Key characteristics
  - T = Thrust acceleration
  - T/W =Thrust-to-Weight ratio
  - I_{sp} = Specific impulse = momentum change per unit weight of propellant mass at sea gravity level g_0 – Constant thrust T yields:

$$I_{sp} = \frac{T \cdot \Delta t}{M_p \cdot g_o} = \frac{\frac{dM}{dt} V_e . \Delta t}{\frac{dM}{dt} \Delta t \cdot g_o} = \frac{V_e}{g_o} \qquad W = Mg_0$$





Propulsion

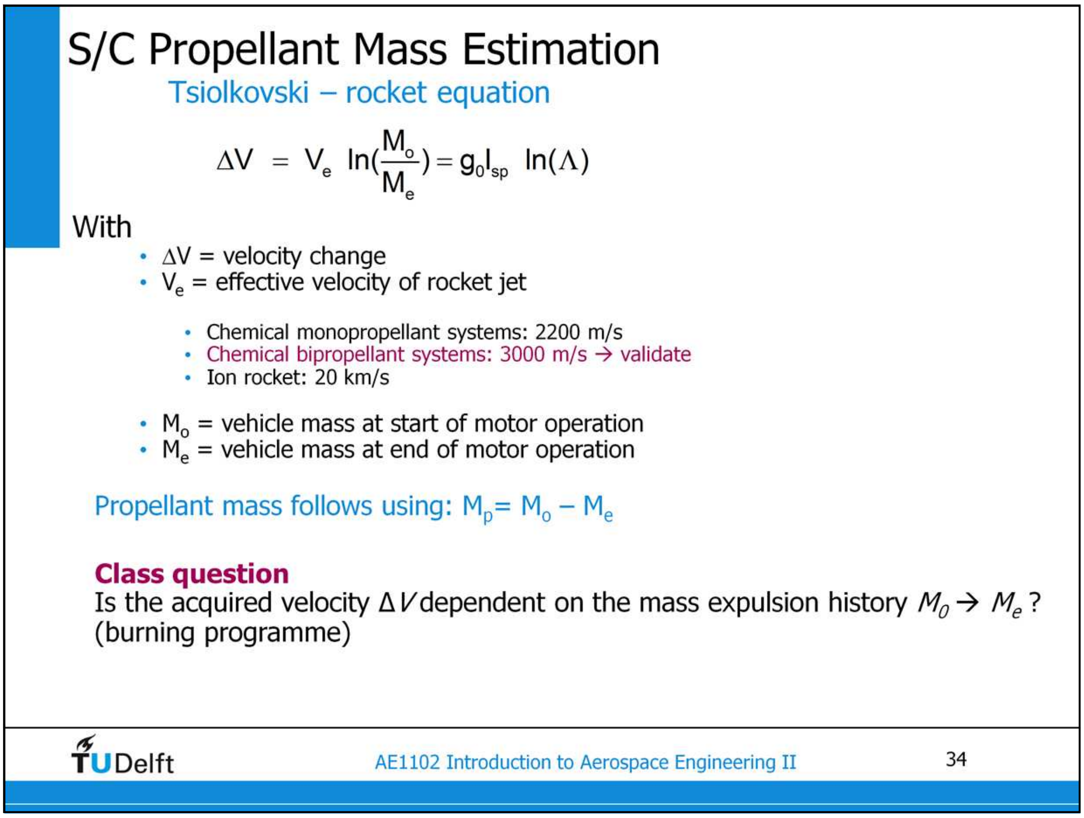
Rocket options

TU Delft AE1102 Introduction to Aerospace Engineering II 33

Comparative performance of some propulsion systems (source: Fortescue & Stark, Fig. 6.2), see also section on propulsion.

Figure on the right taken from the work of H. Leenders.

