AE1222-II: Aerospace Design & Systems Engineering Elements I

Part: Spacecraft (bus) design and sizing

Learning goal
The student shall be able to conduct all steps necessary to perform a conceptual design of a space vehicle and its (sub)systems

Learning objectives
The student shall be able to

- Describe the spacecraft vehicle design process
- Generate a spacecraft requirements list
- Perform spacecraft vehicle sizing with an accuracy fit for conceptual design purposes
- Perform spacecraft (sub)system sizing
- Develop a simple spacecraft configuration
- Perform budgeting
- List limitations related to the methods used

Prerequisites
Student should be able to:

- ➢ Calculate orbital velocity, (maximum) eclipse times, orbital period of some common orbits, ground contact time (overhead pass), ground velocity, gravitational force;
- ➢ Generate a delta-v (Δv) budget;
- ➢ Calculate atmospheric density, magnetic field strength, and gravitational force in relation to location in space.

Study material

- This reader + course slides.

Revisions table

Version	Changes implemented
2012/2013	• Worked in errata list year 2011/2012; • Changed course codes to reflect changes as induced by faculty; • Added overview of chapters to be studied from the textbook Spacecraft systems Engineering, by Fortescue et al in chapter 1; • Extended the number of examples given in text; • Changed layout of examples to make them more visible in text; • Various relations given implicitly in text have been made explicit. Total relations count is 110 (up from 102); • Total page count is 190 pages (up from 182 pages); • Reference to Blackboard / Maple TA for practicing problems has been added to text; • Revisions table added.
2013/2014	• Adapted chapter on budgeting. Moved discussion on margins from chapter 3 to chapter 4; • Removed references to Spacecraft Systems Engineering as the course text book; • Restructured some chapters; • Extended text on spacecraft requirements generation by adding text on spacecraft requirements related to other mission elements (from slides); • Modified chapter 3; • Updated various estimation relationships given in appendix C;

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Abbreviations

ADCS	Attitude Determination and Control System
AOCS	Attitude & Orbit Control System
ARD	Ariane Return Demonstrator
ASM	Attitude Safety Module
ATV	Automated Transfer Vehicle
AU	Astronomical Unit
BOL	Begin Of Life
bps	bit per second
C&DH	Command and Data Handling
CDR	Critical Design Review
CFRP	Carbon Fiber Reinforced Plastic
CM	Command Module
CoM	Centre of Mass
CPU	Central Processor Unit
C^3	Communications, Command and Control
DACU	Data Acquisition and Control Unit
DC	Direct Current
DL	Design Life
DOD	Depth Of Discharge
DR	Data Rate
EIRP	Effective Isotropic Radiated Power
EM	Electro-Magnetic
EOM	End of Mission
EOL	End Of Life
EPS	Electrical Power (generation) System
ESA	European Space Agency
FoS	Factor of Safety
FoV	Field of View
FRR	Flight Readiness Review
FSS	Fixed Satellite Services
FVC	Fine Velocity Control
FY	Fiscal Year
GEO	Geostationary Earth Orbit
GNC	Guidance Navigation and Control
GPS	Global Position System
GSFC	Goddard Space Flight Centre
HGA	High Gain Antenna
HK	House-Keeping
ID	IDentifier
IR	InfraRed
ITAR	International Traffic in Arms Regulations
ITU	International Telecommunications Union
(A/P)KM	(Apogee/Perigee) Kick motor
LEO	Low Earth Orbit
LGA	Low Gain Antenna
LOC	Lines Of Code
LVA	Launch Vehicle Adapter ; Large Velocity Actuator
LoS	Line of Sight
MER	Mass Estimation Relationship
MEO	Medium Earth Orbit
MIPS	Mega Instruction Per Second
MMD	Mean Mission Duration

MMOI	Mass Moment of inertia
MLE	Most Likely Estimate
MLI	Multi-Layer Insulation
MSG	Meteosat Second Generation
NASA	National Aeronautics and Space Administration
NORAD	North American Aerospace Defense Command
OBC	On-Board Computer
OBDH	On-Board Data Handling
OCS	Orbit Control System
OSR	Optical Surface Reflectors
PDR	Preliminary Design Review
PLM	Payload Mass
PMS	Propellant Management System
QSL	Quasi Steady Load
RCS	Reaction control System
RF	Radio Frequency
rpm	rounds per minute
RTG	Radio-Isotope Generator
RX	Receiver
SAR	Synthetic Aperture Radar
S/C	Spacecraft
SRB	Solid Rocket Booster
(S)SD	(Sample) Standard Deviation
SSE	Spacecraft Systems Engineering (the course text book)
SE(E)	Standard Error (of Estimate)
SLOC	Source Lines Of Code
SM	Service Module
TBC	To Be Confirmed
TC	Tele-Command
TCS	Thermal Control System
TDRSS	Tracking and Data Relay Satellite System
TM	Telemetry
TT&C	Telemetry, Tacking & Command
TX	Transmitter
UHF	Ultra High Frequency
UV	Ultra-Violet
VEB	Vehicle Equipment Bay
VDM	Vehicle Dry Mass
VHF	Very High Frequency

List of Symbols

<u>Roman</u>

a	Acceleration
A	Area
B	Earth magnetic field strength, bandwidth
c	Velocity of light
C	Cost, heat capacity (specific heat), Power received
C_D	Drag coefficient
CF	Compression factor
d	Diameter
D	Distance, dipole moment
DR	Data rate
e	Error
E	Young's modulus, energy
f	Frequency
F	Force, failure probability
g_o	Gravitational acceleration at sea level
g	Gravitational acceleration
G	Gain factor
H	Angular momentum
I	Area or mass moment of inertia, impulse, current
I_d	Inherent degradation
I_{sp}	Specific impulse
J	Safety factor
J_s	Solar intensity
k	Spring constant, Boltzmann constant
L	Length, loss factor
L_d	Life degradation
m	Mass flow rate
M	Mass, spacecraft residual dipole
n	number
N	number of loops in coil, noise
P	Power, probability
q	Heat flux
Q	Heat
$\dot{Q}$	Heat flow
r	radius, arm
R	Risk, reliability
S	Surface area
SR	Sampling rate
t	Time, thickness
T	Temperature, thrust, torque
v	Velocity
V	Volume (related to either physical dimensions or data volume)
w	(effective) exhaust velocity
W	Weight, work

Greek

α	Absorbtivity, Specific mass
δ	Deflection, structural mass to total mass ratio
Δ	Delta
ε	Emissivity
λ	Wavelength, failure rate
Λ	Mass ratio
η	Efficiency
θ	Incidence angle of solar radiation, rotation angle
τ	Transmissivity
ρ	Reflectivity, mass density
σ	Stefan Boltzmann constant
ω	Rotational velocity

Subscripts

a	array, albedo
ant	antenna
b	body, burn (in burn time), bit
e	electric, empty
f	final
fc	fuel cell
o	initial
rec	recorder
rw	reaction wheel
s	sun, system
sp	specific
t	thermal
tr	torque rod
T	thrust
W	power system

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1 Introduction

According to the records, each year about 100 spacecraft are launched into space [Jane's]. In 2008 about 68 launches took place orbiting about 91 spacecraft and in 2012 about 75 launches took place orbiting about 150 spacecraft.

What exactly is a spacecraft (S/C)? A S/C is a vehicle or device designed for spaceflight.

From literature, like [Jane's], we learn that many different spacecraft exist. We distinguish:

- (Earth) satellite - man-made equipment that orbits around the Earth or the Moon
- Orbiter spacecraft: A spacecraft designed to travel to a distant planet or moon and enter orbit. It must carry a substantial propulsive capability to decelerate it at the right moment to achieve orbit insertion. An orbiter spacecraft must endure periods during which it is shaded from sunlight, thus it must be resistant to extreme thermal variation and will require power storage capacity if equipped with solar panels. Examples of orbiter spacecraft include Magellan, Galileo and Mars Global Surveyor.
- Flyby spacecraft - This group of spacecraft conducted initial exploration of solar system. They follow a continuous solar orbit or escape trajectory so as not be captured in a planetary orbit.
- (Re-)entry vehicle, module - The part of a spacecraft (or missile) that (re-)enters Earth's atmosphere or the atmosphere of some other celestial body
- Lander - A space vehicle that is designed to land on a celestial body (planet/moon)
- Ascender/launcher – A space vehicle that is designed for launching a payload from a planetary surface into space (this category of vehicles is dealt with separately in this course, see later)
- Space probe - An unmanned spacecraft that undertakes a mission beyond Earth's orbit.
- Rover Spacecraft – A semi-autonomous roving vehicle that is steerable from Earth.
- Service module or kick stage – A vehicle that transports other spacecraft in space.
- Spaceship, starship - a spacecraft designed to carry a crew into interstellar space
- Manned spacecraft - A piloted spacecraft designed to carry astronauts into space. Unlike an unmanned probe, it requires a crew compartment and life support systems. Manned spacecraft are either reusable, such as the Space Shuttle, or designed for one time use, such as Soyuz. The latter type is generally modular, such as consisting of a reentry module which houses the crew and a service module which contains propulsion, power supply and life support. Only the reentry module returns to Earth.
- Robotic spacecraft – Essentially an unmanned spacecraft.

A S/C typically consists of one or more payloads/instruments, like direct and/or remote[1] sensing instruments and/or telecommunications transmitters and receivers, and a services section (i.e. the spacecraft bus or platform) that supports the payload, for instance by providing electric power, controlling the attitude of the instruments and the on board temperatures and protects it against the harsh space environment and other threats if necessary. Table 1 describes some typical spacecraft and their main characteristics.

[1] Direct-sensing instruments interact with phenomena in their immediate vicinity, and register characteristics of them. Remote-sensing instruments record characteristics of objects at a distance, sometimes forming an image by gathering, focusing, and recording reflected light from the Sun, or reflected radio waves emitted by the spacecraft.

Table 1: Spacecraft and some characteristic data

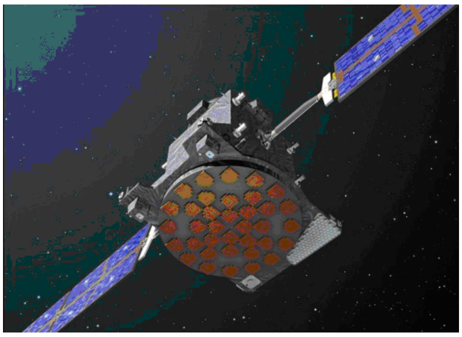
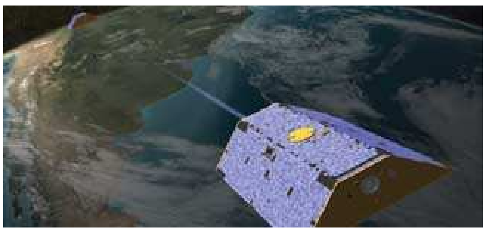
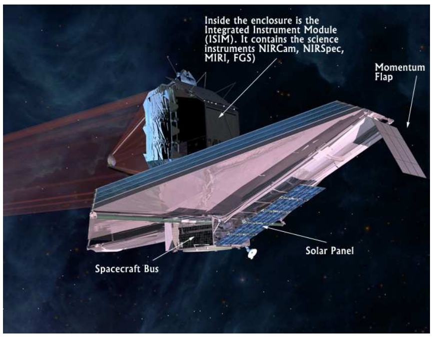
	STAR C1 communications satellite • Payload: 28 C-band & 14 Ku-band transponders + 1 X-band transponder • Orbit: GEO • Mass: 4100 kg • Dry mass: 1750 kg • Dimensions: 4.0 m x 3.2 m x 2.4 m, 22.40 m span • Electric Power: 10.5 kW • Attitude: 3-axis controlled • Life: 15 yr (min.)
	GIOVE A navigations satellite • Payload: L-band navigation transponders • Orbit: MEO • Dimensions 1.3m x 1.3. m x 1.4 m (stowed) • Wet mass : 660 kg • Electric Power generation: Sun tracking arrays • Actuators: wheels, magneto-torquers, thrusters • Sensors: Earth Horizon sensor, gyros, Sun sensors • Pointing: 0.55^{o} pitch/roll, 2.1^{o} yaw • Orbit determination: NORAD, laser ranging, GPS • Propulsion: $\Delta v = 90$ m/s • Communications: S-band • Cost: 33.9 million US$ (FY 2006)
	Meteosat weather satellite (1^{st} generation) • Payload: SEVIRI camera + data dissemination • Service availability: 95% • Orbit: GEO • Mass: 322 kg (282 kg dry) • Payload mass: 63 kg • 3.195 m x 2.1 m (D) • Spin stabilized: 100 rpm • Electric power: 240 W • Mission life: 4-5 yr • Cost : 90.3 million US$ (FY 2000)
	GRACE scientific satellite • Trapezoid body: 3.1 m x 0.8 m x 1.9-0.7 m • Mass: 460 kg • Orbit: LEO (500 km) • Pointing accuracy: 3-5 mrad • Electric power: 160 W • Control: Gaseous nitrogen control system with 12 attitude control thrusters and two orbit control thrusters; Nitrogen mass is 34 kg • Communications: S-band, 1 Mbps data rate • Thermal control: $\pm 0.1^{o}$ on critical components • Life: 5 yr

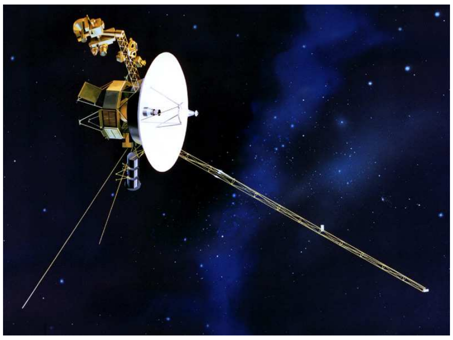
Table 1: Continued

<table>
<tr><td>
</td><td>James Web Space Telescope (JWST)

Payload: 6.5 m diameter telescope with 25 m^2 collecting area
Orbit with period of 1 yr (L2 point)
Mass: 6200 kg
Telescope operating temperature: 40 K
Electrical power: 2000 W
Data rate: 28 Mbps
Life: 5 yr (design)
Cost: 2400 US M$ (FY 2006) + 1000 US M$ for 10 years of operations
</td></tr>
<tr><td></td><td>ENVISAT

Payload: 10 optical and microwave Earth observation instruments
Orbit: Sun-synchronous (LEO)
Mass: 8211 kg
Dimensions:

In orbit: 26 m x 10 m x 5 m (in orbit)
Launch: 10.5 m x 4.57 m (D)

Electrical power generation: 6.6 kW @ End Of Life (EOL)
Useful electric power: 3.8 kW (3.2 kW in Eclipse)
Propulsion: Rocket system with about 300 kg of propellants
Communications: S-band
Life: 5 yr
Cost: ~1500 US M$ (FY 2001)
</td></tr>
<tr><td></td><td>Voyager Deep Space Probe

Mass: 722 kg
3-axis stabilized
Electric power: 421 W @ Begin of Life (BOL)
Communications: @ 8 GHz wavelength with antenna of 3.7 m diameter
</td></tr>
<tr><td></td><td>Mars Express (ESA)

Launch mass: 1223 kg (120 kg adapter + 173 kg payload)
1.5 m x 1.8 m x 1.4 m
Propulsion: 414 N main engine (430 kg propellant)
Attitude thrusters: 2 x 4 10 N thrusters
Pointing performance 0.15^o
Communication: 1.6 m high gain antenna
Electric power: 650 W at max. distance from the Sun
Operating temperature: 10-20 ^oC
Data storage: 1.5 GByte
</td></tr>
</table>

Table 1: Continued

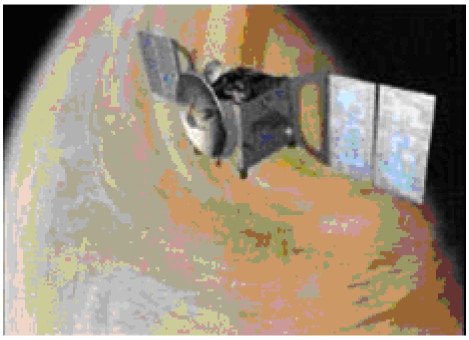
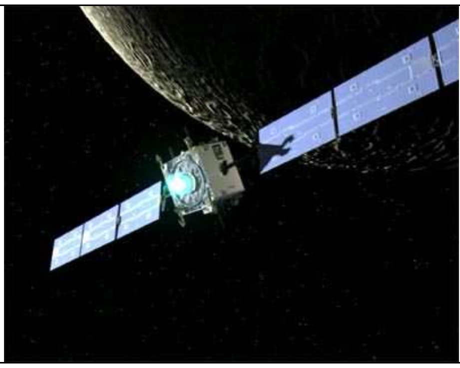
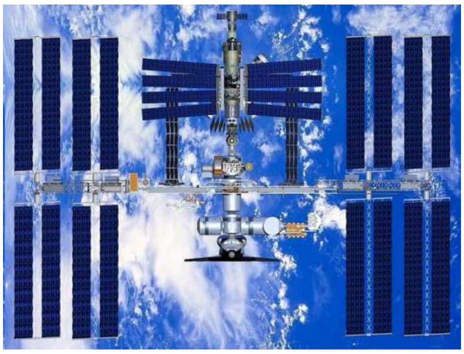
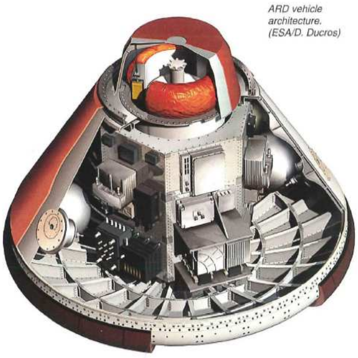
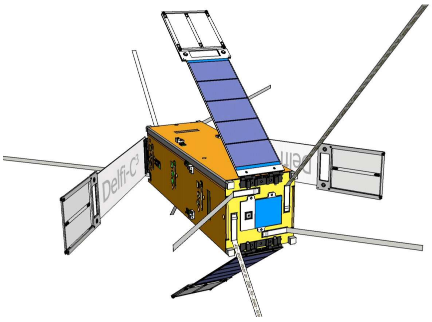
	Venus Express (ESA) • Launch mass: 1244 kg (104 kg payload) • 1.65 m x 1.7 m x 1.4 m • Aluminum structure • Power: 650 W at max. distance from the Sun + batteries for eclipse periods • Propulsion: 414 N main engine (530 kg propellant) • Attitude thrusters: 2 x 4 10 N thrusters • Communication: S-band (5 W) and X-band (65 W) • Data storage: 1.5 GByte • Cost: 262 million US$ (2005)
	SMART-1 (mission to the Moon): • Mass: 370 kg (19 kg payload) • 1m cubic body • Wingspan: 14.0 m • Orbit: GTO to polar orbit about Moon (altitude: 300 to 10000 km) • Electric power:1.9 kW • Life:2-2.5 yr
	International Space Station (ISS) • Mass: 420.6 ton (with 2 Soyuz vehicles docked) • Wingspan: 72.8 m • Length 108.5 m • Assembled in space • Orbit: LEO (altitude/inclination: 407 km/51.6 degree) • 30 large deployable items • Electric power:110-124 kW • Life:10 yr
	Apollo Command & Service Module (CM/SM) • Mass: 30,332 kg • Dimensions: 11.03 m x 3.9 m (diam.) • Endurance: 14 days • Mission Δv: 2.8 km/s • Main propulsion: 91.2 kN thrust • Power: Fuel cells • Attitude control: 16 thrusters • Communications: S-band