Propulsion type	Exhaust Velocity (m/s)	Thrust Acceleration (g_o)
Chemical	600 – 4500	0.1-10
Nuclear	< 9000	0.1-10
Ion	5000 - 50000	10^{-3}-10^{-5}
Plasma	5000 - 20000	10^{-3}-10^{-5}
Resistojet	1000-4000	10^{-3}-10^{-5}



S/C Propellant Mass Estimation

Tsiolkovski – rocket equation

$$\Delta V = V_e \ln\left(\frac{M_o}{M_e}\right) = g_0 I_{sp} \ln(\Lambda)$$

With

- ΔV = velocity change
- V_e = effective velocity of rocket jet
  - Chemical monopropellant systems: 2200 m/s
  - Chemical bipropellant systems: 3000 m/s → validate
  - Ion rocket: 20 km/s
- M_o = vehicle mass at start of motor operation
- M_e = vehicle mass at end of motor operation

Propellant mass follows using: $M_p = M_o - M_e$

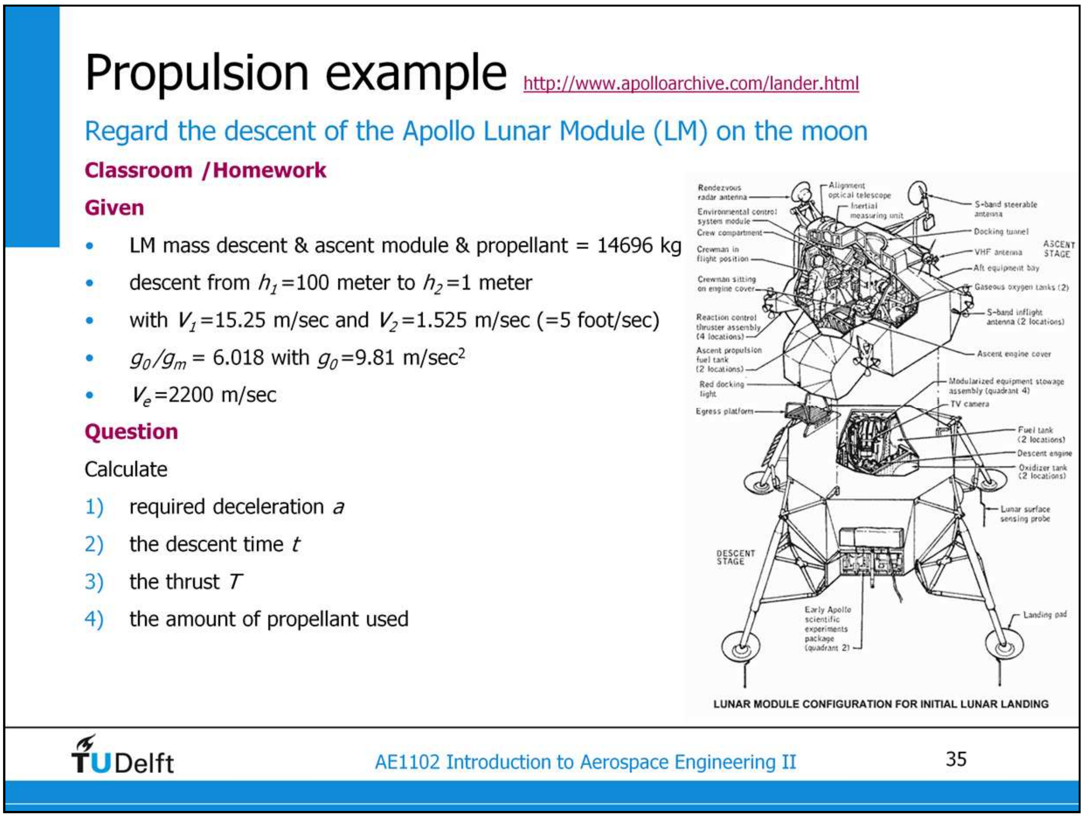
Class question

Is the acquired velocity ΔV dependent on the mass expulsion history $M_0 \rightarrow M_e$? (burning programme)

Tsiolkovsky’s equation, sometimes also referred to as the “Rocket equation”, was first derived by Konstantin Tsiolkovsky in 1895 for straight-line rocket motion with constant exhaust velocity. Later it was shown that it is also valid for elliptical trajectories with only initial and final impulses (impulsive shot).

For details on the derivation of the rocket equation, you are referred to the Section on “Launch vehicle trajectories” in AE1-801.

Empty vehicle mass includes payload mass, structure subsystem mass, propulsion subsystem mass as well as the mass of all the other subsystems. Empty vehicle mass sometimes is also referred as dry mass. In practice, empty mass differs from dry mass in that empty mass also includes residual propellant mass (if any).