Table 1: Continued

	Ariane/Automated Transfer Vehicle (ATV) • Mass: 20750 kg • Dimensions: 10.3 m x 4.5 m (D) • Attitude control: 3-axis controlled (28 x 220 N thrusters for attitude control & braking) • Electric power: 3.8 kW • Main propulsion: 4 x 490 N
	Ariane Return Demonstrator (ARD) • Mass: 2.8 t • Dimensions; 2 m x 2.8 m (D) • Attitude control: 7 blow down hydrazine 400 N thrusters • Telemetry: 200 parameters transmitted to ground for flight analysis • Descent and landing system: Parachute
	Apollo Lunar Lander • Mass: 14696 kg • Dimensions: 6.37 m x 4.27 m (D); Landing gear diameter is 9.4 m • 3-axis controlled (16 thrusters of 441 N each) • Electric power: 48000-60000 Wh (i.e. 480-600 W for a 4 day mission or less for longer mission durations) • Descent stage Δv: 2470 m/s • Main propulsion: 1 x 44.40 kN
	Delfi-C^3 • Technology test satellite • Mass: 2.2 kg • 0.1m x 0.1 m x 0.34 m • Electric Power: 2-3 W • Attitude: Controlled by 1 magnet, 2 hysteresis rods • Communications: Amateur UHF/VHF band • 12 deployable items • Life: 3 months • Cost: 2.000.000 € (FY 2007)

Differences between the various spacecraft described are amongst others in:

- Orbit (i.e. environment): Some popular orbits are the geostationary orbit as well as a range of low Earth polar orbits that allow for viewing every location on Earth at least twice a day. For deep space S/C a whole range of orbits applies without any one particular.
- Performance; for instance some spacecraft are able to support a large and power-hungry payload, whereas others are only able to support a small (light) payload. Others are capable of transporting the payload over a large distance (deep space) as compared to Earth satellites that stay in the vicinity of Earth.
- Looks: The figures in *Table 1* show that spacecraft essentially consist of a central body with a number of appendages attached. The central body may be shaped like a rectangular or cubical box, a cylinder, a sphere, etc. The figures furthermore show that most of the spacecraft are equipped with photo-voltaic cells for electric power generation. These cells are either mounted on the spacecraft body (see e.g. Meteosat and GRACE) or are mounted on deployable panels (see e.g. STAR C1, GIOVE A, and Mars Express).
- Size: Some spacecraft are huge measuring several tens of meters, whereas others are as small as a 1-littre milk pack.
- Mass: Spacecraft mass varies from close to 1 kg up to several thousands of kg and with some exceptional spacecraft having a mass of several hundreds of tons.
- Power: On board power varies from a few Watt to several tens of kilowatt. Operation times vary from minutes to several years.
- Cost: S/C cost varies from about € 100,000 for a small and simple spacecraft to several hundreds of millions of Euro for larger and more complex spacecraft.
- Life/endurance: S/C operational life ranges from a few days up to about 15 years for the more recent telecommunications satellites.
- Reliability: As most S/C are difficult to maintain and the cost of S/C failure is high, S/C reliability over the operational life tends to be in the range of 0.5-0.9 (50%-90%).
- Operations; some spacecraft can operate autonomously whereas others rely on controllers residing on Earth (ground control).

For an explanation/definition of some specific terms you are referred to Table 2.

Table 2: Some definitions

• Spacecraft (S/C) Performance: A measure for how well the spacecraft does do what it needs to do (how well it functions). • S/C wet or loaded mass is mass of S/C with consumables (propellants, pressurant gases, etc.). • S/C dry mass is mass of S/C excluding consumables. • Launch mass is sum of S/C wet mass, launch adapter, separation system and kick stage (if present) • Life: Distinction should be made between time that the S/C is operational also referred to as operational life and the (on ground and/or on-orbit) storage life of the spacecraft. • Reliability is probability that an engineering system will perform its intended function satisfactorily (from the viewpoint of the customer) for its intended life under specified environmental and operating conditions. Reliability is basically a design parameter and must be incorporated into the system at the design stage. It is an inherent characteristic of the system, just as is capacity, power rating, or performance.

<u>Why do spacecraft differ or what makes them different?</u>

Spacecraft differ because they have different payloads, mission duration, target destination, are operated differently, use a different launcher to get into space, work on their own or in unison with others (e.g. in a constellation), etc. For instance:

- To supply water and food to the international space station requires a completely different spacecraft from one that supports a camera continuously taking images from Earth surface.
- An identical camera used to observe Earth, Mars or Venus requires quite different means to control its temperature. In general the closer we get to the Sun the higher the heat flow from the Sun. So going to Venus might make it difficult to cool the vehicle, whereas a mission to Mars might require heaters to keep the vehicle at a proper operating temperature. Also communications take much longer when the communication distance increases, etc.

Different payloads exist because of the different functions/tasks they have to fulfill, like communications, observations, science, navigation, space station supply, and sample return. An overview of current applications of typical mission payloads is given in Figure 1, see also [SSE, chapter 1.1].

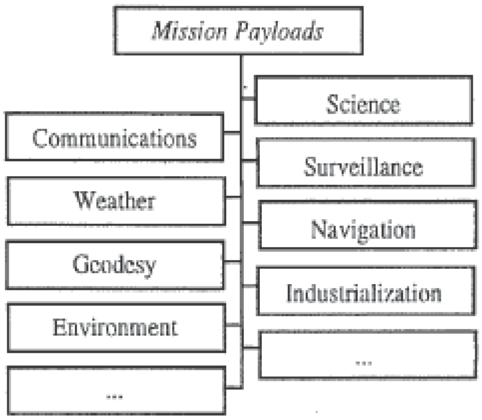


Figure 1: Mission payloads

The different payloads are sometimes used to classify S/C and more particular satellites. For instance we distinguish:

- Science, including solar physics, space plasma physics and high energy astrophysics;
- Earth Observation, including S/C dealing with for instance Earth's weather (cloud profile, rain, wave height, temperature and humidity), the chemistry of Earth's atmosphere (ozone, carbon-dioxide, etc.), imaging Earth's surface, altitude profiling;
- Communication, including for instance mobile satellite communications, (video/radio) broadcasting, multicasting, and internet communications;
- Navigation;
- Surveillance (for the military);
- Technology development;
- Etc.

Still, some S/C carry several different types of payload and are then difficult to classify.

Other reasons for why spacecraft differ are, because of:

- Different solutions to conduct the mission; most of the times there are different solutions to accomplish the same thing. Different design teams tend to come up with different solutions for the same problem. This may lead to differences in launch vehicle to launch the spacecraft into orbit leading to a different size and mass of the spacecraft, how autonomous the vehicle is, how the vehicle communicates to ground (directly or via a relay station on ground or a satellite relay station), etc.

- Payload improvement; over the years, payloads are improved, so we get equally capable instruments weighing less, using less power and so on, or more capable instruments weighing the same, using identical power, etc.
- Differences in available budget (money); we can use low- or high grade-equipment. Low grade equipment are less expensive than high grade equipment, but they have shorter life and/or fail more often and hence have a lower availability.
- Different political and ethical constraints.
- Etc.

The design process

Compared to a car most S/C are highly complex vehicles that bring with it high cost and long development times as well as a high design and development risk, i.e. the probability that design and development failures are incurred times the consequence of the failures.

To ensure a proper design[2] and hence to reduce the above mentioned design and development risk we need to follow a proper design process. The design process for a complex vehicle as a spacecraft normally requires several design cycles even for preliminary designs. We start by generating the requirements for the spacecraft to be designed. Next step is to come up with a feasible vehicle design, which is updated in the subsequent phases. This process is schematized in Figure 2. In this course we focus on vehicle-level estimation and system-level estimation based on prior experience only. For system-level design based on discipline oriented analysis you are referred to other courses.

The word “system” in the figure is contextual in nature. For example, a radio transmitter/receiver can be considered as a system, or as a subsystem of the spacecraft system, or as comprised of a number of other subsystems. To provide a framework, consider the following “system levels”:

- **Level 1**: Space mission segments: Space segment, ground segment, operations segment, etc.;
- **Level 2**: Space mission elements: Spacecraft, launcher, ground station, tracking station, payload, etc.;
- **Level 3**: Major spacecraft elements/subsystems: Communications, structures, propulsion, attitude, thermal, command & data handling, etc.;
- **Level 4**: Subassemblies: Thruster assembly, antenna assembly, etc.;
- **Level 5**: Components: Thruster, solar panel, reaction wheel, sensor, battery, antenna, camera, etc.;
- **Level 6**: Parts (fittings, fasteners, blades.....);

Level 1 and 2 design aspects have been dealt with in an earlier course and mostly focus on how the various segments/elements interact and what constrains their design. In this course we amongst others focus on spacecraft design to a level of detail sufficient for level 2 design as well as the design of the major spacecraft elements (level 3 design).

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[2] The dictionary gives a great number of entries for design of which we like to present the ones that are considered most applicable to the case at hand. From Dictionary.com:
Design (noun):
    A drawing or sketch
    The purposeful or inventive arrangement of parts or details: the aerodynamic design of an automobile; furniture of simple but elegant design.
Design (verb): To plan out in systematic, usually graphic form: design a building; design a computer program.

The spacecraft design process as depicted in Figure 2 starts by first defining the objectives and the system requirements after which the actual design takes place in several rounds or design phases.

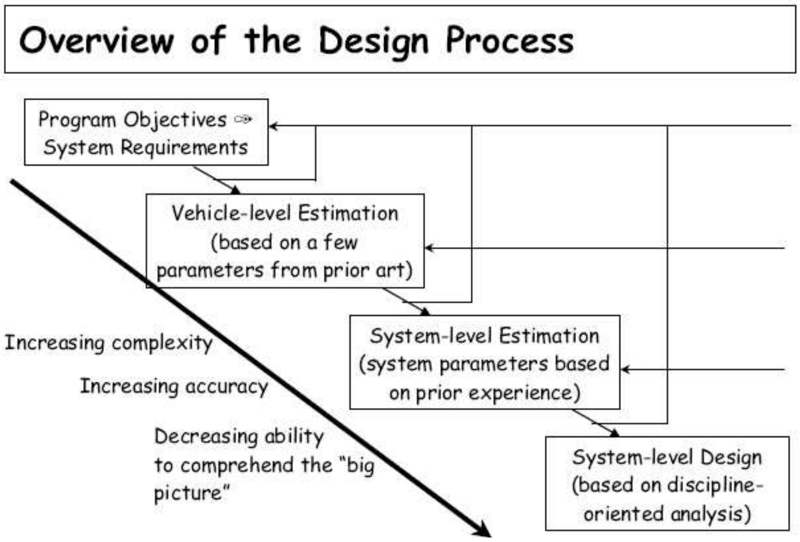


Figure 2: The Design Process [Maryland]

With each round the level of detail as well as the accuracy of the design results increases (we are becoming more confident about the design). With each round also the number of people involved in the design (usually from different companies/organizations) as well as the cost associated with the design will increase. Contrary there is an decreasing ability to comprehend the big picture. In design, we typically distinguish between three rounds/design phases, being:

- Conceptual[3] design, wherein we aim to select the 'best' concept. Estimation accuracy of the vehicle level parameters typically is in the range of 50%.
- Preliminary[4] design, wherein the concepts selected are worked out into more detail. Estimation inaccuracy of the vehicle level parameters decreases to 15-25%.
- Detailed design which ends with identification of manufacturer, manufacturing methods to be used, etc. Typical estimation inaccuracy decreases to about 10%.

Exercise: Try to define the various design phases in your own words and discuss how these phases tend to differ from each other.

In each phase we go through more or less identical steps, but with increasing level of detail. Each phase ends with a review (evaluation). This generally is referred to as a structured design approach. Basic steps in each phase (but at different level of detail and applied to different) are:

1. Define (design) problem; Proper knowledge of the problem that is to be dealt with is necessary to allow for providing a proper design solution, see for instance Figure 3. For this, one needs to have a proper understanding of the mission. In relation to spacecraft design, one should know what the spacecraft is supposed to do.

[3] Conceptual design in engineering generally deals with the generation of the basic ideas of how to solve a particular engineering problem, i.e. the selection of the most appropriate technology/ies. Conceptual design takes place very early in the design process, under pressure, and must usually be accomplished within a short time and, if done incorrectly, many late engineering design changes. It has been shown that most of the product life cycle costs are determined during this important stage and cannot be reduced in later stages. Hence the methods used should allow for the selection of the right concept within a short time, thereby reducing the amount of late design changes through the use of proper design margins

[4] The principal purposes for preliminary design of any device: (1) To obtain quantities of materials for making estimates of cost. (2) Obtain a clear picture of the structural action, (3) Establish the dimensions of the structure, and, (4) Use the preliminary design as a check on the final design.

2. Establish requirements; Requirements essentially are the criteria that provide direction to the design and in the end are used to judge whether the design is successful or not. They are closely related to the design problem. For instance, in the case of the above introduced Saturn sample return mission, it might also be the case that no requirements have been generated for how much (mass and size) samples should be returned.

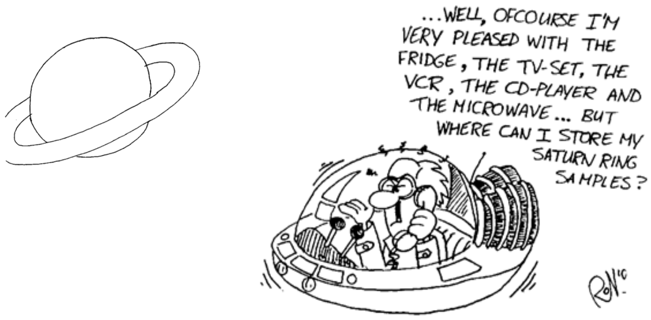


Figure 3: Illustration of bad problem definition / requirements generation

3. Set up options; This is the creative part of the design ("brainstorm") wherein different approaches to solving the problem are generated. Some approaches may be directly copied from earlier solutions, but may also be entirely new.
4. Analyze options; This is the calculation intensive part, where the options generated are analyzed to some detail. Level of detail differs with the required accuracy and hence the design phase. Sometimes a very simple analysis is conducted by just listing known advantages and disadvantages. Other times more detailed analysis is performed, using computer models, extensive testing, etc.
5. Compare options: In this step the design options (from step 3) are compared using the analysis results (from step 5).
6. Make choice; This is where the final selection takes place based on the comparison. Sometimes there is a clear choice, other times there might be two or more options that score about the same. In any case the purpose is to limit the number of options that flow to the next level of design for further study to limit the development time and cost
7. Evaluate outcome (how well did we solve the problem); If necessary iterate: Iteration is needed in case no satisfactory solution has been found, i.e. no solution has been found that fulfills all requirements or the accuracy of the analysis performed is not satisfactory.

The underlined letters together make up the acronym DESACME.

In the lectures S/C bus design we focus on the sizing of the spacecraft in conjunction with the other elements of the mission and the major spacecraft elements mostly based on prior art. Such methods usually are suited for conceptual and preliminary design purposes. In more detail, we will discuss the spacecraft vehicle design process (see earlier in this text) and how to:

- Generate a spacecraft requirements list (Chapter 2)
- Perform spacecraft vehicle sizing with an accuracy (see earlier in this text) fit for conceptual design purposes and develop a simple spacecraft configuration (Chapter 3)
- Perform budgeting and add design margins (Chapter 4)
- Evaluate the spacecraft sizing and budgeting results as obtained from the steps described in the chapter 3 and 4 (Chapter 5)
- Perform spacecraft (sub)system sizing (Chapter 6)

Course material

The course material essentially consists of the material offered in this syllabus complemented by the course slides.

2 Generating a spacecraft requirements list

In this section the purpose is to discuss the generation of a spacecraft requirements list.

Or to find out what we need to design for!!

According to the dictionary, a requirement is something that is imposed as an obligation; a necessity. So a requirements list is a list of things that provide a statement on what a system is obliged to do and how well and under what constraints. It may also provide a list of how we would like to interact with the system and again how well. A spacecraft requirements list is then nothing more than a numbered list of all requirements relating to the spacecraft as a whole (not its elements). For illustration, an example requirements list (not complete) is provided in Table 3.

Table 3: Example requirements list

Category	**ID**	**Requirement**	**Rationale**
Payload	1.1	Payload dimensions: 1150 x 1410 x 1950 mm^3	-
	1.2	Payload mass: 296 kg	-
	1.3	Payload power: 280 W orbital average, 792 W peak (792 W when imaging, <25 W non-imaging (how much time imaging vs non-imaging?)	-
	1.4	Pointing accuracy: ≤ 0.1 degree	-
	1.5	Pointing stability (drift): ≤ 0.01 degree/s	-
	1.6	Pointing knowledge: ≤ 0.001 degree	-
	1.7	Temperature range: -10 deg C - + 30 deg C	-
	1.8	Transmission capability: ≥ 320 Mbps	-
Orbit	2.1	Target: LEO (polar)	-
	2.2	Drag compensation Δv = 700 m/s	-
Launcher	3.1	Launch mass: ≤ 1200 kg	-
	3.2	Launcher payload envelope: 4000 mm x 1500 mm (D)	-
	3.3	Direct launch in polar orbit	-
Other	4.1	Reliability: 0.9 (to ensure reliability of S/C including payload better than 0.8)	-
	4.2	Cost: ≤ 80 M$ (FY 2000)	-
	4.3	Lifetime: ≥ 5 yr in orbit (+ 2 yrs in ground storage)	-
	4.4	Series size: 40	-

Note that in the table the space system element is identified where the requirement originated from next to a category of "other requirements" that are related to programmatic issues, operations, or still other.

In this section we will deal with the requirements relating to a spacecraft. We will show/discuss how the payload poses requirements that need to be fulfilled by the bus. In addition we will show that requirements can come from various other sources than just the payload and may depend heavily on program objectives[5]. In addition we discuss how constraints are placed on the project as we have limitations concerning available finances, time, etc.

Related material can be found in AE1110-II where a system view of the spacecraft is presented including a view of the spacecraft as part of a larger whole (i.e. the mission) and the spacecraft as consisting of several subsystems.

[5] **Program management** is the process of managing several related projects, often with the intention of improving an organization's performance.

2.1 Payload requirements

A spacecraft essentially is a platform (sometimes referred to as bus) carrying/supporting one or more payloads. Payloads can be instruments, supplies of consumables like water and oxygen (for instance for the international space station), broadcast (TV, radio, etc.) equipment or other spacecraft, see for instance Figure 4, where we have a crew module acting as the payload of some service module that provides for the necessary support. In turn the crew module has one or more crew as its payload and provides support to the crew (for instance keeps them alive).

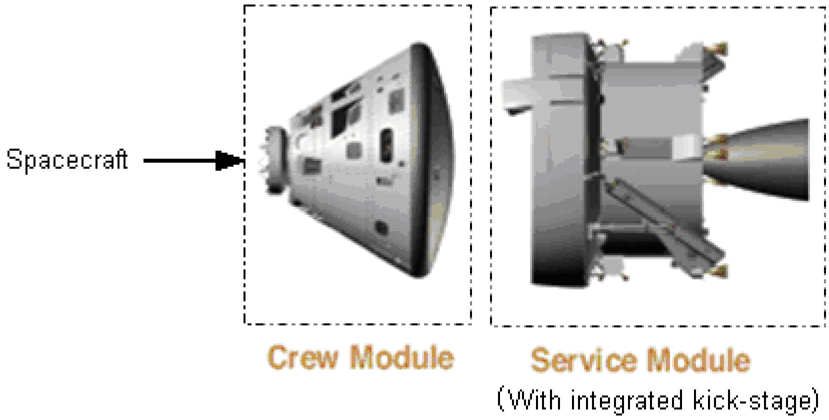


Figure 4: Spacecraft consisting of a service module carrying as payload a crew module

Each payload brings its own requirements to the spacecraft bus or service module. An important step in the requirements generation process is to list the characteristics of the payload(s) considered, draw them out and gather and list its/their needs for support. For instance, it must be clear that a large and heavy payload demands more support from the spacecraft than a small and lightweight payload or that a mission close by requires less propellant that a mission at the outer rim of our solar system. As an example, we refer to Table 5 and Table 6 that provide figures showing the lay-out of various instrument-type payloads as well as characteristic data. Information in the tables includes (when available):

1. Instrument mass
2. Instrument size/dimensions
3. Electrical power needed
4. Sensor/antenna orientation and pointing stability
5. Camera Field of View (FoV) or angle of view or antenna beam width
6. Data rate to be communicated to ground (or bandwidth)
7. Operating temperature range
8. Reliability
9. Life
10. Etc.

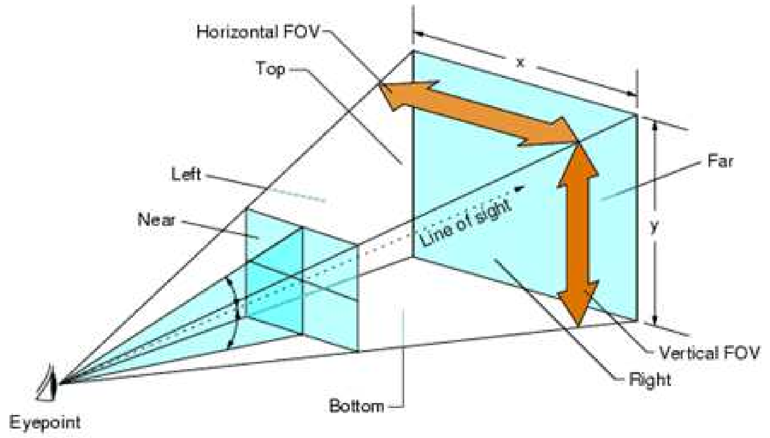
Below, some of the above parameters are commented upon.

Ad 1/2) All instruments are characterized by a certain mass and size. The spacecraft should allow for carrying the mass and provide for sufficient space to carry the instrument.

Ad 3) Most if not all instruments require some power source that allows them to operate. Generally the spacecraft provides for the necessary power required by the payloads.

Ad 4) Most instruments/antennas require stable pointing as a means to ensure measuring a stable signal from a given location (direction) or that a stable and clear signal is received on ground. For spacecraft, this may lead to requirements on both pointing direction and pointing stability. For instance for the Near Earth Object Surveillance Satellite (NEOSSat), a Canadian microsatellite using a 15-cm aperture http://en.wikipedia.org/wiki/Maksutov_telescope telescope to search for interior-to-Earth-orbit (IEO) asteroids, it is required that the vehicle is stabilized about the 3 body axes with pointing stability of ~2 arcsec (1 arcsec is 1/3600th of a degree) in a ~100 second exposure. So the Line of Sight (LoS) of the telescope is not allowed to move over more than 2 arcsec over a 100 second period.

Ad 5) Many optical instruments, particularly binoculars or spotting scopes, are advertised with their field of view specified in one of two ways: angular field of view, and linear field of view. Angular field of view is typically specified in degrees, while linear field of view is a ratio of lengths. For example, binoculars with a 5.8 degree (angular) field of view might be advertised as having a (linear) field of view of 102 mm per meter. Note that both descriptions apply to the same instrument. Also communication antennas require a certain field of view relating to the beam width of the antenna. Knowing about the field of view is important as solar panels or other extendable may not block (part of) the field of view.



$$\text{Aspect Ratio} = \frac{y}{x} = \frac{\tan(\text{vertical FOV}/2)}{\tan(\text{horizontal FOV}/2)}$$

Figure 5: Definition of Field of View, Line of Sight and Aspect ratio

Ad 6) Data rate: Measure of amount of data generated/transmitted. It can be viewed as the speed of travel of a given amount of data from one place to another. In general, the greater the bandwidth of a given path, the higher the data transfer rate. In telecommunications, data transfer is usually measured in bits per second. For example, a typical low-speed connection to the Internet may be 33.6 kilobits per second (kbps). In computers, data transfer is often measured in bytes (1 byte is 8 bits) per second.
Bandwidth: In electronic communication, bandwidth is the width of the range (or band) of frequencies that an electronic signal uses on a given transmission medium. It is measured in Hz or a multiple thereof. The larger the bandwidth the more information can be send. For instance, a typical voice signal has a bandwidth of approximately three kilohertz (3 kHz); an analog television (TV) broadcast video signal has a bandwidth of six megahertz (6 MHz) -- some 2,000 times as wide as the voice signal.

Ad 7) An operating temperature is the temperature at which an electrical or mechanical device operates. The device will operate effectively within a specified temperature range which varies based on the device function and application context, and ranges from the minimum operating temperature to the maximum operating temperature (or peak operating temperature). Outside of this range, the device may fail. Aerospace and military-grade devices generally operate over a broader temperature range than industrial devices; consumer-grade devices generally have the lowest operating temperature range.

Ad8/9) No further comments.

Table 4 provides the same parameters and a few more all helping to ensure the proper operation of the payload. Noteworthy of mentioning are command, control and telemetry which usually is provided for by the spacecraft and electromagnetic interference and contamination. The latter two may affect the design greatly, but are not dealt with in this course.

Table 4: Payload Accommodation Support Issues (from [NRC])

Mechanical	
	Size (outline and mounting dimensions)
	Mass
	Moments of inertia
	Uncompensated momentum
	Launch loads (shock and vibration)
	Disturbances
Thermal	
	Conducted and radiated heat flux to/from payload
	Thermal gradients and base plate distortion
Electrical	
	Power requirements
	Output data rate
	Command, control, and telemetry
	Electromagnetic interference
Optical	
	Sensor orientation and clear fields of view
	Pointing stability, agility
	Contamination: particulates, outgassing

Table 5: Overview of specific payloads and their characteristics

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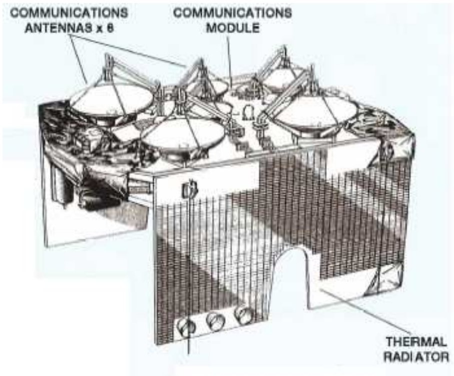
Communications Payload



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Communications payload
Typical parameters:

- From a few up to 30 or more transponders (combination of transmitter + receiver), plus a number of spares (roughly about 1/4th of the active transponders)
- Frequency (dependent on application)
  - Transmission between fixed points (FSS): 10.75-10.95 GHz
  - Broadcasting: 12.5 GHz
- Bandwidth: 4/6 MHz, 26 MHz, 36/72 MHz (in part depending on application)
- Transmitter power: several tens of W up to several 100 W
- Mass: from a few up to 10-20 kg/transponder
- Pointing accuracy: of the order of 0.1^{o}
- Reliability: of the order of 0.8-0.97 for a 10 year operational life

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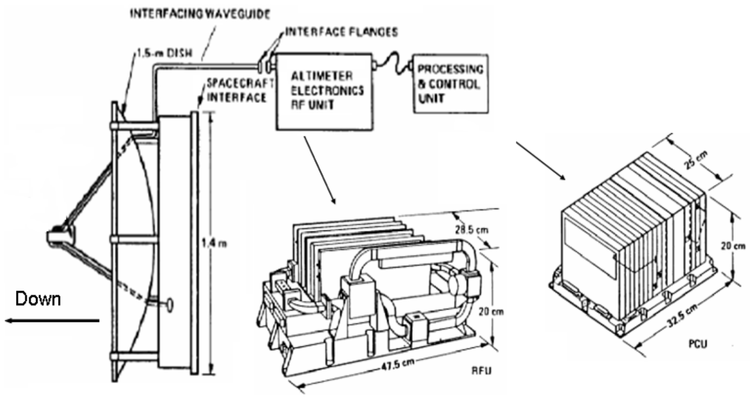
Altimeter



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Instrument: Altimeter
Instrument parameters

- Mass: 50 kg
- Power consumption: 75W
- Frequency of transmission: 13.65 GHz
- Data rate: 1200 bps (12 kbps during calibration)
- Size: See fig.
- Beam width: 1 degree
- Pointing requirements: Antenna must be nadir pointed
- Reliability: > 94% for a mission life of 4 yrs (Envisat RA-2)

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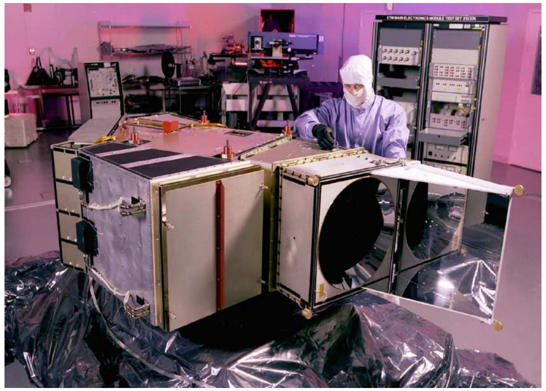
Remote Sensing instrument (camera)

	Instrument: Landsat-7 Enhanced Thematic Mapper (ETM)
	Instrument parameters
	• Mass: 425 kg
	• Power: 590 W (imaging), 175 W (standby)
	• Duty cycle: 15% imaging
	• Thermal control: 90 K (focal plane)
	• Pointing requirements:
	  o Control: 60 arcsec (1 sigma)
	  o Knowledge: 45 arcsec (1 sigma)
	• Jitter (jitter can be thought of as shaky motion) : 4 arcsec (1 sigma)
	• Physical Size:
	  o Scanner Assembly: 196 x 114 x 66 cm
	  o Auxiliary Electronics: 90 x 66 x 35 cm
	Data taken from:
	http://eospso.gsfc.nasa.gov/eos_homepage/Instruments/ETM/

Observation camera 	**Instrument: Ball High Resolution Camera** Instrument parameters • Design Live: > 5 years achieved with redundant architecture for orbits between 400 to 900 km from 0 degrees to sun-synchronous • On-board Storage Capacity: Optional equipment scalable up to 200 Gbits (equivalent to over 90 square images) • Communications Image Data: Optional 320 Mbps X-band transmitter and gimballed antenna • Payload Mass: Total weight is 296 kg, total weight with options is 342 kg • Power Consumption: 792 W when imaging (peak), < 25 W non-imaging (orbital average) • Telescope Size: 115 cm x 141 cm x 195 cm (rectangular) Note: Numerical ranges reflect orbit altitude options (400 km to 900 km)

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Navigation payload

GIOVE B payload module on sled ready for thermal testing
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Galileo Payload:

- Mass: 81 kg (navigation antenna 8-10 kg, 3 Rubidium atomic clocks of 1.4 kg each)
- Antenna size: 1.4 x 1.6 x 0.2 m^3
- Power consumption: 474 W
- Required attitude control accuracy: 0.5 deg
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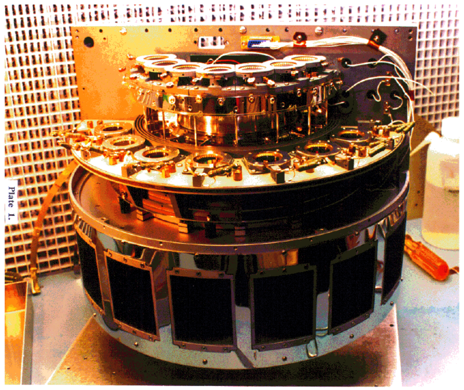
Science payload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Thermal Ion Dynamics Experiment (TIDE)/Plasma Source Instrument

- Investigation of Earth’s plasma environment
- Total Field of View (FOV): 96% of 4π sr
- Resources:
    - Mass: 17.1 kg
    - Power: 9.1 W
    - Data rate 4.0 kbps
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Table 6: Overview of payload characteristics (see also [SMAD])

Purpose	Instrument Name	Size L x W x D (m)	Mass (kg)	Avg. Power at 28 V (W)	Data Rate (Mbps	Aperture (m)	Pointing Accuracy (deg.)
Resources	Gravity Gradiometer	0.23 m sphere	10	1			1 -- 2
	Synthetic Aperture Radar	2.8 x 3.7 x 1.4	808	3000	120	8 x 2.8	205
	Multi-Spectral Mid-IR	1.5 x 1 dia.	800	900	30	1	0.1
	Thematic Mapper	2 x 0.7 x 1.1	258	385	85	0.406	0.08
	ENVISAT ASAR	10 x 1.3	832	1365	100		
Environ-mental	Limb Scanning Radiometer	4.8 x 1.9 dia.	800	125	0.52		
	Microwave Radiometer	4 x 4 x 4	325	470	0.2	4	ca. 0.1
	Dual Freq. Scatterometer	4.6 x 1.5 x 0.3	150	200	0.01	4.6 x 0.3	1
	Ocean SAR	20 x 2 x 0.2	250	300	120	20 x 2	0.1
	Solar Spectrum	0.4 x 0.3 x 0.6	16	60	Low		ca 3
	Doppler Imager	1.25 x 0.6 x 0.8	191	165	20		ca 3
	Photometric Imaging	1.4 x 1.4 x 0.5	147	330	0.01		ca 1.5
Solar Physics	Lyman-Alpha Coronograph	2.8 x 0.88 x 0.73	250	87	13.5		0.003
	X-ray Telescope Spectrometer	2.7 x 1 dia.	465	30	0.4		0.003
	Solar Optical Telescope	7.3 x 3.8 dia.	6600	2000	50+	1.25	
	Solar magnetic Velocity Field	2 x 0.4 x 0.4	183	322	2+		0.003
	100 m Pinhole Camera	1 x 1 x 2	1000	500	0.5		
	Extreme UV Telescope	2.78 x 0.86 x 0.254	128	164	1.28		
	Solar Gamma Ray Spectrometer	1 x 1 x 3	2000	500	0.1	0.134	0.003
Space Plasma Physics	Ion Mass Spectrometer	0.5 x 0.5 x 0.4	80	334	0.01		1
	Beam Plasma	0.6 x 0.7 x 0.7 + two 0.7 dia. ant.	17	38	0.016		5
	Plasma Diagnostics		2000	250	50		
	Doppler Imaging Interferometer	$(0.25)^3$	100	620	0.2		
	Proton (Ion) Accelerators	6.7 x 3.4 x 3.10	500	1500	0.256 (4.2 TV)		1
High Energy Astro-physics	Gamma Ray Burst	2 x 4 dia.	1000	120	0.01	3	
	Cosmic Ray Transition	3.7 x 2.7 dia.	1500	230	0.1	2.7	
	X-Ray Spectrometer/Polarimeter	1.6 x 1.6 x 3	2000	300	0.03		0.1
	Short X-Ray	1 x 1 x 3	1000	300	0.025	1 x 3	0.1
	High Energy Gamma Ray Telescope	4 x 3 dia.	10000	100	0.003	3	0.1

Some S/C carry multiple payloads/instruments with each their own requirements as is illustrated in Table 7. This of course leads to a more complex design as now we have to satisfy the demands of multiple instruments/payloads.

Table 7: Payload package on board of solar orbiter for research of the Sun's environment

Instrument	Mass [kg]	Power [W]	kb/s
Solar Wind Plasma Analyzer (SWA)	6	5	5
Radio & Plasma Waves Analyzer (RPW)	10	7.5	5
Coronal Radio Sounding (CRS)	0.2	3	0
Magnetometer (MAG)	1	1	0.2
Energetic Particle Detector (EPD)	4	3	1.8
Dust Detector (DUD)	1	1	0.05
Neutral Particle Detector (NPD)	1	2	0.3
Neutron Detector (NED)	2	1	0.15

More data on payloads can be obtained from:

- [SSE]
- Books like
  - Jane's Spaceflight directory
  - Observation of the Earth and its environment by H.J. Kramer
- Internet, like:
  - CEOS EO handbook Catalogue of EO instruments: http://www.eohandbook.com/eohb2008/earth_sat_instruments.html

2.2 Requirements from other space system elements

Next to requirements originating from the payload, other requirements originate from considerations concerning the interaction of the spacecraft with the other elements in the space system (see AE1110-II). Like launcher, ground station, communications architecture, mission operations center, communications, command and control (C^3) center and the trajectory/orbit to be flown Some typical requirements are listed in Table 8.

Table 8: Other sources of spacecraft requirements

Spacecraft shall: fit in launcher Mass Size withstand hostile environment communicate with ground and/or other spacecraft respond in a timely manner to commands control orbit transport payload to final destination (target orbit), if and when necessary

How the various sources mentioned above lead to spacecraft design requirements is discussed in more detail hereafter. Goal of the discussion is to provide guidelines for students on how to derive such requirements.

Spacecraft shall fit in launcher
The spacecraft designer should make sure the spacecraft will fit in the launcher and more particular in the designated payload area (payload bay). The size of the payload bay is given by the payload dynamic envelope, i.e. the envelope taking into account the reduction in available space because of the vehicle dynamics (vibrations). For illustration, Figure shows Payload Transport Vehicle (PTV) fitted into Ariane 5 payload bay. PTV is ATV (Automated Transfer Vehicle), but adapted for carrying payload and is considered an intermediate step to a Crew Transfer Vehicle (CTV). Figure shows a somewhat peculiar shape of the dynamic envelope (dotted line encompassing PTV), not uncommon to most launchers. Maximum available diameter is 4.57 m. It also shows that the payload is mounted onto a payload adapter (conical ring in figure) carrying the PTV. From the figure we also learn that the cylindrical section of the payload bay is roughly 8m high. Parabolic section of payload bay may also be occupied, but this may require adapting the shape of the spacecraft so that it fits in the bay. Information on dynamic envelope of a launcher generally is contained in the launcher manual, a many page document describing the launcher, launch operations, launch site, launch performances, etc. A good secondary source may be the Launch Vehicle Catalogue (available on the course blackboard pages).