Propulsion example http://www.apolloarchive.com/lander.html

Regard the descent of the Apollo Lunar Module (LM) on the moon

Classroom /Homework

Given

- LM mass descent & ascent module & propellant = 14696 kg
- descent from h_1=100 meter to h_2=1 meter
- with V_1=15.25 m/sec and V_2=1.525 m/sec (=5 foot/sec)
- g_0/g_m = 6.018 with g_0=9.81 m/sec^2
- V_e=2200 m/sec

Question

Calculate

1) required deceleration a
2) the descent time t
3) the thrust T
4) the amount of propellant used

LUNAR MODULE CONFIGURATION FOR INITIAL LUNAR LANDING

Newton’s second law

Change of momentum and exchange of momentum is key for both rocket operation and an active ADCS operation. For these impuls conservation for a closed system yields the formula given in the formula sheet addendum. In delta (Δ) form the formula is applicable to limited time intervals as was shown in the Moonlander descent example during the lectures.

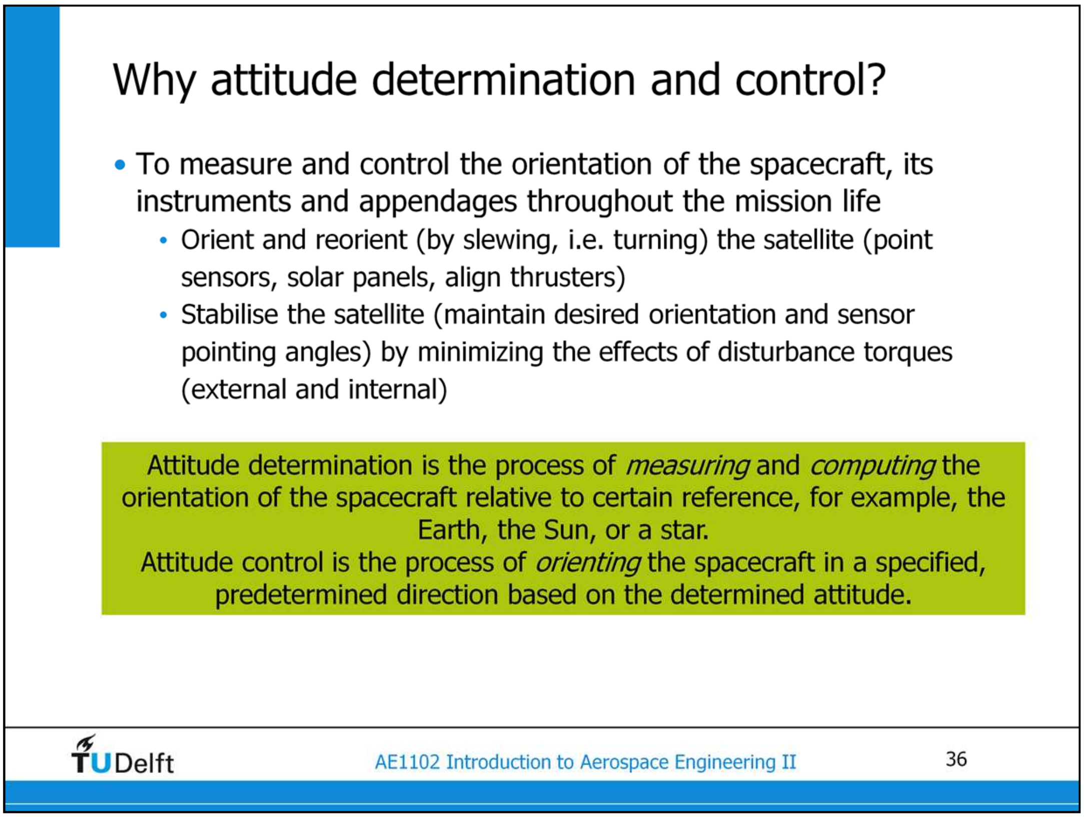
Answers to the problem above (to be worked out and checked using the formulas given) are:

(1) 1.163 m/sec2

(2) 11.8 sec

(3) 41 kN

(4) 220.2 kg

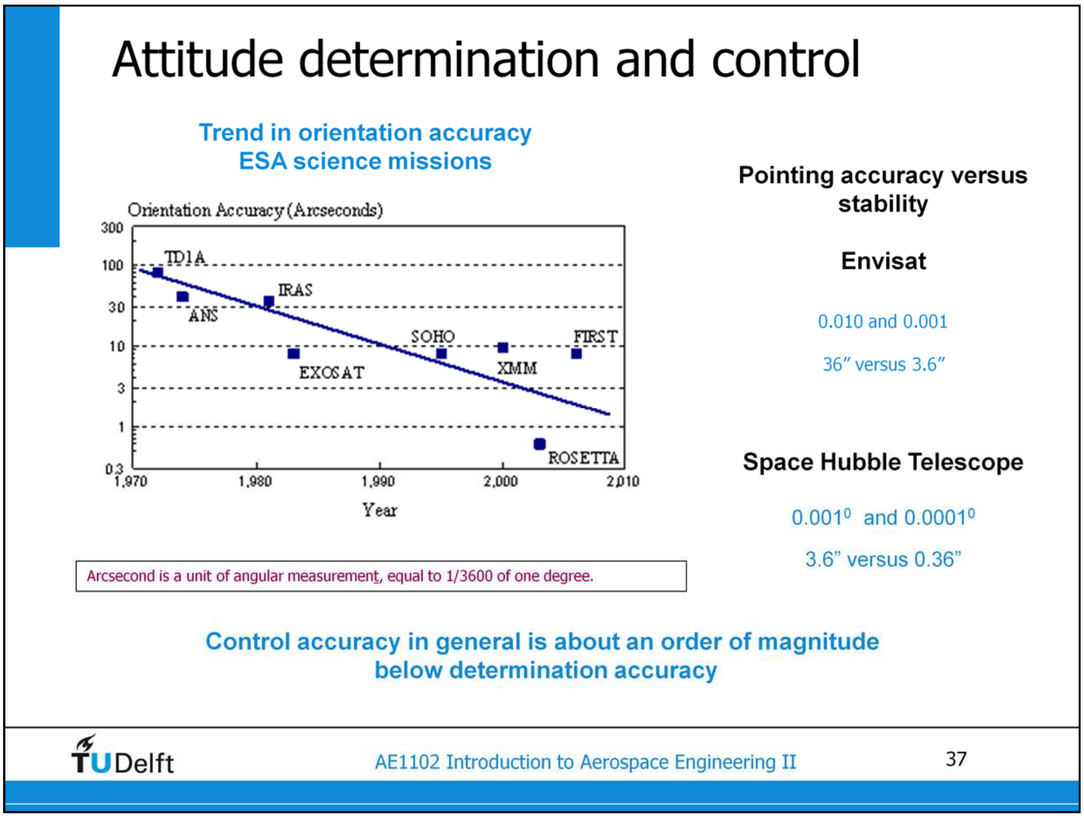


Why attitude determination and control?

- To measure and control the orientation of the spacecraft, its instruments and appendages throughout the mission life
  - Orient and reorient (by slewing, i.e. turning) the satellite (point sensors, solar panels, align thrusters)
  - Stabilise the satellite (maintain desired orientation and sensor pointing angles) by minimizing the effects of disturbance torques (external and internal)

Attitude determination is the process of *measuring* and *computing* the orientation of the spacecraft relative to certain reference, for example, the Earth, the Sun, or a star.
Attitude control is the process of *orienting* the spacecraft in a specified, predetermined direction based on the determined attitude.

Slew maneuver is a maneuver that intends to reorient the vehicle.



Attitude determination and control

Trend in orientation accuracy
ESA science missions

Arcsecond is a unit of angular measurement, equal to 1/3600 of one degree.

Pointing accuracy versus stability

Envisat

0.010 and 0.001

36″ versus 3.6″

Space Hubble Telescope

0.001^0 and 0.0001^0

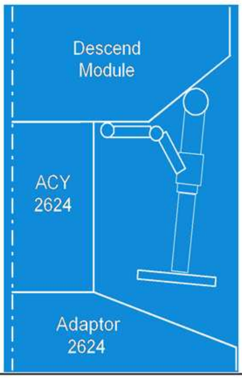
3.6" versus 0.36"

Control accuracy in general is about an order of magnitude below determination accuracy

Slew maneuver is a maneuver that intends to reorient the vehicle.