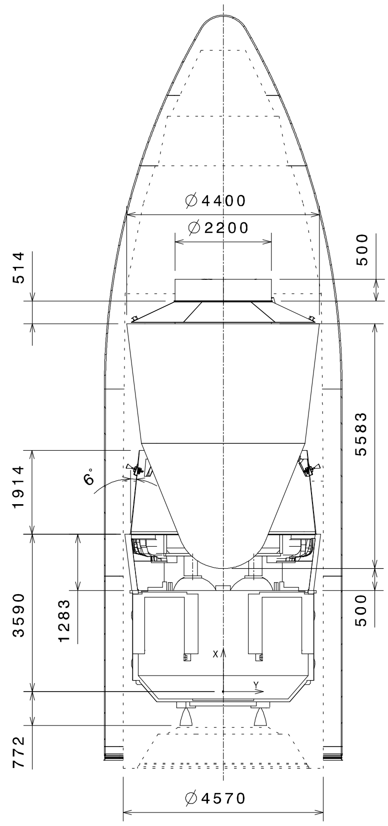

Figure 6: PTV fitted into Ariane 5 payload bay

Spacecraft mass limited by maximum mass that can be carried by launcher in to the designated orbit
A launcher can only launch a certain payload mass into some orbit. For the Taurus rocket, this is illustrated in Figure 7 for two different versions with a LEO target orbit. Important is to realize that performance depends on the orbit to be reached, but also on location of launch site and orbit inclination to be attained. Another important issue is that launch mass generally is not the same as spacecraft mass. Besides the mass of the spacecraft, it also includes the mass of the adapter and maybe even the mass of a kick and/or upper stage. Sometimes even more than 1 vehicle is launched with the launcher.

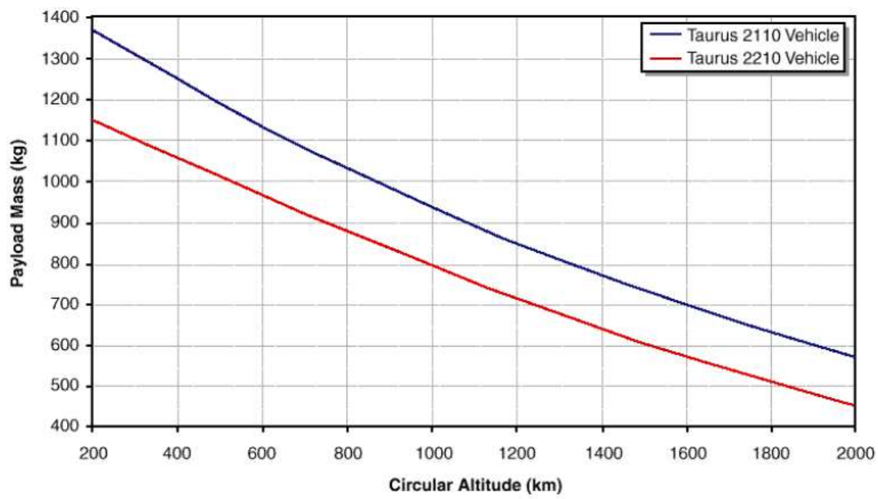

Figure 7: Taurus performance to 28.5^o LEO orbits [LVC]

The spacecraft designer should make sure the spacecraft will not exceed the maximum mass that can be carried by the designated launcher into the designated orbit. If the spacecraft turns out to be more heavy, another launcher (and probably more costly) may need to be selected, which may lead to a heavy cost burden on the project.

Spacecraft shall transport the payload to its final destination
As launchers have limited delta- v (velocity change) performance, the spacecraft may need to perform one or more maneuvers before reaching its final orbit. This is illustrated in Figure 8 for a mission to the Moon.

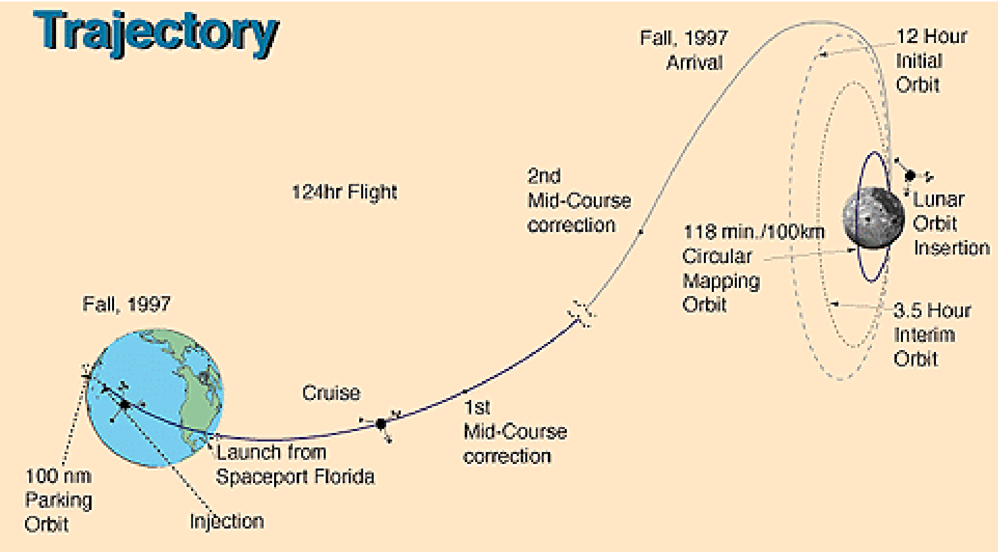

Figure 8: Manoeuvres needed for spacecraft to travel from parking orbit to final lunar orbit

From the figure, it follows that we need various manoeuvres to first inject the spacecraft into lunar transfer orbit, to perform mid-course corrections and at arrival at the Moon to attain the final orbit. Each manoeuvre requires a certain delta-v, which may be obtained from past missions or orbit analysis. A collection of delta-v data for a number of manoeuvres is contained in this reader, appendix A. The sum of all manoeuvre velocity changes (absolute value) is referred to as *mission characteristic velocity*. The spacecraft designer now should consider whether the mission characteristic velocity is to be delivered by the spacecraft itself or that a kick stage (like the Russian Fregat upper stage) is used to deal with at least part of the required manoeuvres.

Spacecraft shall be able to withstand hostile environment
The principal environment a spacecraft experiences is of course the space environment. This environment is characterized by (hot) plasma, highly energetic particles, cosmic rays, solar flares, debris, monatomic oxygen, etc. Some details of the spacecraft environment have been discussed in AE1110-II. Effects of the space environment may be spacecraft heating, charging, upset of electronics and so on, see figure . Next to the space environment, also the space launch environment should be considered, where we have to deal with large acceleration loads and heavy vibrations. An important source describing the launch environment is the Launch Vehicle Catalogue earlier referred to. Finally also other environments may lead to design requirements. For instance in case of road transportation, it might be the height of bridges that limit the size of a spacecraft or the loads during transportation might exceed those during launch. Also we should consider hoisting loads that may damage the spacecraft. So this one requirement on the spacecraft being able to withstand a hostile environment may lead to a range of requirements, like:

- Spacecraft shall be able to resist a certain acceleration load. An acceleration load of for example 6g means that the spacecraft has to resist an acceleration load of about 60 m/s^2. Compare with the load factor as defined for aeronautical applications. See http://en.wikipedia.org/wiki/Load_factor_(aeronautics) or http://en.wikipedia.org/wiki/G-force for a discussion on human tolerance of G-force.
- Spacecraft shall be able to work over some temperature range (range is to be defined)

- Spacecraft shall be able to cope with a certain dose of high/low energy particle radiation (electrons, protons, etc.)
- Spacecraft materials are to be selected than can withstand monatomic oxygen. For further reading, see e.g. http://www.reading.ac.uk/infrared/library/spaceenvironment/ir-spaceenvironment-atomicoxygen.aspx
- Spacecraft shall be able to resist handling loads (hoisting, transportation, etc.). For a good requirement, we need to define the loads in more detail, e.g. acceleration loads, humidity, temperatures, etc.

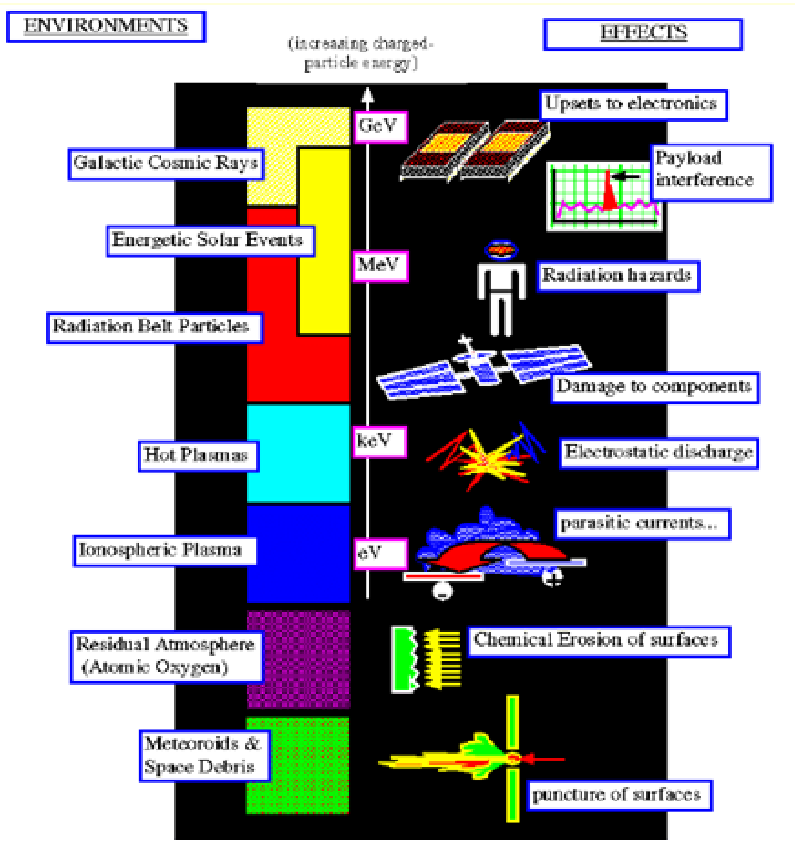

Figure 9: Space environment and effects on spacecraft (courtesy of ESA SME initiative training course)

Spacecraft shall be able to communicate with ground

Most times when designing a space mission, use is made of existing ground stations for communications with and tracking off the spacecraft. This is because the development of a new ground station is quite expensive. However, when selecting an existing ground station, this does require for the spacecraft designer to select certain communication frequencies. For illustration, Table 9 shows characteristics of an ESA Deep Space Network ground station.

From this data one learns that essentially two frequency bands are available for transmission and two for receiving. It should also be immediately clear that the spacecraft should be able to receive in the same band, otherwise they will not be able to communicate.

Table 9: Typical deep space ground station parameters

Characteristic	Typical range
Antenna dish diameter	15m, 35m
Transmit frequency	
S-band	2025-2120 MHz
X-band	7145-7235 MHz
Receive frequency	
S-band	2200-2300 MHz
X-band	8400-8500 MHz
Telemetry (downlink)	
Normal data rate	up to 1 Mbps
Maximum data rate	up to 105 Mbps
Telecommand (up-link)	
Normal data rate	2 Kbps
Tracking	
Range accuracy	1 m
Range rate accuracy	0.1 mm/s

- From the table also follows a normal data rate of 1 Mbps. Selecting this data rate, it more or less determines that the technology on board of the spacecraft should be able to handle this data rate.
- The very large antenna size indicates that this ground station has a very clear “voice” and can “hear” very well. This means that the antennas on board of the spacecraft can be relatively small. In other words, the larger the antenna on ground, the smaller the antenna in space and vice versa.

Still more spacecraft requirements
Many more requirements may be generated related to considerations of how the spacecraft interacts with the other mission elements. We mention:

- requirements with respect to reliability, availability, maintainability and Safety (RAMS).
- Ground station (location) and orbit together determine contact time available for communications
- Mission operations and orbit together determine the level of autonomy of the spacecraft
- C^3 system determines how the data is transported to ground
- Etc.

2.3 Financial budgetary envelope and political constraints

In practice, many requirements relate to the mission financial budgetary envelope and political constraints. Typical such **constraints** are:

- ESA may require one to buy European. Only when really needed, we buy foreign. For instance, for ESA science missions: Soyuz Fregat launcher is the current workhorse. As such, ESA may require you to design for launch on a Soyuz Fregat launcher.
- In the USA a set of government regulations referred to as **International Traffic in Arms Regulations** (**ITAR**) controls the export and import of defense-related articles and services on the United States Munitions List. Rocket motors and a number of other such items are included on this list.
- The United Nations have prepared regulations stipulating that to ensure sustainable access to space, spacecraft need to be designed such that they either burn up in the atmosphere and/or are injected into a graveyard orbit at End Of Life (EOL).
- Russians have launched many times a nuclear reactor in space, whereas the Western world is somewhat more reluctant to this.
- Nowadays, some space agencies specifically request for spacecraft to be de-orbited at end of life and or to place spacecraft in a “graveyard” orbit where they can do little harm.

2.4 Types of requirements

Requirements are categorized in many ways. Here only a few types of requirements are discussed.

Functional requirements
These are requirements that relate to the functions that shall be performed by the system, i.e. what the system is obliged to do. Functional requirement are usually phrased as “The system shall do <requirement>”. For instance, the spacecraft bus or service module should:

- Provide structural support
- Generate electrical power
- Ensure a proper thermal environment
- Handle data produced

- Transmit data to ground
- Provide for a pointing capability
- Provide for a stable platform
- Perform maneuvers to allow targeting different locations
- Provide landing capability (lander vehicles only)
- Provide life support (manned spacecraft only)

Non-functional requirements
These are requirements that specify criteria that can be used to judge the operation of a system, rather than specific behaviors of the system. They are usually phrased as: “System shall be <requirement>”

Other terms for non-functional requirements are "constraints“, "quality attributes", "quality goals", "quality of service (or operations) requirements" and "non-behavioural requirements".

Interface requirements
Requirements that stem from that the spacecraft interfaces with the other elements in the space system are sometimes also referred to as interface requirements.

2.5 Steps in requirements generation

Steps in requirements generation include:

- Establish a list of functions to be performed and constraints to be considered
- Determine a characteristic parameter that can be used to judge how well a certain function is performed or express a constraint
    - The parameter should be <u>measurable</u>, like thrust, pointing accuracy, mass, cost, life, etc.
    - If no such parameter can be found, then consider detailing the function or constraint (splitting it up into sub-functions, etc.)
- Develop criteria for how well the function is to be performed
    - For instance: Payload mass shall be equal or in excess of 100 kg
    - Criteria could be developed from already existing designs
- Document requirements in a requirements list + rationale

To keep a clear overview of the requirements and not to forget any, they are usually collected in a so-called requirements list; see the earlier introduced Table 3. To keep track of the requirements each requirement is given a unique identifier. A column is provided to also add the rationale behind the requirement. This could be a referral to some analysis document or just a short statement. In practice many different ways exist to keep track of requirements, but the principles are generally the same.

Requirements flow down
Some platform requirements flow down from the program objectives and constraints. Typically a space program shall be conducted within a certain time frame and at a certain cost. As all elements constituting the space mission bear costs, it should be determined early on the budget available for the spacecraft and in more detail the platform. Hence this will lead to requirements flowing down the spacecraft to the spacecraft subsystems, their components and so on. Hence, once the spacecraft requirements are know we can start its design. From the design requirements will be derived for the subsystems and so on (requirements flow down).

2.6 Requirements on requirements generation

Defining requirements can be a lengthy process, but ill-defined requirements can be very detrimental for a space project.

For instance, a requirement like that the spacecraft shall never fail is not considered a good requirement as this will become a very expensive requirement. For a requirement to be a good requirement, it needs to be defined in a “SMART” way. The meaning of the acronym SMART in this sense is:

- Specific – Requirements should specify what they are to achieve.
- Measurable – The requirements should provide a metric whereby all stakeholders can determine if the objectives are being met.
- Achievable – Are the requirements’ objectives achievable and attainable
- Realistic – Are the requirements realistic with respect to available resources?
- Time-bound – When is the team to achieve the requirements’ objectives?

Hence if one on the above criteria is not met, we do not have a properly defined requirement.

2.7 Problems

A number of problems for exercising upon are available via Blackboard (Maple TA), whereas a few are also contained in a separate workbook available from the TU-Delft online print shop. Of this workbook also an electronic copy is available on blackboard.

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3 Spacecraft design: Vehicle level estimation

In the early stages of spacecraft design we tend to investigate different options and it is important to quickly determine the feasibility of our design. For instance, we need to know if our spacecraft fits in the launcher, is not too costly, is reliable and so on. Such studies are typically referred to as feasibility studies.

In this section, we discuss a simple 'design' method that allows us to come up with a first design of our spacecraft with in a relatively short time and allowing to judge feasibility of the design. This method relies on the use of relatively simple relations derived at using statistical analysis methods and essentially consists of 6/8 steps:

1. Determine the type of vehicle to be designed
2. Determine the vehicle properties to be established
3. Determine whether estimation relationships (ER) are available for the estimation of the vehicle properties established in the previous step. If yes, continue with step 6, else continue with step 4.
4. Collect (historical) data from <u>comparable</u> spacecraft.
5. Perform data analysis and develop (new) estimation relationships
6. Estimate vehicle properties using known or newly developed estimation relationships
7. Generate a straw man configuration (i.e. a first configuration that can later be used for referencing to) and determine mass properties
8. Perform budgeting and include margins

In the remainder of this chapter these steps are discussed in some more detail, be it not necessarily in the order indicated above.

3.1 Type of spacecraft

First step in preliminary and/or conceptual design is to determine the type(s) of vehicle to be designed. In the foregoing, we have already shown that artificial satellites can be classified by function, for instance, orbiter, lander, ascender, kick stage and so on, and by target orbit Earth satellite, deep space, planetary probe. Also for some missions a variety of vehicles may be necessary. For instance, for some planetary missions, we may need both a kick stage and an orbiter. For some other planetary missions, we may even need a lander and/or an ascender (for instance for a sample return mission). Also for the important category of Earth satellites, as this category contains a large variety of different spacecraft, we usually make a further distinction into:

- Navigation satellites
- Mobile communications satellites
- Fixed communications satellites
- Earth observation satellites
- Science or technology demonstration satellites
- Other (space station, space observatory)

Other distinctions made are by mass, cost, size, and so forth, each having its own reasons. For instance, classification by mass is useful because it has a direct bearing on the launcher vs. cost trade-off. Also small satellites (and especially micro/nano/pico/femto satellites) are built quite differently from larger satellites, because they are built simpler (without propulsion, appendages, etc.), compacter and sometimes using components of a lesser quality. Table 10 provides a distinction of small satellites partly based on data taken from [Surrey].

Table 10: Satellite classification

Class	Mass [kg]	Cost [M$]	Time [yrs]
Large and medium heavy	>1000	200	5-15
Small	500-1000	40-80	2-3
Mini	100-250	20	2
Micro	10-100	10	1.5
Nano	1-10	1	1
Pico	0.1-1.0	>0.1	<1
Femto	<0.1	NA	NA

The satellites below 1000 kg are sometimes referred to under the common denominator “light satellites”. In addition, large satellites are considered to include:

- Heavy satellites: Mass > 3500 kg
- Medium heavy satellites: 1000-3500 kg

Depending on the vehicle to be designed and/or the specific mission, you may decide on other, more fitting/narrow, distinctions between spacecraft.