Other Systems

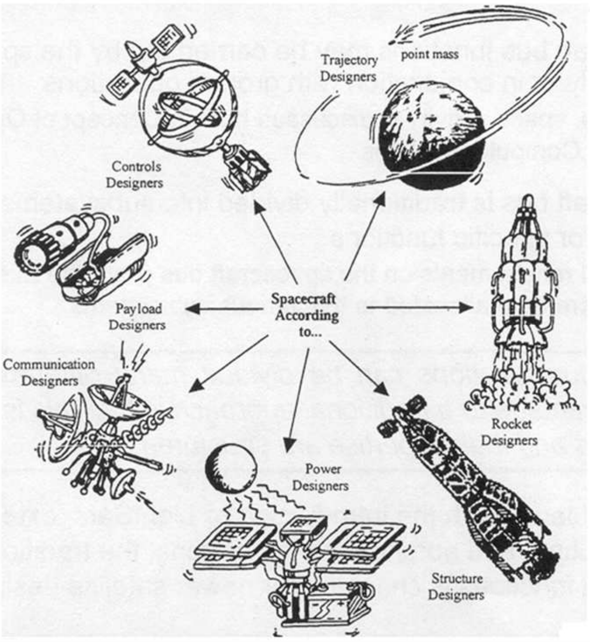
- Environmental and life support
- Destruction system
- Lander system
  - parachute, landing gear, balloon
- Recovery equipment
- Aerodynamic system
- Abort system
- Etc

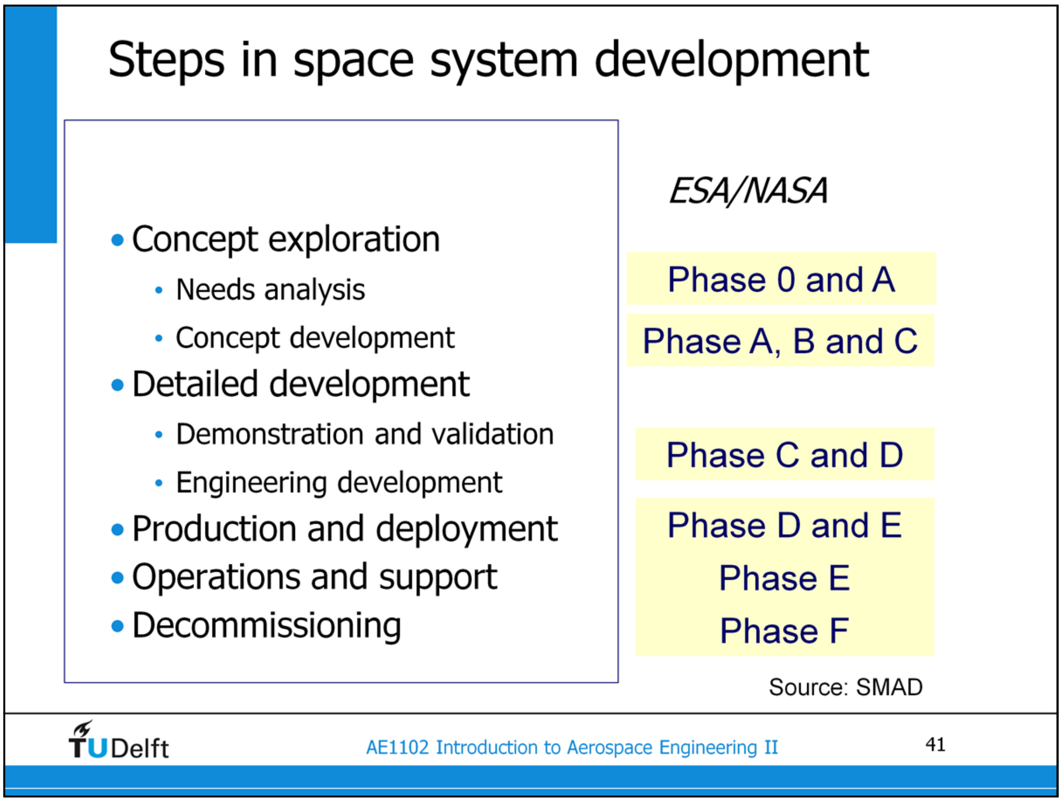


Spacecraft Subsystem functions

Subsystem	***Principal functions***	***Other names***
Propulsion	Adjust orbit and attitude, dump angular momentum	Reaction Control System (RCS)
Guidance, Navigation & Control (GNC)	Determine and control orbit	Orbit Control System (OCS)
Attitude Determination & Control (ADCS)	Determine and control attitude, point and manoeuver the spacecraft, manage angular momentum	Attitude Determination and Control System (ADCS) or Control System
Communications	Communicate with ground, support spacecraft tracking	Tracking, Telemetry, Command (TT&C)
Command & Data Handling (C&DH)	Process commands, perform data processing/ formatting, provide computing power	Spacecraft Computer System Spacecraft Processor
Thermal	Control equipment temperature	Environmental Control System
Power	Generate and distribute power	Electric Power Subsystem (EPS)
Structures & Mechanisms	Provide structural integrity, provide motion possibility to structural parts	Structure Subsystem

What we don't want to end up with!





Steps in space system development

- Concept exploration
  - Needs analysis
  - Concept development
- Detailed development
  - Demonstration and validation
  - Engineering development
- Production and deployment
- Operations and support
- Decommissioning

ESA/NASA

Phase 0 and A

Phase A, B and C

Phase C and D

Phase D and E

Phase E

Phase F

Source: SMAD

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Any mission, whether communications, navigation, weather, military, or earth observation, goes through a number of development steps. In Space Mission Analysis and Design (SMAD), 2nd. Ed., by Larson and Wertz, 1992, 4 development steps are distinguished:

Concept exploration: Results in a broad *definition* of the *space mission* and the various elements that work in unison to realize the mission.

Detailed development: Results in a detailed definition of the system components and, in larger programs, development of test hardware and/or software.

Production and deployment: The construction of the ground and flight hardware and launch of the (constellation of) satellite(s).

Operations and support: The day-to-day operations of the space system, its maintenance and support, and finally its deorbit or recovery at the end of the mission life.

Space mission concept exploration is just one step in the development of a space system/mission.

Other space development organisations, like ESA, sometimes use a different phasing, but the essential steps are in essence the same.

Definition of steps

Concept exploration

- Results in a broad *definition* of the element to be designed and the various elements it has to interact with

Detailed design & development

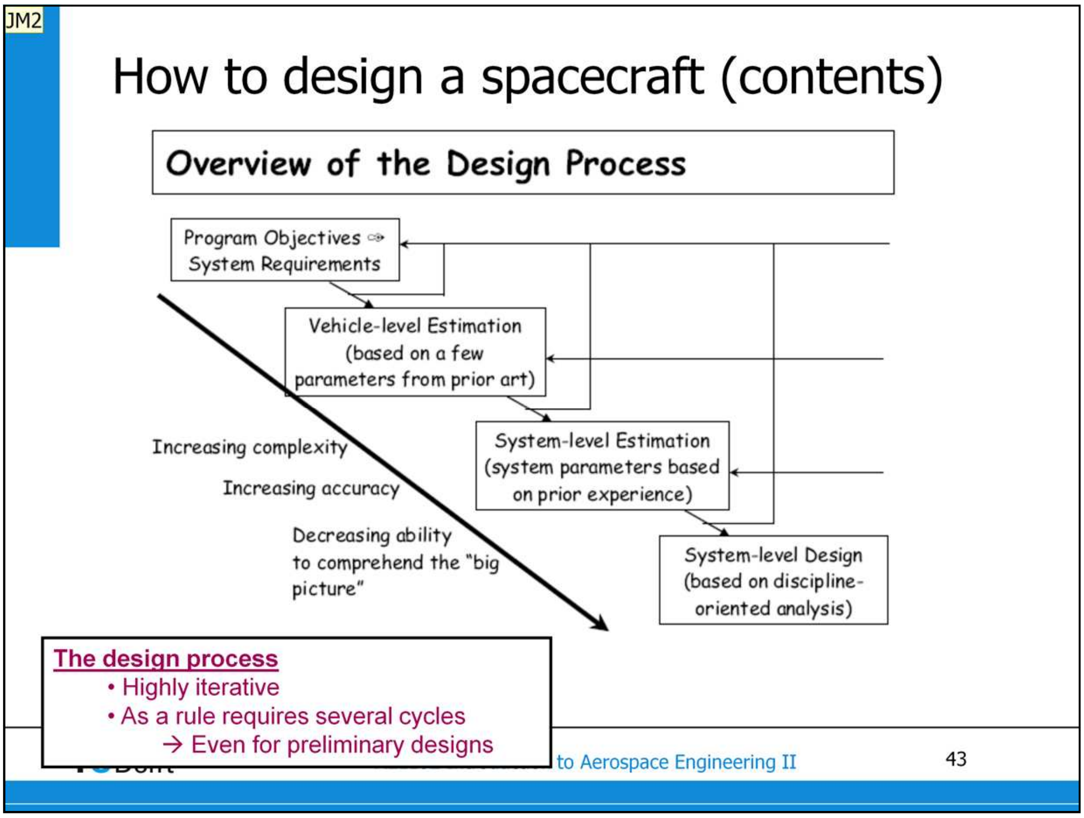
- Results in a detailed definition of the system components and, in larger programs, development of test hardware and/or software