At the early stages of design, maybe you do not know whether a kick stage is needed or whether the spacecraft is large or small, but that does not really matter. What matters is that one develops sufficiently accurate methods to quickly determine the main features of the spacecraft under design. Such methods then will allow for quickly going through many different design options. By comparing the designs, the ‘best option’ can then be selected for further study. Below the method advocated in the present work is described in more detail.

3.2 Vehicle properties to be calculated/determined/established

Next step in the design is to decide what are the vehicle properties of interest. Typical properties that need to be determined for judging the feasibility of a spacecraft include a.o.:

- Vehicle total mass and size (to see whether there is a suitable launcher available)
- Vehicle power (to see whether the required power levels are doable)
- Vehicle cost (to see whether we stay within the allocated budget)
- Vehicle reliability (to see if the mission success probability can be guaranteed)
- Development risk (data is generally not widely available)
- Vehicle configuration
- Etc.

Depending on the needs of course also other parameters can be included.

3.3 Method for spacecraft preliminary sizing

In the next few sections, we will present a method for estimating a number of important characteristics of spacecraft. Characteristics include such parameters like spacecraft launch mass, spacecraft power, spacecraft size, cost, reliability and risk. The method as presented uses a mixture of analytical and statistical estimation relationships. Next to presenting the method we also aim to explain why the various parameters are of importance and to define the various parameters in more detail.

Launch mass estimation

Launch mass (in combination with the target orbit) is critical for launcher selection. In addition, launcher selection greatly determines the launch cost. Typically we have a cost per kilogram to LEO in the range 10 k$ - 100 k$ per kilogram [NRC]. For GEO, this cost is a factor 2-3 higher.

When studying mass figures from existing spacecraft we learn that many different mass items (each with their own definition) are distinguished. For instance when considering a spacecraft in launch configuration, it may consist next to the spacecraft itself of a kick stage (KM) or service module (SM), see for instance Figure 4, and a launch vehicle (or payload) adapter (LVA), see Figure 10.

Figure 10: Payload adapter (courtesy RUAG)

Of these, a kick stage is a propulsive stage connected to the spacecraft. It essentially consists of a large rocket engine, fuel tanks and a navigation and communication system. Its purpose is to give the actual passenger spacecraft an extra kick needed to reach its final destination. Once at its destination the kick stage separates and the actual spacecraft starts operations. The LVA provides for a physical interface between the launcher and the spacecraft and ensures that the spacecraft is properly connected to the launcher during launch. An overview of different masses related to spacecraft is given in Table 11 together with their definitions.

Table 11: Some vehicle mass definitions

Term	Definition
(Vehicle) launch mass (VLM)	Gross vehicle mass plus mass of kick stage (if applicable) and mass of launch vehicle adapter (LVA)
Vehicle injected mass (VIM)	Vehicle mass just after separation from the launcher. Gross vehicle mass minus mass of LVA
Vehicle gross/loaded/wet mass (VWM)	Total vehicle mass (sum of dry mass of vehicle and propellant mass) plus mass of kick stage (if applicable)
Vehicle on-station mass also referred to as vehicle mass in orbit or vehicle mass Begin Of Life (BOL)	Total vehicle mass when arriving on-station, i.e. in target orbit when starting operational life.
Propellant mass	Mass of propellants
Vehicle dry mass (VDM) or net mass	Gross vehicle mass minus the mass of propellants, pressurant and other liquids (e.g. coolant).
Vehicle empty mass (VDM)	Vehicle dry mass plus residuals; VDM and vehicle empty mass usually are fairly close
Vehicle mass End Of Life (EOL)	Mass of vehicle at end of operational life. mission
Payload mass (PLM)	Mass of useful load
Mass of spacecraft bus or platform	Vehicle empty mass minus payload mass
Vehicle structural mass ratio	Ratio of spacecraft bus mass to gross vehicle mass
Vehicle propellant mass ratio	Ratio of propellant mass to gross vehicle mass

The launch mass of a spacecraft of course includes the spacecraft itself, but may also include the earlier referred to adapter device and/or kick stage. It follows:

$$M_L = M_{SC} + M_{KM} + M_{LVA} \quad [1]$$

As in many cases a kick stage is not present, we also have:

$$M_L = M_{SC} + M_{LVA} \quad [2]$$

Notice that in case a kick stage is present, propellant needed for maneuvering is loaded on board of the kick stage. When no kick stage is present this propellant should be loaded on board of the spacecraft itself and hence a different mass results.

So to estimate launch mass, we need to estimate spacecraft and kick stage (or service module) wet/gross/loaded mass and LVA mass. This will be focus for the next two sections, where the first section deals with spacecraft mass estimation and the second with the mass estimation of kick stages and LVA.

A) Spacecraft wet/gross/loaded mass estimation

Various methods exist for estimating spacecraft wet/gross/loaded mass. The most simplest method is to consider that spacecraft wet mass, also referred to as gross mass, Begin of Mission (BOM) or loaded mass. can be estimated based on payload mass only. The reasoning being that with an increased mass of the payload also the vehicle mass itself will increase. Another method is by taking the sum of dry vehicle mass, i.e. the vehicle mass excluding propellant mass (and other expendables) and propellant mass. The rationale behind the latter method is that surely empty vehicle mass and propellant mass will change when payload mass changes, but by estimating propellant mass separately, we can also take into account the effect of spacecraft life and more importantly a change in the destination (target location) of a spacecraft. Compare for instance the European developed Venus Express and Mars Express vehicles mentioned in Table 1 which have quite distinct destinations, but for which the dry vehicle mass only changes slightly. Vehicle mass is more different though, because of a change in target planet.

Hereafter, we will describe the two methods mentioned in the foregoing in some detail.

<u>Method A: Estimating wet mass based on payload mass only</u>

In this method the wet mass of the spacecraft $(M_{SC})_{wet}$ is estimated based on known payload mass only. Typical relations providing spacecraft wet (on station) mass for various types of spacecraft are given in appendix C. In Figure 11, some of these relations have been plotted for comparison. The term on station mass is used here to denote that the vehicle mass is considered for the vehicle when on station, i.e. in the target orbit.

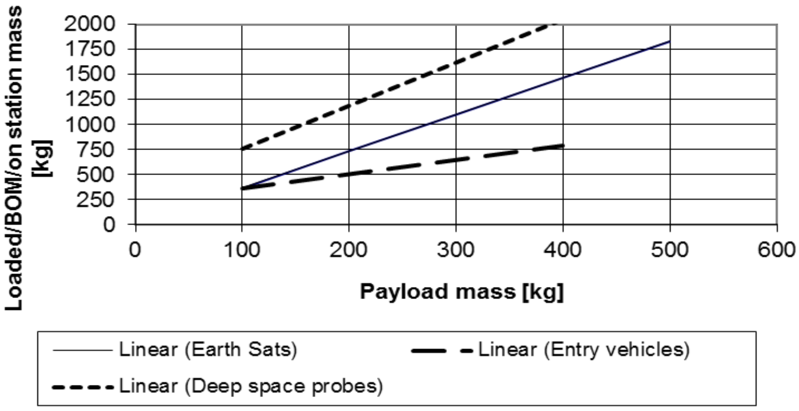

Figure 11: Wet mass versus payload mass for various types of spacecraft

The figure shows that for each of the vehicle types the loaded mass increases with payload mass (as could be expected). The results also confirm the importance of considering different relationships for different spacecraft with deep space probes showing highest gross mass (for a given payload mass) over the full range plotted. Note that unlike for deep space probes and (unmanned) entry vehicles the range of payload masses for Earth Sats is much larger (up to about 2000 kg), but for clarity only a small part of the range is plotted here.

Method B: Estimating wet mass based on estimating dry mass and propellant mass separately

In this method, spacecraft wet mass is determined by summing spacecraft dry mass and propellant mass according to:

$$(M_{SC})_{wet} = (M_{SC})_{dry} + M_{propellant} \quad [3]$$

So this method comes down to determining vehicle dry mass, also referred to as empty mass, net mass or mass at burnout, and propellant mass. although technically they might not be all exactly the same. For now, this difference is neglected. This is discussed in the next few sections.

Vehicle dry mass estimation

In case payload mass increases, it is logical to expect that also vehicle dry mass will increase. Hence vehicle dry mass generally is estimated by assuming a linear relation between vehicle dry mass and payload mass:

$$(M_{SC})_{dry} = a \cdot M_{payload} + b \quad [4]$$

The values of a and b are constants that depend on the type of vehicle and the mass range considered. Various such relationships are collected in Appendix C. A comparison of three relationships to estimate dry mass of Earth satellites is provided in Figure 12.

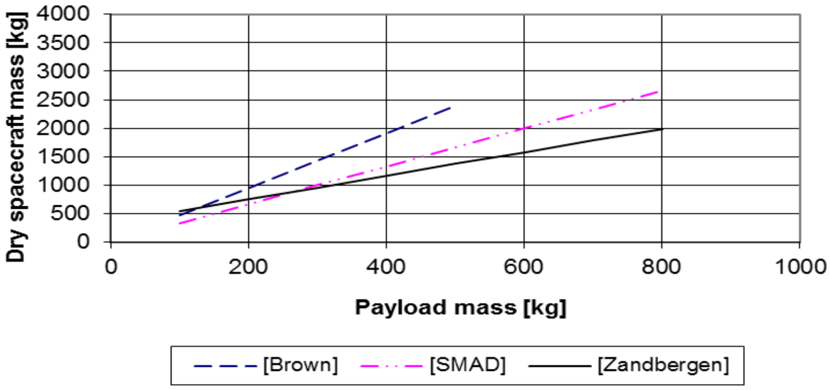

Figure 12: Comparison of 3 relationships for estimating dry mass of Earth Satellites

The figure shows that even though all 3 relationships apply to Earth satellites, the results for one and the same value of payload mass can be quite different. Also note that the relationship provided by Brown is only valid up to about 400 kg payload mass, whereas the relationship provided by Zandbergen is valid up to about 2000 kg payload mass. For the SMAD relationship no such range is given. In Table 12 the estimated results are compared for the ERS-1 spacecraft to the actual value as reported on [eoPortal Directory].

Table 12: Comparison between predicted and actual dry mass for ERS-1 spacecraft.

	Brown	SMAD	Zandbergen	Actual value
Payload mass [kg]	888.2 kg	888.2 kg	888.2 kg	888.2 kg
Dry mass [kg]	4263 kg	2958 kg	2171 kg	2066.4 kg

It clearly shows that the Brown relationship for estimating dry mass is significantly off, thereby demonstrating the danger of using relationships outside the range for which they have been developed.

Propellant mass estimation

Propellant can account for anywhere from a very small portion to as high as 35–45% of a spacecraft's wet mass, depending more fundamentally on the design altitude, design lifetime, and stabilization scheme of spacecraft. The required propellant mass can be estimated using the rocket equation[6] as given in Equation [5].

$$\Delta v = w \cdot \ln\left(\frac{M_o}{M}\right) \quad \rightarrow \quad (\Delta v)_e = w \cdot \ln(\Lambda) \qquad [5]$$

With:

$$\Lambda = \frac{M_o}{M_e} \qquad [6]$$

Here M_o is the vehicle mass at start of the maneuver and M_e the vehicle mass at end of maneuver. Λ is denoted as the vehicle mass ratio and w is the effective exhaust velocity of the rocket. For details on the derivation of the rocket equation, you are referred to the Section on "Launch vehicle trajectories" in AE1110-II. Using:

$$M_o = M_e + M_{propellant} \qquad [7]$$

We obtain (depending on whether initial mass is known or final mass):

$$M_{propellant} = M_o \ \left(1 - e^{-(\Delta v/w)}\right) \qquad [8]$$

or:

$$M_{propellant} = M_e \ \left(e^{\Delta v/w} - 1\right) \qquad [9]$$

The first step toward estimating propellant mass is establishing a delta v (Δv) budget. This budget includes allowances for orbit injection, drag compensation, attitude control, and deorbit at end of life. Typical values for Δv can be found in the annex A. Next step is to select the type of rocket system to be used as this greatly determines the effective exhaust velocity of the system, see propulsion subsystem design for more details. Typical values of rocket exhaust velocity for spacecraft propulsion are:

- Primary propulsion: 2200 m/s - 3200 m/s
- Advanced primary propulsion: 10000 – 20000 m/s
- Secondary or Reaction Control System (RCS) propulsion: 600 m/s - 2200 m/s

Example: Consider a 1000 kg heavy satellite that has to deliver a Δv of 2000 m/s. In case we equip this vehicle with a propulsion system with an effective exhaust velocity of 3000 m/s, it follows using the rocket equation a mass ratio of 1.95. This means that at end of this maneuver, the satellite mass is reduced to 513 kg. Propellant mass expelled is thus 1000- 513 = 487 kg.

Some spacecraft have to conduct various maneuvers each having its own Δv requirement. To calculate the total propellant load required, the delta-v for the maneuvers may be summed provided that the rocket exhaust velocity for all maneuvers remains the same. In case different systems (with different exhaust velocity) are used to conduct the various maneuvers, one need to carefully consider the order of the maneuvers to be calculated.

Sometimes one makes a distinction between large maneuvers, like orbit insertion, which are being carried out by some main or primary propulsion system and small maneuvers, like drag compensation, attitude control and station keeping, which are carried out by a secondary propulsion system or the

[6] The rocket equation, sometimes referred to as Tsiolkovsky's 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).

reaction control system. This is because the delta v for large maneuvers is reasonably well known, whereas for small maneuvers they are less well known. So, to estimate the propellant mass for the large maneuvers the method described earlier can be used. RCS propellant mass can be estimated in an identical way, but as the RCS propellant mass is relatively benign (up to about 10-11% of spacecraft wet mass as compared to the earlier mentioned 35-45% of total propellant mass when including main maneuvers), an alternative method is by using an RCS propellant mass estimation relationship like the one given in appendix C. The relationship is plotted in Figure 13 and is valid over a spacecraft wet mass range of 500-2400 kg.

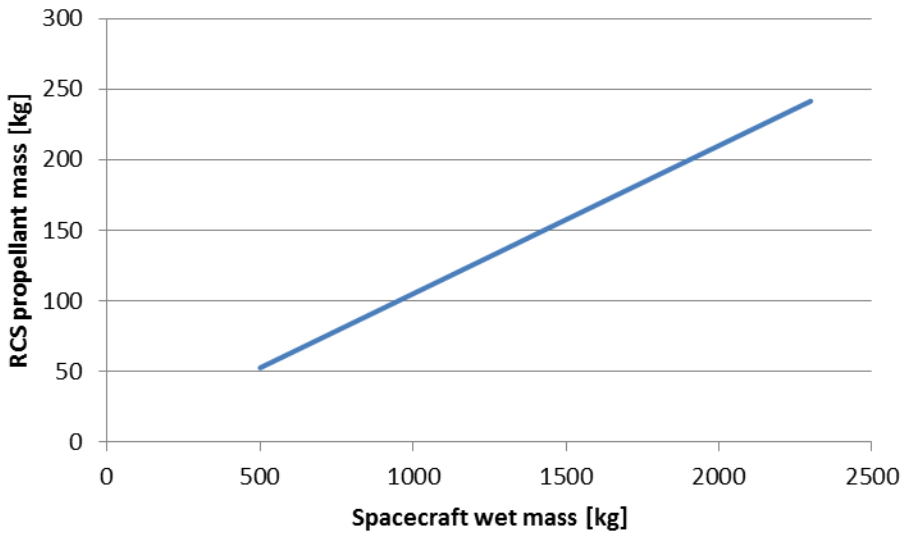

Figure 13: RCS propellant mass versus spacecraft wet mass

B) Kick stage mass and adapter mass estimation

In agreement with the calculation of spacecraft mass, kick stage mass is considered as the sum of kick propellant (from propellant budget) and kick stage dry mass:

$$M_{KM} = (M_{KM})_{Dry} + (M_p)_{KM} \qquad [10]$$

$$(M_{KM})_{Dry} = 17.5\% \text{ of } (M_p)_{KM} \quad ; \quad \% \text{ range is 10-25\%} \qquad [11]$$

Relation [11] indicates that kick stage dry mass is in the range 10-25% of propellant mass carried on board of the kick stage. In case no further data is known, it is advised to use the mid-range percentage value (here 17.5%) to estimate kick stage dry mass (for further info, see launch vehicle design). Propellant mass is estimated using the same method as for the spacecraft. The only critical thing is to determine how much of the velocity change is to be given by the kick stage and how much by the spacecraft itself. Some further info can be obtained when discussing launcher design.

Adapter mass:

- From [SMAD]:

$$M_{LVA} = 1\text{-}2\% \text{ of injected mass} \qquad [12]$$

- From [Brown]:

$$M_{LVA} = 0.0755 \cdot M_L + 50 \quad (M_L \text{ is in the range 200-3500 kg}) \qquad [13]$$

Power estimation
Spacecraft power estimation is important as the required power determines/drives to a large extent the mass and size of the solar array. The mass and size of the array are important for the calculation of the Mass Moments Of Inertia of the spacecraft, see later, whereas the size of the solar array also determines whether we should opt for body mounted fixed array or for a deployable solar array design, see also later.

Typical power estimation relationships are provided in appendix C. Some (low payload power) relationships are plotted in Figure 14 over their range of validity. For high payload power they are plotted in Figure 15.