Production and deployment

- The construction of the ground and flight hardware and launch of the (constellation of) satellite(s)

Operations and support

- The day-to-day operations of the system, its maintenance and support, and finally its de-orbit or recovery at the end of the mission life



JM2

How to design a spacecraft (contents)

Overview of the Design Process

The design process

- Highly iterative
- As a rule requires several cycles
  - → Even for preliminary designs

Overview of bus design and sizing (contents)

- About spacecraft
- The design process
- Prepare list of spacecraft bus requirements and constraints
- Develop baseline spacecraft design
  - Generate configuration drawings
  - Generate mass, size, and cost estimates, etc.
- Design of subsystems
- Establish budgets for spacecraft bus power, mass, size, etc.
- Review

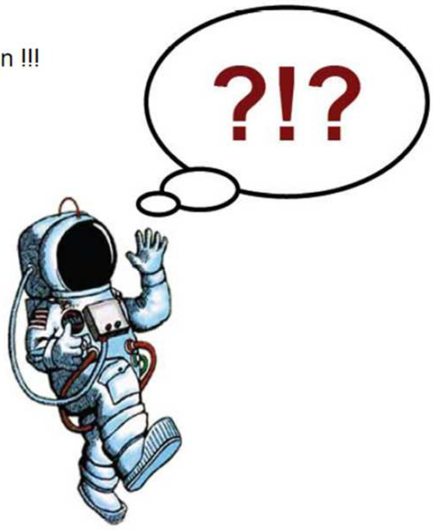
Figure taken from University of Maryland presentation on propulsion system design, by David L. Akin, 2002.