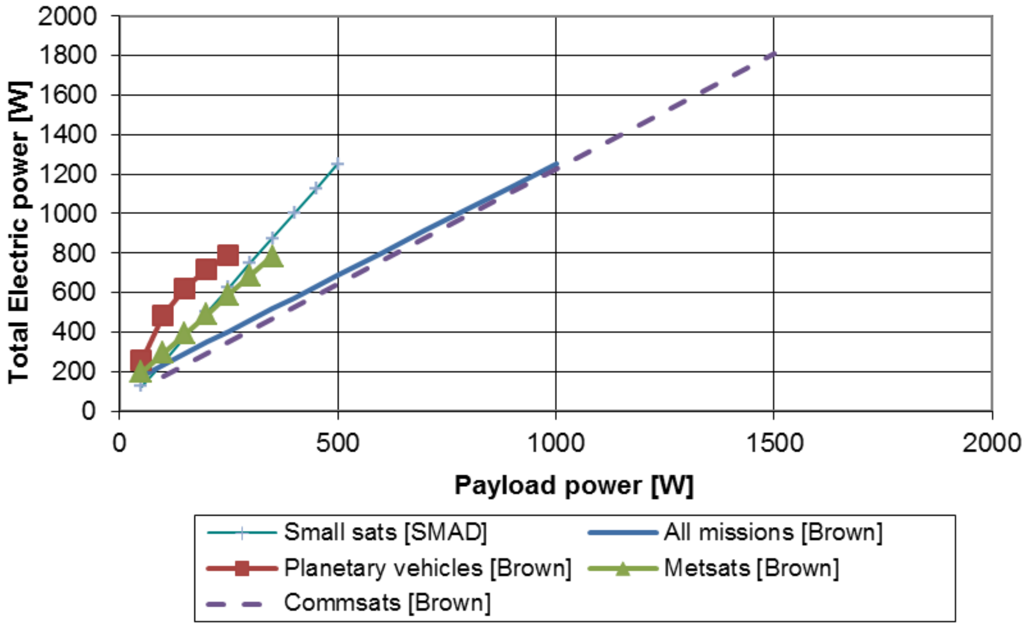

Figure 14: Total electric power versus payload power for various S/C types (low power regime)

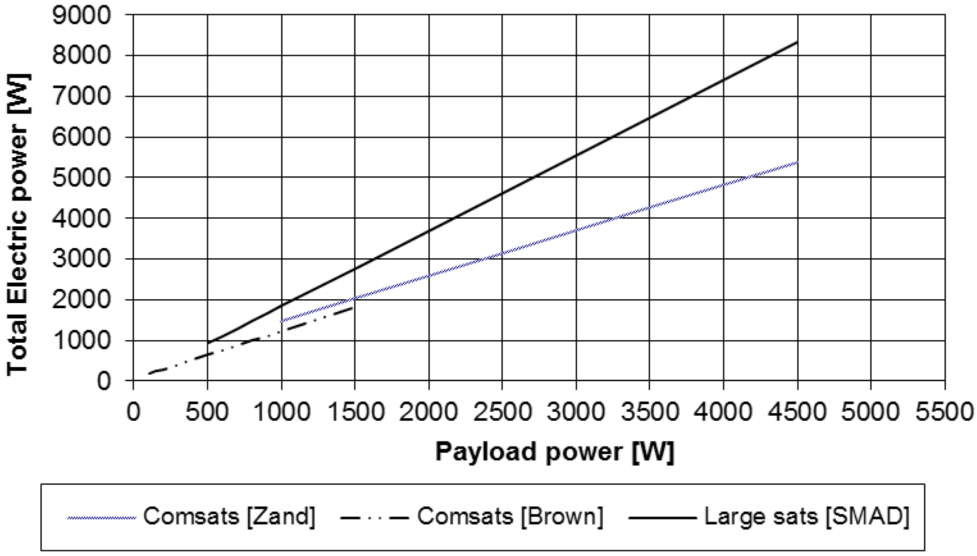

Figure 15: Total electric power versus payload power for various high power S/C types

Results again show the differences that exist between the various spacecraft types. So make sure you select the proper relationship.

Size estimation

Spacecraft size (or volume) is important to determine at an early stage in the design, as it must allow for accommodating the payload and the spacecraft must fit in the launcher. Spacecraft volume and solar panel surface area (needed for wing design) can be estimated using the earlier determined spacecraft mass and power estimate.

The volume of the spacecraft body is estimated using:

$$V = \frac{\left(M_{S/C}\right)_{wet}}{\rho} \qquad [14]$$

With ρ is spacecraft mass density. The latter is determined using known envelope size and mass of existing spacecraft, assuming that the vehicle mass is homogeneously distributed over the spacecraft envelope. For instance:

- Large spacecraft ([SMAD], 75 S/C, 136 kg < total mass < 3625 kg): 20-179 kg/m^3, average is 79 kg/m^3
- SmallSats (18 S/C, dry mass < 300 kg): 200-1000 kg/m^3, Average is 338 kg/m^3

Once the volume is determined, spacecraft body size follows once the basic shape of the body is determined. Typical shapes of spacecraft include sphere, cylinder, rectangular, octagonal, etc. For instance, assuming a cubical body, we obtain a body linear dimension as given in Equation [18].

$$L_b = V^{1/3} \text{ or } \mathrm{L_b} = \left(\frac{M}{\rho}\right)^{1/3} \qquad [15]$$

For instance, for a mass density of 64 kg/m^3 follows:

$$L_b = 0.25 \cdot M^{1/3} \qquad [16]$$

This relation is fairly easy to remember.

Now that the linear dimension of the body is determined, we can easily determine the body area of importance for drag, and solar pressure force calculation. It follows:

$$A_b = L^2 \qquad [17]$$

Solar array area A_a and solar array mass M_a can be estimated using Equations [18] and [19] [SMAD]

$$A_a = \frac{P}{P_\delta} \qquad [18]$$

$$M_a = \frac{P}{P_{sp}} = 0.04 \cdot P \qquad [19]$$

With:

- A_a = array area = array length x array height (array height usually depends on spacecraft height, length may be distributed over two or more wings)
- P_δ is power density, which in SMAD is given a value P_δ = 100 W/m^2.
- P_{sp} is specific power, which in SMAD is given a value of 25 W_e/kg for standard solar panel.

Notice that values of power density and specific power also can be determined based on data collected from comparable vehicles. Values given above apply to spacecraft in Earth orbit using standard Silicon panels @ 1 AU. Values for other panel types can be obtained from the section on electrical power subsystem design in Chapter 6. Values for other distances to the Sun can be determined using Equations [20] and [21], With d is distance to Sun expressed in AU.

$$P_\delta = \frac{100}{d^2} \ [\mathrm{W/m^2}] \qquad [20]$$

$$P_{sp} = \frac{25}{d^2} \ [\mathrm{W_e/kg}] \qquad [21]$$

Notice that when a spacecraft moves away from Earth, both power density and specific power decrease. This is of course because the available solar power decreases with increasing distance. Besides distance also the type of solar cell used has an effect on specific power and power density. For a discussion on the effect of selecting other cell types, see chapter 6.

Cost estimation

Spacecraft cost include development cost (one-time or investment cost) and production cost. In general, we can write:

$$C_{total} = C_{development} + N \cdot C_{production} \qquad [22]$$

And:

$$C_{SC} = \frac{C_{development} + N \cdot C_{production}}{N} \qquad [23]$$

Here C refers to cost and N is number of spacecraft produced. C_{total} gives the total cost of N spacecraft, whereas C_{SC} gives the cost per spacecraft. One typically find that with increasing numbers produced of some spacecraft the cost per spacecraft decreases. Available cost information allows for integral cost estimation, meaning that the cost estimate encompasses both development and production cost. For now, it is assumed that all costs determined hold for the development and production of a single spacecraft. How to take into account costing of large series of spacecraft is for further study.

Cost tend to increase with the size of the spacecraft. Cost data shows that spacecraft cost are in the range from 0.1 M€ for a simple Cubesat to well over 500 M€ for a large complex spacecraft, see Table 1 and for spacecraft with a dry mass in range 40-2350 kg can be estimated using:

$$C_{SC} = 0.3531 \cdot \left(M_{S/C}\right)_{dry}^{0.839} \qquad [24]$$

The above equation gives spacecraft cost (in M$, Fiscal Year (FY) 2000 money) as a function of spacecraft dry mass (in kg). The reason for using dry mass is that propellant cost, even though propellant mass can be quite large, is usually very small as compared to total vehicle cost. Hence, in that sense, vehicle dry mass is much more representative of vehicle cost.

To convert FY2000 money to FY2013 money, we have to take into account inflation. In general, inflation is a measure for the rate at which the general level of prices for goods and services is rising, and, subsequently, purchasing power is falling. As inflation rises, every dollar will buy a smaller percentage of a good. For example, if the inflation rate is 2%, then a $1 pack of gum will cost $1.02 in a year and so on. More in general, we find that the inflation correction factor can be expressed as:

$$Infl = \left(1 + \text{inflation rate / yr}\right)^{\text{\# of years}} \qquad [25]$$

For instance, when taking a period of 12 years and an inflation rate of 0.02/year (2%/year), we obtain a factor of about 1.268. A complicating factor is that inflation rates may vary from year to year. In that case the above simple relation is not to be used.

Figure 16, taken from a 1996 NASA study, shows specific spacecraft cost (i.e. spacecraft cost per unit of spacecraft dry mass) of a number of science spacecraft in relation to spacecraft dry mass. The figure shows two trend lines both with appreciable data spread. The first one is that for NASA's main line of spacecraft, which shows specific cost in the range 150.000 – 450.000 $/kg and that the larger values are applicable for smaller spacecraft. This is explained by that in terms of engineering small spacecraft can be as demanding as larger spacecraft, but as for larger spacecraft all equipment is heavier, this leads to lower cost per kg.
The second (linear) relation applies to missions with specific cost below roughly 100.000 $/kg. Here smaller vehicles seem to cost less than the larger ones. This is explained by that for this category the spacecraft are designed much simpler, for instance by selecting no propulsion and so on, thereby reducing complexity).

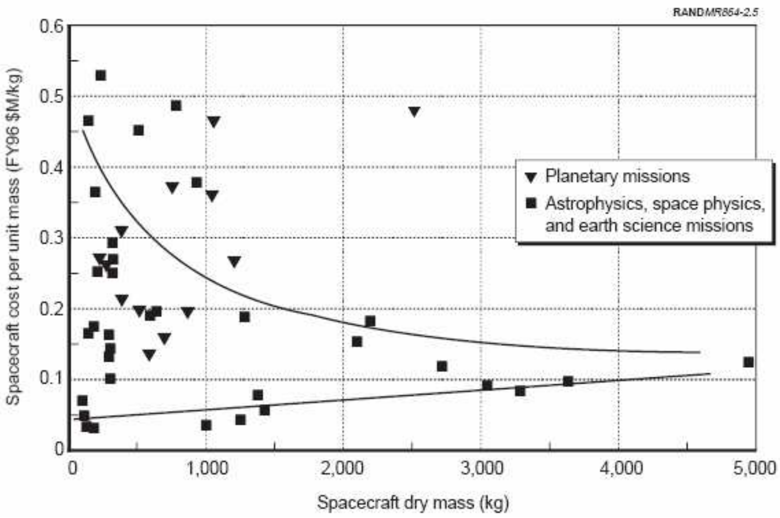

Figure 16: Effect of size on specific cost of science and planetary spacecraft [Sarsfield]

An explanation for the increasing specific cost with spacecraft mass is that the larger the spacecraft become, they tend to become more complex.

Example: Cost estimation
Consider the cost of a new EO satellite with a mass of 1000 kg. Using the cost estimation relationship given in Appendix C for EOsats, we obtain a value of 219.4 M$ in FY 2000 money. For comparison, equation [24], *gives a cost estimate of 116 M$. This then demonstrates the inadequacy of the latter relation to estimate the cost of this EO satellite.*

In case we are dealing with a science spacecraft Figure 16 could be used. For a spacecraft with a mass of 1000 kg we estimate a specific cost of 230.000-240,000 $/kg which leads to a total spacecraft cost of 230-240 M$ in year 1996 money. Correcting for inflation, see [SMAD, Table 20-1], we find this is equal to 245 – 255 M$ in year 2000 money.

Spacecraft life

Spacecraft life is important as we need to be able to determine how long the vehicle can be active and how long it can be stored (inactive) on ground or in space. The active life of a spacecraft is also referred to as operational spacecraft life. Data shows that the operational life of a spacecraft can be from up to 7-8 years for LEO spacecraft to 10-15 years for GEO spacecraft. Over this period, we need to consider that the harmful space environment (radiation, small particle impact damaging the solar panels, etc.) cause ageing, for instance of the solar panels, and as of ageing will lead to an increase in failure rate. Now the goal is to obtain a reasonable duration of the operational life with some probability of occurrence. This typically translates into a reliability figure, see a later section, for the given life duration.

Some spacecraft are kept as spares on ground and/or in space. As also some degradation takes place in storage, it is important to limit the storage time. Currently, existing designs show that storage times can be anything from just a few months up to the order of years.

Development time

The time needed to design and develop a spacecraft is generally referred to as the development time. Knowledge of realistic development times is needed so that one can estimate when the satellite will be available for service and for ordering a launch and setting a launch date. Some spacecraft, take roughly 10 years to develop, like very complex interplanetary spacecraft and science spacecraft, whereas others take only a few years to build/develop as the spacecraft is based on an already existing spacecraft with just a few modifications or is a very simple spacecraft.

The whole of the spacecraft life from definition and feasibility studies to operational usage and end of life is referred to as the spacecraft life cycle. For further distinction and to control the development of a space vehicle, ESA considers the following phases in the life cycle:

- Phase O/A: Definition and Feasibility studies, wherein a valuable and affordable mission is defined, a feasible solution is generated and technical support studies are performed in parallel to the generation of a feasible system
- Industrial competition to design and develop the spacecraft
- Phase B: Detailed design and team build up wherein the payload is optimized, the spacecraft design is tentatively frozen, building blocks are specified, spacecraft performances are refined, and the industrial team is build up through a competitive process.
- Phase C/D: The spacecraft is developed, assembled, tested and qualified for flight, i.e. the spacecraft is produced.
- Phase E: Operational usage.

Figure 17 shows the various phases distributed in time. Each phase is ended with a review to determine whether it is worthwhile to start the next phase or not. Generally the phases in a life cycle follow one after another, but sometimes some of the phases run partially in parallel. This is for instance to reduce development time.

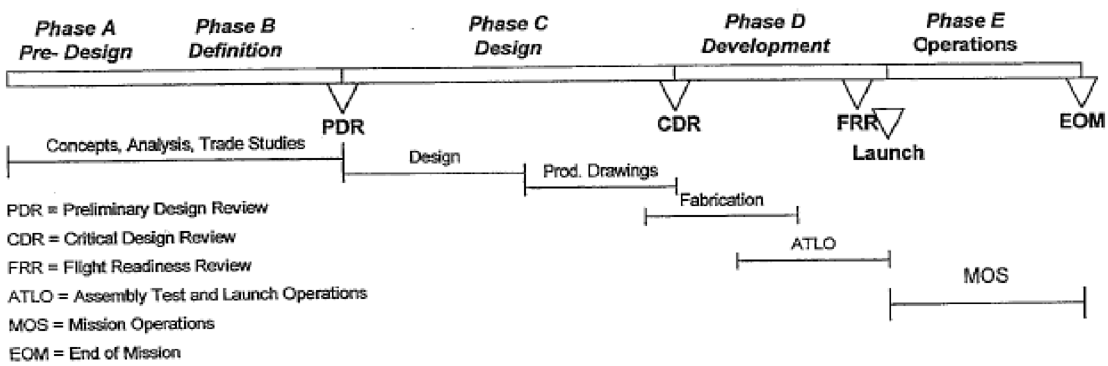

Figure 17: Typical phases in the life cycle of a spacecraft and their distribution in time

To estimate S/C development time it shall be clear that data on the phases in the spacecraft life cycle shall be collected for more or less comparable spacecraft.

Data on development time shows that development time ranges from just a few years up to 6 years for highly complex spacecraft (for space launchers it may even be up to 10-12 years.). Looking in more detail into how this total development time is distributed over the phases A to D, we find that phase A may take 1 month up to a few months, whereas phase B typically takes about 15-30% of the time it takes for the phase C/D. The duration of phase C/D greatly depends on the complexity and the uniqueness of the vehicle. It must be clear that a small and simple satellite relying on the use of off-the-shelf technology can be built faster than a large complex satellite requiring highly advanced technologies. Typical phase A, B and C/D durations are given in Table 13.

Table 13: Typical values on development phase duration

Phase	Duration
A	1-3 months
B	4-24 months (typically 15-30% of phase C/D time)
C/D	
Commercial GEO communications S/C	30 months
Science S/C	36-72 months

Reliability

Most items do fail at some point in time. Also spacecraft do fail. For instance, GSFC (Goddard Space Flight Centre) reported in 2003 a total of 439 anomalies for a total of 62 large spacecraft orbited successfully. Of these anomalies, 2% had a major or catastrophic effect, 13% a minor effect and the remainder a negligible effect. Another research of 310 small satellites inserted successfully, showed that 293 (95%) operated successfully until end of life.

As defined earlier, reliability (R) of an engineering system is the probability that this system performs its intended function satisfactorily (from the viewpoint of the customer) for its intended life under specified environmental and operating conditions. Likewise, for a large number of systems operating under prescribed conditions, the reliability is given by the ratio of systems still operating after a specified time period or number of uses (cycles).

Likewise, we can state that the probability of failure (F) of a spacecraft during the mission life is given by:

$$R = 1 - F \qquad [26]$$

> *Example: Reliability and failure probability estimate*
> *Say that we launch 1000 spacecraft. After 10 years only 200 spacecraft are still operating. In that case we find that the reliability of these spacecraft to survive a mission life of 10 years is 0.2 (or 20%). Additionally, it follows a failure probability over the 10 year mission life of 0.8 or 80%.*

Reliability of some item depends on its failure rate (λ), i.e. the percentage (or fraction) of items failing per unit time/cycle/launch or in FITS (total number of failures of an item in 10^9 hours; FIT = Failure In Time) [SSE]. Mathematically, this translates into:

$$\frac{dy}{dt} = -\lambda \times y \Rightarrow \frac{y}{y_o} = R = e^{(-\lambda t)}$$

Here y is the total number of items operating at any one time and y_o is the initial number of operating items. For a given failure rate it follows for the reliability[7]:

$$R = e^{-\lambda t} \qquad [27]$$

From Eq.[27], we find that reliability depends on the period we are considering and of course, if we consider a shorter period, fewer items will have failed over this period than when considering a longer period.

[7] In case the failure rate is given in number of failures per use cycle, the time t in relation [27] is replaced by the total number of use cycles during its life

Failure rate data of some item can be obtained by operating large numbers of this item and then keeping track of how many items fail per unit of time or per number of used (cycles).

Example: Reliability estimate based on failure rate
Using the earlier reported GSFC data, of 439 anomalies for 62 operating satellites in one year and focusing on the 2% serious failures only, we find 0.14 serious failures per spacecraft per year. Using this failure rate, we find a reliability $R = e^{-(0.14 \times 10)} = 0.25$ *for a 10 year mission.*

Hence the probability of a single spacecraft surviving a 10 year mission is 0.25 or if we have 1000 spacecraft operating at any one time, 10 years later only 250 will still be operating.

Typical values for spacecraft reliability are in the range 0.5-0.9 with typical lifetimes from 5-7 years see also the annex B. Typical values for spacecraft failure rate are in the range 0.056-0.139 serious failures per spacecraft per year and depend on the complexity of the spacecraft (complex spacecraft fail more often than simple spacecraft) and the quality of the materials/components used.

A single system may consist of multiple items or devices that can fail. The failure rate or FIT rate of such a system can be predicted by the sum of the failure/FIT rate of each of the devices in the system:

$$\lambda_{total} = \sum \lambda_i \qquad [28]$$

Here the various devices in the system are denoted by the subscript "i". For instance for a spacecraft consisting of a payload and the spacecraft bus itself, the failure rate of the spacecraft can be written as the sum of the failure rate of the payload and of the bus. Using Eq.[27], it can be shown that spacecraft reliability follows from payload reliability and bus reliability and vice versa according to:

$$R_{SC} = R_{Payload} \cdot R_{Bus} \qquad [29]$$

Eq.[29] actually means that for the spacecraft to operate, both the payload and bus should function correctly. If one of the two fails, it means the spacecraft fails.

Another important point is that if we have 1000 of the same (non-repairable) items, than the number of items that fail increases with time. Hence, reliability is time dependent.

Reliability is basically a design parameter and hence must already be incorporated into the system at the design stage. It is an inherent characteristic of the system, just as is capacity, power rating, or performance. The simple theory based on constant failure rate presented in this section is considered suitable for use during the preliminary design stages of a project. However, the success of the method depends on the failure rate data available or generated.

Some words of caution to the above:

- Eq.[27] is only valid in case:
    - Failure rate is constant in time. Unfortunately this is not always the case. Typically we find that especially at the start and end of life failure rates can be higher, due to e.g. infant mortality and burn in and (excessive) wear. Still [27] is a very useful relation for preliminary analysis in the early design stages.
    - Failure rate of the various components of a system are independent from each other. In real life, some failure of some component may also lead to failure of another component, this however is not considered here.
- Above relations are limited to non-repairable systems. For repairable systems the analysis becomes more complex
- To improve the reliability of some item, we need to improve its failure rate. This can e.g. be done by carefully controlled production/manufacturing and assembly and integration. Another way is by incorporating redundancy, i.e. the use of back up devices that take over if the original device operating fails.

- When determining failure rates, it is important to unambiguously determine the cause of failure. Causes of failure can be various, including design failure, operator failure, failures due to the environment and other failures. The importance of determining the cause of failure is because in case of an operator failure and or failures due to excessive environments, it is actually not the device that fails, but the conditions under which it is expected to operate are violated. Nobody can expect a mobile phone to still operate after mishandling it. So when determining the failure rates of some device, first the design and operator failures as well as the failures due to the environment need to be removed. The remaining failures can then be used to come up with some failure rate for the device under investigation.

Development Risk

Development risk is related to design and development failures. Risk is probability of failure (F) times consequence or severity of failure (S):

$$Risk = F \cdot Severity \qquad [30]$$

For estimating development risk, the probability of development failure has to be estimated as well as the level of severity. For preliminary analysis, usually three levels of severity/probability are distinguished:

- Low
- Medium
- High

Development risk tends to be high for new satellite projects requiring highly advanced technologies and low in case we use off the shelf technology. In case of high risk, it is important to have adequate project reserves (in terms of for instance funds and scheduled time). At low risk project reserves usually are 10% or less. At high risk, project reserves are > 25%. So for a low risk project, we generally add 10% of money or development time to make sure that we do not end up without having sufficient money and/or scheduled time.

To determine high risk items use can be made of a so-called risk map. This is a graphic method that allows depicting the risk of various elements that make up the system. For illustration, Figure 18 provides a risk map of 4 vehicle concepts that could be pursued to perform the mission. Clearly concepts A and E provide highest risk. This could be because for these concepts we need to develop both a new payload and a new S/C bus. The other concepts are less risky, which might be because of re-use of some well-proven bus and/or payload. Hence typical questions to be answered when determining the risk level is whether the S/C to be developed is a completely new S/C or that it reuses an existing payload and/or S/C bus.

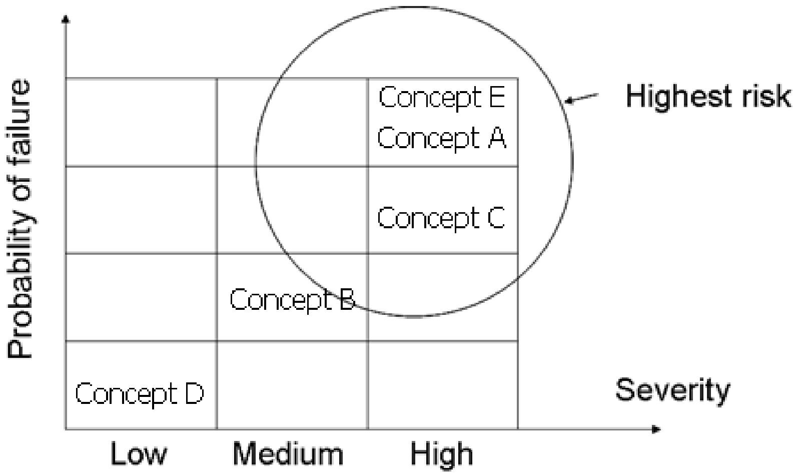

Figure 18: Risk map

Insurance rate/cost
Because of the high cost of spacecraft, spacecraft and or launcher failures tend to have high cost consequences. The risk of failure of a spacecraft or launcher can be determined using equation [30] taking F as the probability of a launch/spacecraft failure and the severity S as the cost of a launch/spacecraft failure. To cover these consequences insurance can be bought at the expense of some additional cost, but with the assurance that it the spacecraft or launcher fails, the insured value is reimbursed. Typically the insurance rate is slightly higher than the risk as also the insurance company needs to make a profit. A first estimate for the insurance cost can be obtained using:

$$Insurance = 0.175 \cdot C_{S/C} \qquad [31]$$

This equation indicates that insurance cost on average is 17.5% of the insured value. This is on the premises that in case of a launch failure most launcher providers guarantee a free replacement launch.

3.4 Example sizing

In this section an example is given on the sizing of an Earth Observation spacecraft, but the method can also be applied to the sizing of other spacecraft, be it that, depending on the spacecraft, different estimation relationships are to be used.

The following inputs are used:
- *Payload data:*
  - *Earth observation camera*
  - *Mass: 300 kg*
  - *Power: 280 W average, 790 W peak*
  - *Dimensions: 1.5 m x 1 m x 0.5 m*
- *Mission data:*
  - *Life 10 yr*
  - *Maneuvering: Δv = 800 m/s (incl. 100 m/s for margin + reaction wheel unloading)*
  - *ESA mission*
- *Launcher data:*
  - *Maximum diameter under fairing: 3 m*
  - *Vehicle is injected into final orbit by launcher, so no separate kick stage is needed.*

Example estimation of S/C properties:

1. *S/C on-orbit dry mass is estimated using the on orbit dry mass estimation relationship from Brown (see appendix C). It follows a vehicle dry mass of about 1440 kg*
2. *Propellant mass follows from Δv of 800 m/s and selected propulsion system. Selecting for the rocket exhaust velocity w = 3000 m/s we find a propellant mass of 440 kg and a total vehicle mass of 1880 kg*
3. *No separate kick stage needed (direct launch)*
4. *Adapter mass is calculated using Eq.[12] or [13]. For now we use [12] and select an intermediate value. It follows M_{LVA} = 1.5% of loaded S/C mass = ~ 28 kg*
5. *The launch mass can now be determined and amounts to 1440 + 440 + 28 = 1908 kg*
6. *Spacecraft power is estimated using SMAD power relation for large spacecraft taken from appendix C. As payload power is on average 280 W, this gives a total operating power of 1.85 x 280 W = 518 W*
7. *Spacecraft volume follows using an average spacecraft density of 79 kg/m^3. The value of the mass density is an average number derived for large spacecraft, see the reader, page 30. This gives 1908/79 = 24.2 m^3. Given the maximum diameter of 3 m of the launcher, it follows a spacecraft height of 3.42 m (i.e. $\pi/4$ x 3^2 x 3.42 = 24.2 m^3). Note that here we have determined spacecraft volume based on total launch mass and not say loaded spacecraft mass only. First of all, the difference is only small, but this way we also take into account the dimensions/size of the LVA.*

8. *Solar panel area is 518 W / 100 W/m^2 = 5.2 m^2 ~ 2 (for two wings) x (3.42 x 0.76 m^2)*
9. *Solar panel mass: M_a = 0.04 (518 W) = 20.7 kg*
10. *For the body moment of inertia (kg-m^2), we find using Equation* [33]*:*

$$I_b = 0.01 \cdot M^{5/3} = 0.01 \cdot (1880)^{5/3} = 2864\ kg - m^2$$

    This value can later be used to design the attitude control system (see later section).

    Note that when using the more exact relation [32] with L = L_b (body linear dimension) and L_b = $(M/\rho)^{(1/3)}$ it follows a MMOI for the cubic vehicle of 2596 kgm^2. This is a difference of about 10-15%.

11. *Cost estimate is determined using the relation* [24]*. For a dry mass of 1468 kg (including the LVA), we obtain: 0.3531 $(1468)^{0.839}$ = US\$ 160.2 million (FY 2000)*
12. *Development time is estimated using Table 13. Text indicates that for science missions phase C/D typically ranges from 36-72 months. Taking the mean value, we find for the C/D phase a duration of 54 months. To this we add 20% (11 months) to take into account the phase B and 3 months for the phase A. This gives a total duration of 68 months or about 5-6 years.*
13. *Reliability estimate. In section on reliability it is mentioned that spacecraft failure rate typically is in the 0.056-0.139 serious failure/SC/yr. Using a value for the failure rate of 0.08 (slightly better than average), we find a reliability of 0.45 for the required lifetime of 10 year.*
14. *Risk estimate. Assuming that the payload is already developed and that we only have to develop the bus, it is clear that the highest development risk is related to the spacecraft bus. For this, we take 25% of total cost, which leads to a risk estimate of US\$ 58 million (FY 2000).*
15. *Insurance cost: Insurance cost is estimated based on Figure 23, which indicates an average insurance rate of 17.5 % of the insured value. Question is what the insured value is? We could take of course the cost estimate determined under point 11, but then we forget that this cost estimate includes more than just the production cost of the vehicle. For now we assume production cost is about 50% of the total cost (remaining cost is development cost). As a result, we obtain a total insurance cost of 0.175 x 0.5 x US\$ 160.2 million = US\$ 14.0 million (FY 2000 money).*

In this example, a simple sizing of a spacecraft is performed using simple relations. Of course the outcome will vary depending on the estimation relationships used and the assumptions made. Still, when using proper relationships, the results tend to converge. If not, one should look into other ways of determining first estimates.

Another important limitation in the 'design' method used here is that since it is based on using prior art (i.e. historical data), we must realize that this design method holds limitations for vehicles that incorporate lots of new technology. For instance a breakthrough in technology can lead to a quite different result in terms of mass, power, etc. for the design at hand.

3.5 Quick-look spacecraft configuration

An important activity in spacecraft conception is to draw out a quick-look configuration as it allows for getting a first idea of the overall shape, size and geometry of the spacecraft and for allocating space to the payload(s) and the spacecraft bus. Two quick-look configurations should be considered, being the launch (stowed) and in orbit configuration (fully deployed). The main issue for the launch configuration is that it must fit within the launcher and that its appendages are stowed so that they can survive the launch loads.

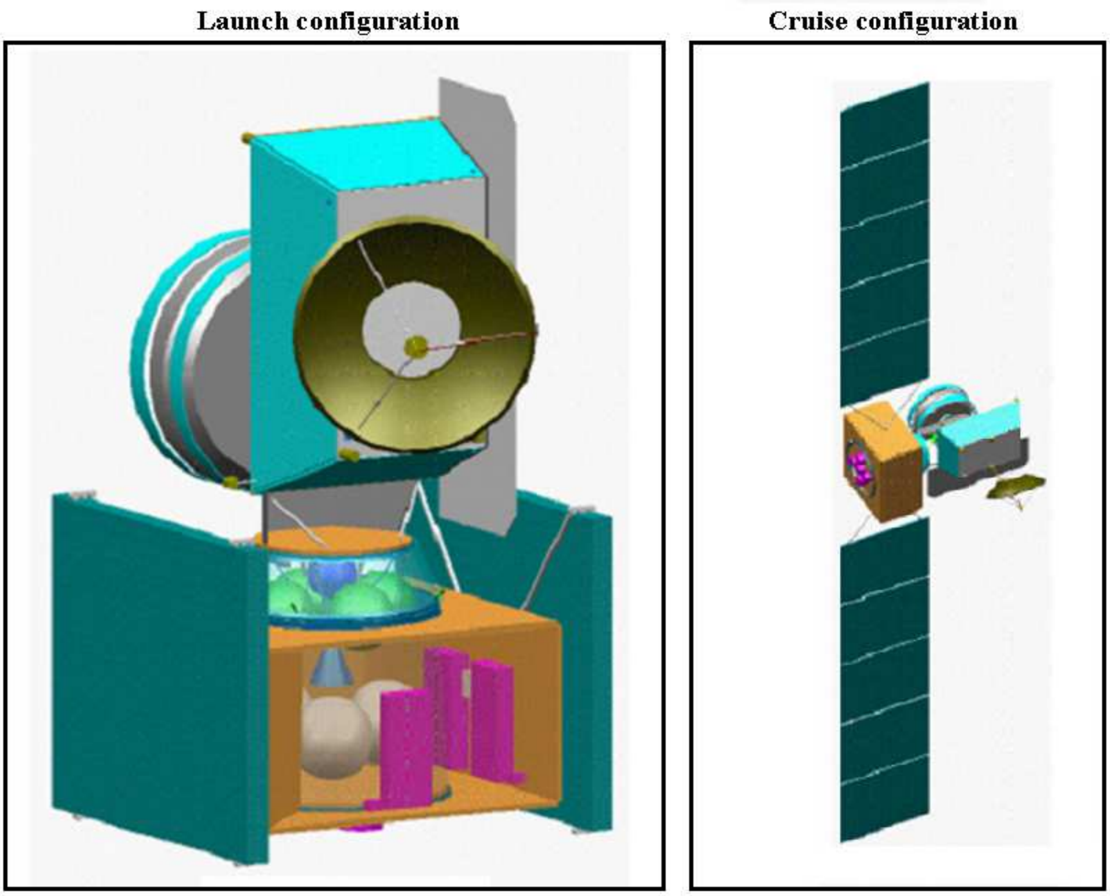

Figure 19: Different configurations of a typical spacecraft (Bepi Colombo), courtesy ESA.

For the cruise or in-orbit configuration it is important that the solar array can be directed toward the incident solar radiation, while also allowing for communications and or correct pointing of the payload when needed.

Initially simple sketches will do, like the ones shown in the next figure, but some measure of the size of the spacecraft (its main dimensions) shall be incorporated.

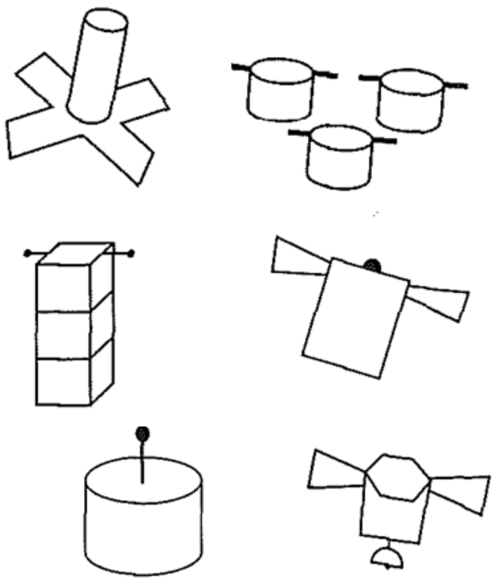

Figure 20: Simple sketches showing spacecraft configuration options

In later stages of the design the drawings will become much more sophisticated also allowing for internal lay-out of the spacecraft as well as antennas, thrusters, solar arrays, and attitude sensors.

Steps in developing a configuration

- Define body axis frame and decide how body axis system is oriented wrt to orbit reference frame (origin orbits with CoM of spacecraft with z-axis pointing to planet about which the S/C is orbiting).
- Locate Sun.
- Draw out the payload instrument(s) and their field of view.
- Identify best location with respect to the body axis system for the payload.
- Allocate volume for the spacecraft bus. Two options might be considered:
  - Payload and platform are two separate items that get integrated right before launch. A modular design allows for the instrument also to be carried on a different mission without large redesign. Only the bus needs to be redesigned.
  - Payload and platform are highly integrated. This generally allows for a much more compact spacecraft, but design changes might turn out to be very costly.
- Select body shape and architecture.
- Sketch a 'quick-look' deployed configuration for payload, solar arrays, and communications antenna even when no sufficient information is available yet.
- Sketch a 'quick-look' stowed configuration and fit payload inside stowed static envelope and identify available bus envelope & volume as well as interface plane with launcher or launch vehicle adapter. Indicate size (diameter) of interface.
- Find stowed locations for deployable appendages and package larger components.
- Assess high-level subsystem requirements such as field of view; identify potential problems (when possible).
- Calculate spacecraft's mass properties.
- Release configuration for detailed subsystem trades and analyses.
- Continue to develop configuration with feedback from subsystem trades.

Example: Size determination
Suppose we have estimated a body volume of 5.69 m^3. Assuming a cubical shaped body, this gives a body of linear dimension 1.79 m. In the next table results are given for three basic shapes as to illustrate their effect. All shapes have identical volume and fit in a cylindrical shaped payload volume of diameter 2.54 m.

Shape of ground plane	***Square***	***Cylindrical***	***Hexagon***
Area of ground plane	*1.79 m x 1.79 m = 3.29 m^2*	*π/4 x (2.54 m)^2 = 5.05 m^2*	*4.84 m^2*
Height of geometry	*1.79 m*	*1.12 m*	*1.17 m*

From this example it is clear that the cubic shaped vehicle requires largest height for storing the spacecraft.

Estimate mass properties
Mass properties estimation is about the location of the center of mass (CoM), the principal axis of the vehicle and the mass moments of inertia (MMOI) about the principal axis of the vehicle. A proper determination of the CoM is important to limit disturbance torques and hence for the control of the vehicle. The mass moment of inertia plays much the same role in rotational dynamics as mass does in basic dynamics, determining the relationship between angular momentum and angular velocity, torque

and angular acceleration. In general we find that the larger the moment of inertia about some axis the more effort is needed to rotate the vehicle about this axis. For more details, see later in this lecture series (ADCS). For now we just focus on determining these properties for some simple bodies. As a first estimate, the CoM can be taken at the geometric center of the body. This though is only valid in case the body is of a homogeneous mass distribution.

Example: Center of Mass determination
You are combining a rectangular payload box with mass of 400 kg, sides of 4 m and height of 2 m with a cubical spacecraft bus of mass 1200 kg and height of 4 m. The payload is placed on top of the spacecraft. At what height is the CoM of the combined spacecraft from ground?

Solution:

$$1200 \cdot a = 400 \cdot (3 - a) \quad \rightarrow \quad a = 0.75 \text{ m}$$

The height above ground level in that case is 2.75 m.

The principal axes are often aligned with the object's symmetry axes and have their origin in the CoM. Mass moment of inertia relations about the principal axes for various bodies can be obtained from the annex E.

For instance for a cubical body of mass M (homogeneously distributed) and linear dimension L follows for the body moment of inertia (kg-m^2):

$$I_b = \frac{1}{6} \cdot M \cdot L^2 \qquad [32]$$

Assuming a body mass density of 79 kg/m^3 (a realistic value), we obtain:

$$I_b \approx \frac{1}{6} \cdot M \cdot \left(0.233 \cdot M^{1/3}\right)^2 \approx 0.01 \cdot M^{5/3} \qquad [33]$$

In case spacecraft mass density differs from the above assumed value and in case of different vehicle shapes (cylindrical, sphere, etc.) the relations in annex E allow for calculating the appropriate values.

Mass moments of inertia
For ADCS it is not the MMOI of the body alone that is important. We should also pay attention to the contribution of appendages and deployable items. These items substantially can increase the MMOI of the complete spacecraft as usually these items have a large distance to the CoM. A first estimate of the effect of a solar wing on the MMOI can be obtained using Figure 21.