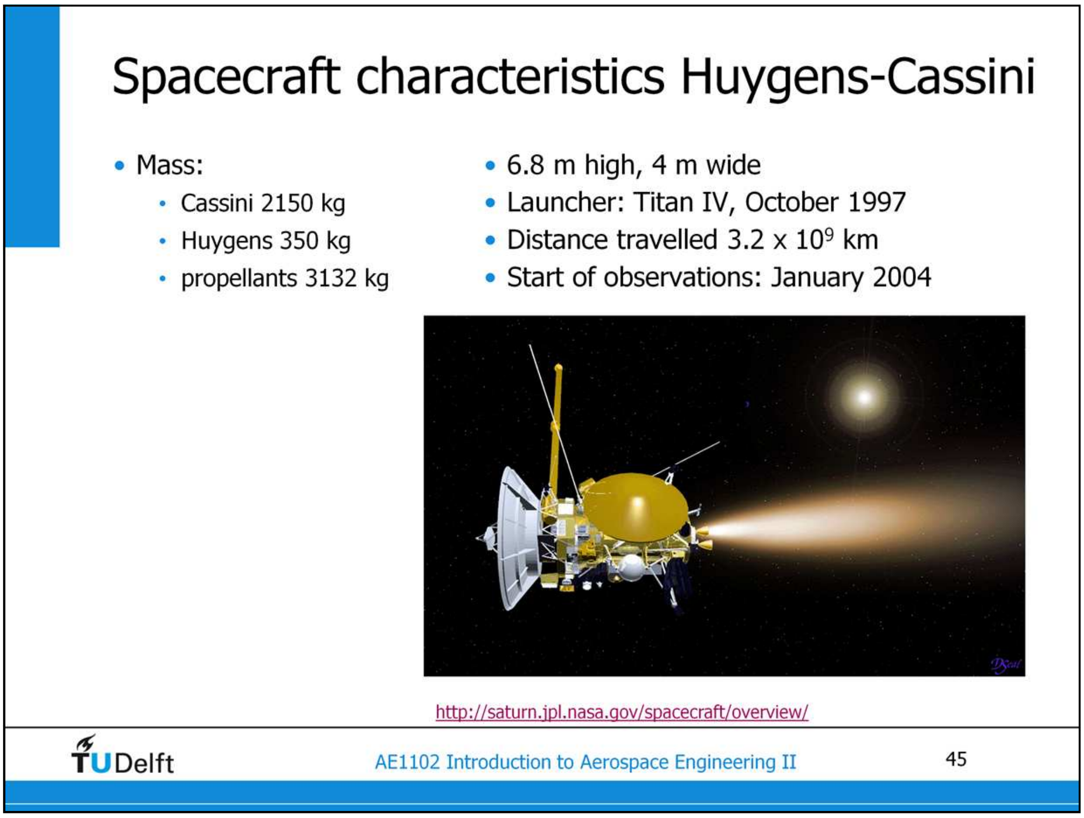
Slide 43

JM2 Kuiper; 18-9-2009

Questions?

- Thank you for your attention !!!





Spacecraft characteristics Huygens-Cassini

- Mass:
  - Cassini 2150 kg
  - Huygens 350 kg
  - propellants 3132 kg
- 6.8 m high, 4 m wide
- Launcher: Titan IV, October 1997
- Distance travelled 3.2 x 10^9 km
- Start of observations: January 2004

http://saturn.jpl.nasa.gov/spacecraft/overview/

In many respects Saturn's largest moon, Titan, is one of the most Earth-like world we have found to date. With its thick atmosphere and organic-rich chemistry, Titan resembles a frozen version of Earth, several billion years ago, before life began pumping oxygen into our atmosphere.

Titan is of great interest to scientists because it has a substantial, active atmosphere and complex, Earth-like processes that shape its surface. The moon is enveloped by an orange haze of naturally produced photochemical smog that frustratingly obscured its surface prior to Cassini's arrival. Since 2004, the spacecraft's observations have taken the study of this unique world into a whole new dimension.

Cassini has revealed that Titan's surface is shaped by rivers and lakes of liquid ethane and methane (the main component of natural gas), which forms clouds and occasionally rains from the sky as water does on Earth. Winds sculpt vast regions of dark, hydrocarbon-rich dunes that girdle the moon's equator and low latitudes. Volcanism may occur as well, but with liquid water as the lava.

On its journey to Saturn, Cassini carried the European-built Huygens probe. On Jan. 14, 2005, Huygens achieved humankind's first landing on a body in the Outer Solar System when it parachuted through Titan's murky skies. Huygens took measurements of atmospheric composition and wind speeds during its decent, along with an incredible series of images showing telltale patterns of erosion by flowing liquid. The probe came to rest on what appeared to be a floodplain, surrounded by rounded cobbles of water ice.

As the Cassini Equinox Mission progresses, the spacecraft will monitor Titan's atmosphere and surface for signs of seasonal change. The spacecraft's radar and camera systems will continue to peer through the haze, expanding our high resolution maps of the surface. And scientists will eagerly await new data that could confirm the presence of a liquid ocean beneath the giant moon's surface.

The exploration of this amazing place is just beginning. Frigid and alien, yet also remarkably similar to our own planet, Titan is a new world – revealed before our very eyes by the Cassini and Huygens spacecraft.

‘Moore’s Law’ 1971-2008

CPU transistor count doubles every two years