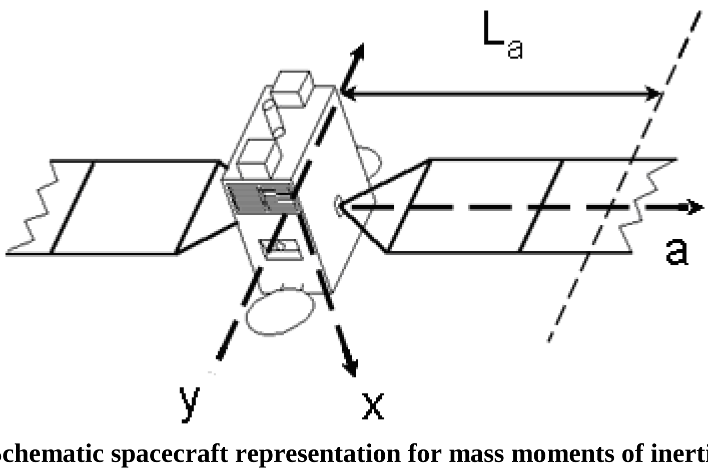

Figure 21: Schematic spacecraft representation for mass moments of inertia calculation

First we estimate solar array mass and size using the relations [18] and [19]. Next we compute:

Area Offset (m) is distance of CoM of an array to CoM of body:

$$L_a = \frac{3}{2} \cdot L + \frac{1}{2}\left(\frac{A_a}{2}\right)^{1/2} \qquad [34]$$

Moment of Inertia (kg-m^2) perpendicular to array face:

$$I_{ax} = \left(L_a^{\ 2} + \frac{A_a}{12}\right) \cdot M_a \qquad [35]$$

Moment of Inertia (kg-m^2) perpendicular to array axis:

$$I_{ay} = \left(L_a^{\ 2} + \frac{A}{24}\right) \cdot M_a \qquad [36]$$

Moment of Inertia (kg-m^2) about array axis**:**

$$I_{aa} = \left(\frac{A_a}{24}\right) \cdot M_a \qquad [37]$$

The Equations [28] to [31] are exact in case:

1. S/C body is of cubical plan form with CoM in geometric center
2. S/C is equipped with two identical arrays on opposite sides of the spacecraft
3. Solar array is of <u>square</u> plan form with CoM in geometric center
4. To prevent shadowing of the solar array the length of the yoke holding the array away from the S/C body is taken equal to the S/C body length.
5. Plane of array coincides with y-z plane of S/C (x-axis is perpendicular to plane of array) with z-axis parallel to array axis.

For background info on mass moments of inertia (MMOI) see for instance Engineering Mechanics by Meriam & Kraige, 1993 or http://emweb.unl.edu/negahban/em373/note18/note18.htm

Example: Spacecraft MMOI
In this example the MMOI about the principal axis of the spacecraft introduced in the example on page 40 are estimated.

Inputs (see results from page 40):
Spacecraft mass (excluding LVA) = 1880 kg
Spacecraft linear dimension = 2.876 m (S/C is assumed cubical with mass density of 79 kg/m^3).
Solar array area = 5.2 m^2
Solar array mass = 20.7 kg

Solution:

1. *Body moment of inertia is calculated using $I_b = 1/6\ ML^2$. It follows a body MMOI of 2593 kgm^2.*
2. *Area off set = 1.44 m + 2.88 m + $(5.2/2)^{0.5}$ = 5.12 m.*
3. *MMOI about S/C body x-axis (perpendicular to array face) = (5.12^2 + 5.2/12) x 20.7 = 553 kgm^2.*
4. *MMOI about S/C body y-axis (perpendicular to array axis) = (5.12^2 + 5.2/24) x 20.7 = 549 kgm^2.*
5. *MMOI about S/C body z-axis (parallel to array axis) = (5.2/24) x 20.7 = 4.5 kgm^2.*
6. *This gives for the deployed S/C MMOI:*
   - $I_x = I_b + I_{ax} = 2593 + 553 = 3146$ *kgm^2.*
   - $I_x = I_b + I_{ax} = 2593 + 549 = 3142$ *kgm^2.*
   - $I_z = I_b + I_{az} = 2593 + 4.5 = 2597$ *kgm^2.*

3.6 Budgeting and design margins

Budgets are lists of elements and a numerical allocation of resources like time, money, volume, mass, power, etc. to each. In spacecraft design, we generally have budgets for:

- Mass
- Size/volume
- Cost
- Power and/or energy
- Reliability
- Risk
- Propellant needed per manoeuvre
- Etc.

Budgets are used to ensure that all of the elements are accounted for and are not counted twice. It also allows for setting proper margins. The latter do account for uncertainty in the estimated resources, which in the early phases of a project are simply guesses based on overall system estimates (top down approach). Taking proper margins will leave room for growth resulting from design definition and development without the need of major redesign. How to set proper margins will be discussed later in some more detail.

An example mass budget is given in Table 14.

Table 14: Example mass budget

Venus Express Mass Budget			
	Power	20	kg
	Propulsion System	30	kg
	Communications	25	kg
	Data Handing System	8	kg
	AOCS	10	kg
	Structure	45	kg
	Harness	6	kg
	Thermal	10	kg
	Payload	48	kg
	Total Dry Mass	202	kg
	Dry Mass Margin	20%	
	Dry Mass Margin	40	kg
	Estimated Dry Mass	242	kg
	Fuel Mass	158	kg
	Fuel Margin	20%	
	Fuel Margin	32	kg
	Estimated Fuel Mass	190	kg
Estimated Total Mass		**433**	**kg**

Budgeting generally requires the collection of historic data on mass, power, size, cost, reliability, failure rate, etc. of the spacecraft and its subsystems. Once this is done, the data needs to be elaborated upon to allow for generating budgets, see later. Preliminary or early budgets usually are guesses based on overall system estimates and will vary greatly as the design evolves. Later budgets will become more stable, will be used to monitor and control the progress of the design, and will be the subject of substantial negotiation. Ultimately, budget numbers will need to be validated by measurement, test, or analysis to ensure that the system will meet its specifications.

Hereafter, a simple method is described for determining a first budget for some spacecraft design related parameters based on historic data using the aforementioned top down approach, i.e. working from the highest level downwards. Before we start though, it is again stressed that we should be careful to apply relationships based on historic data when completely new and/or advanced technologies are considered for which little or no information is available. Still when doing so, appropriate margins (see later section on setting margins) should be taken.

Budgeting for mass
In this section we discuss the generation of a spacecraft mass budget.

Mass budgeting starts with collecting mass data of spacecraft that are of the same category as the spacecraft we are designing. In particular we are interested in the mass of the various subsystems in relation to vehicle dry mass. Once the mass data are available we generate subsystem mass relationships e.g. by averaging or linear regression. It is important to realize that the mass data themselves are not important, but rather the mass ratios or mass percentages for the various subsystems. In general we consider that for a large (high mass) spacecraft the mass of some subsystem may be higher than for a small (low mass) spacecraft. However, if the vehicles are more or less comparable, then the mass percentages are not expected to change greatly and this assumption provides the basis for budgeting.

The next table gives an example of typical mass percentages obtained for a range of (more or less identical) spacecraft[8]. The table provides percentage of spacecraft subsystem (dry) mass in relation to total spacecraft dry mass. The table also includes a row providing for each subsystem the average (percentage) value and a row providing the (sample) standard deviation.

Table 15: Mass percentage data table

Spacecraft		Percentage of Spacecraft Dry Mass by Subsystem						
Name		Payload	Struct.	Thermal	Power	TT&C	AOCS	Propul.
1.	FLTSATCOM 1-5	26.54	19.26	1.75	38.53	2.98	7.01	3.94
2.	FLTSATCOM 6	26.38	18.66	1.99	39.39	2.99	6.77	3.83
10.	GPS Blk 1	20.49	19.85	8.70	35.77	5.84	6.16	3.61
15.	DMSP 5D-2	29.85	15.63	2.79	21.48	2.46	3.07	7.42
16.	DMSP 5D-3	30.45	18.41	2.87	28.97	2.02	2.92	8.66
	Average Values	29.1	21.0	4.2	28.8	4.2	6.0	5.1
	Standard Deviation	6.2	3.3	3.1	5.6	1.6	2.1	2.7
	Avg. % of Payload Mass	100	72	14	99	14	21	17

Based on the average percentages subsystem estimation relationships can be derived. An example of how to generate a mass budget is given hereafter.

[8] Note that not all data collected are included in the table. The full table can be found in appendix D.

Example: Generating a mass budget

*In this example a mass budget is generated for the spacecraft introduced in the example "Sizing of an Earth Observation spacecraft" using the data provided in **Table 15**.*

Inputs (see results/inputs from above referred to sizing example):
Spacecraft dry mass = 1440 kg
Payload mass = 300 kg (~21% of S/C dry mass); This mass is considered to hold no uncertainty.

Solution
1. *As a first step, we compute bus dry mass. It follows a bus dry mass of 1440 kg – 300 kg = 1140 kg.*
2. *From Table 15, we learn that structures is on average 21% of S/C dry mass. This is 21%/70.9% = 29.6% of the dry mass of the bus. 29.6% of 1140 kg is 337 kg.*
3. *Result for the various subsystems are given in the third column of the next table:*

Subsystem	*% contribution*	*Subsystem mass*	*SSD*	*Subsystem mass minus margin*
Structures	*29.6%*	*337 kg*	*47.5 kg*	*303 kg*
Thermal	*5.9%*	*67 kg*	*44.6 kg*	*60 kg*
Power	*40.6%*	*463 kg*	*80.6 kg*	*415 kg*
TT&C	*5.9%*	*67 kg*	*23.0 kg*	*60 kg*
AOCS	*8.5%*	*97 kg*	*30.2 kg*	*87 kg*
Propulsion	*7.2%*	*82 kg*	*38.9 kg*	*74 kg*
Miscellaneous	***2.3%***	***26 kg***		***24 kg***
Margin	*-*	*-*	*-*	*117 kg*
Total	*100%*	*1140 kg*		*1116 kg*

4. *The result in column 3 is without taking any margin into account. To determine the margin, we first calculate the SSD for the various subsystems. For instance, for the TT&C system the **Table 15** shows a standard deviation of 1.6% of S/C dry mass or 0.016 x 1440 kg = 23.0 kg. The results for the various subsystems can be found in the column 4 of the above table. Taking the mass estimates of the subsystems as independent estimates, see appendix E, it follows a total SSD (see page 176) of 117 kg. Subtracting this SSD from 1140 kg gives a mass of 1023 kg to be budgeted. For the structures subsystem this is a mass of 0.296 x 1023 kg = 303 kg. The subsystem masses excluding the margin of 117 kg are given in column 5.*

It is noted that:
- *The row miscellaneous has been added to ensure that totals add up to 100% and 1140 kg. This is because the percentages in table 13 do not add up to 100% (Verify).*
- *For the margin calculation we have settled for 1 SEE, but a higher margin may be taken to reduce the probability of a costly redesign in case mass of the spacecraft becomes too high.*
- *Results need to be adapted in case also the mass estimated for the payload holds some uncertainty. In this example, it was assumed that the payload mass is well known and holds little to no uncertainty.*

As an alternative to averaging the mass percentages, we may also use regression analysis, provided that we can identify (at least) one parameter that has a noticeable effect on the percentage value.

As can be seen from the example mass budget in in Table 14, we need to add the propellant mass to obtain the spacecraft wet mass. Sometimes also other masses are included in the mass budget, like the apogee kick motor (if needed) and the mass of the adapter. This is, for instance, when the mass budget is needed to permit eventual launch vehicle choice.

Budgeting for power, size or volume, and cost

The same method as used for the generation of a first mass budget can also be applied to for instance power, volume and cost budgeting. To give you a head start, we have collected related budgeting data for various types of spacecraft in appendix C. An example power budget is shown in Table 16.

Table 16: Example power budget (adapted from MSG spacecraft [Haines])

Element	Nominal conditions (W)	Peak (W)
Payload (SEVIRI)	166	324
Payload (S&R + data communications)	154	154
Antennas	12	12
Command and data handling	62	62
AOCS	20	20
Thermal control	38	50
TTC	32	32
Total Power	**484**	**654**
Total with 6% margin	**513**	**693**

This example shows the power that goes to the various subsystems. Not included as an element though is the electric power subsystem itself. In most designs, the EPS requires a lot of power as energy is needed to charge the batteries, but also to make up for losses induced in the system.

Reliability budgeting

Reliability budgeting is a bit peculiar as compared to mass, power, volume and cost budgeting. For reliability, this is because it also depends on the life of the spacecraft. To allow for reliability budgeting, we need to know the S/C failure rate and the percentage failures of each of the subsystems. Once these are known, we can use [21] to compute the reliability of the various subsystems. Typical such percentages are given in Figure 22.

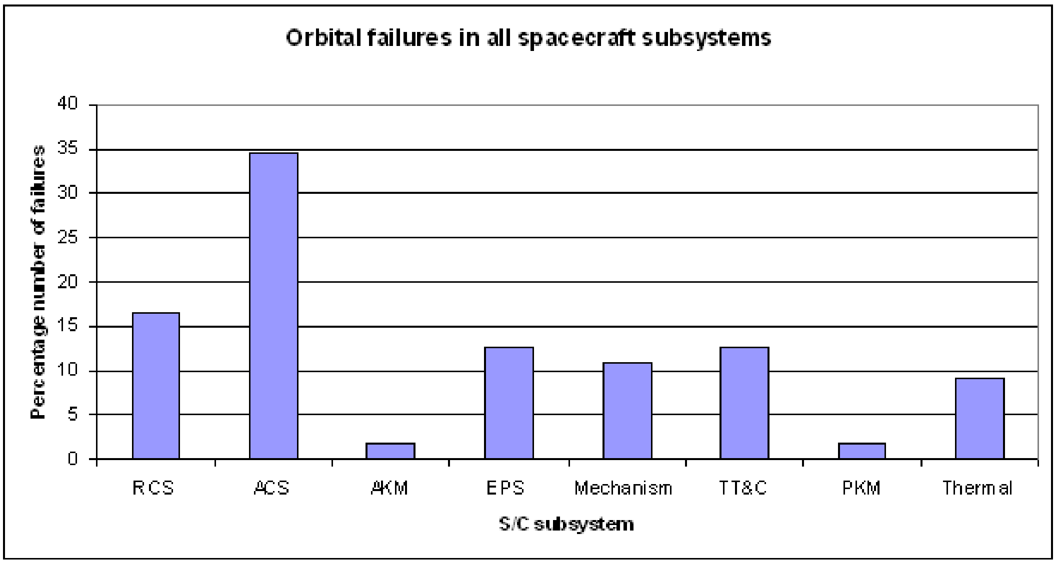

Figure 22: Spacecraft failure rate data per subsystem

From the reliability of the subsystem then also the reliability of the spacecraft itself can be calculated. Using:

$$R = e^{-\sum \lambda_i t} = \prod_i e^{-\lambda_i t} = \prod_i R_i \quad [38]$$

In words it means that total reliability follows from the product of the reliability of the subsystems.

Of course this total reliability should meet the spacecraft reliability requirement.

Example: Reliability budgeting
Consider a spacecraft bus with a reliability of 0.90 over a 10 year design life. Using $R = e^{(-\lambda t)}$ *we find an allowable spacecraft bus failure rate of 0.0105 failures/yr. The reliability of the RCS system can now be determined as follows.*

- *From the figure it follows RCS makes up 16% of all S/C failures. Given the total failure rate of 0.0105 failures/yr it follows for the RCS a failure rate of 0.00169 failures/yr.*
- *Over a 10-year period RCS reliability must be better than or equal to* $R_{RCS}(10) = e^{(-0.00169 \times 10)} = 0.9833$
- *Other systems: Values are calculated likewise and documented in table*

The calculation is slightly more complicated in case no AKM and PKM are foreseen in the design. In that case we first must correct the percentage number of failures of the RCS for the absence of AKM and PKM. In this case the influence of the AKM and PKM is limited as their presence is limited, but in case we leave out mechanisms, the consequence is much more pronounced. This though is left for self-study.

An example reliability budget is given in Table 17.

Table 17: Example reliability budget [Chen]

Element	Reliability
C&DH	0.9147
ADCS	0.9127
EPS	0.9131
TT&C	0.9325
RCS	0.9858
TCS	0.9989
Total S/C	**0.7019**

Risk budgeting
To allow for risk budgeting, we need to find out for every item in the budget the probability of failure (high, low or intermediate). Next we determine the consequence of the failure in terms of schedule delay, or cost over-runs. Once a risk budget is established, it becomes feasible to generate a first estimate of project reserves (in terms of cost and schedule) needed.

Example: Risk estimate
Suppose that for 5 out of 6 subsystems performances can be met quite easily using existing equipment, but for the one remaining subsystem, we need to use some items that have a very low level of development. Assigning values to the probability of failure, like 10% in case of a low probability and 40% in case of a high probability and combining this with data on development time and cost allows us to estimate the risk in terms of cost and schedule. Suppose all systems cost an equal amount of money (each costing C) and assuming that the risks can be considered independent, it follows for the SSD, see annex E, section on dealing with uncertainty:

$$\mathrm{SSD} = \sqrt{5 \text{ x } (0.1 \cdot \mathrm{C})^2 + (0.4 \cdot \mathrm{C})^2} = 0.45 \cdot \mathrm{C}$$

As total vehicle Cost is 6C, we find that the SSD = 7.5% (0.45C/6C) of the total vehicle cost. So to be 98% sure that the project will not be short of funds, we need to put about 2 x 7.5% = 15% of the total money available in reserve. The remaining money can then be given out as constraint to the development of the various subsystems.

The same exercise can also be performed for the development time. The result can then be used to generate a schedule with sufficient built in margin to guarantee some end date.

Propellant mass budgeting
Propellant mass on board of spacecraft can be quite high. When looking for mass reduction, it is good to have an insight in what manoeuvre's require most propellant as a few percentage savings on a large amount of propellant mass may be much more easily accomplished than a large saving on a small amount of propellant. To generate a propellant mass budget, one should go through the following steps:

- Generate a list of required Δv per manoeuvre. A list of typical values for various manoeuvres can for instance be obtained from
- Select type of propulsion system to accomplish the manoeuvre and determine specific impulse (or effective exhaust velocity) for this propulsion system
- Use ideal rocket equation to convert Δv to propellant mass

Adding design margins
Design margins or contingencies are needed to allow for growth resulting from design definition and development.

Design margins can be applied to mass, power, cost, etc., be it that for different parameters different margins may be used.

A common definition for design margin applicable to mass, power and cost is:

$$\text{Design Margin} = \text{Total Capability} - \text{Current Best Estimate} \qquad [39]$$

The margin in % is the margin divided by the total capability times 100 %.

> *Example: Design margin (1)*
> *You have contracted for a launcher capable of launching a S/C of 1000 kg in a low Earth Orbit. The spacecraft currently being designed has a launch mass of 964 kg. The design margin for this spacecraft is 1000 kg – 964 kg = 36 kg or 36 kg/964 kg = 3.7%.*

In the early phases of a project wherein our estimations are not really accurate, it is wise to set the margin(s) not too low as it may force you later in the project to carry through extensive design modifications. For instance in case we have underestimated launch mass, it may be necessary to consider selecting a more capable, but also more expensive launcher.

From [Brown] we learn that a committee of the AIAA has reviewed industry-wide historical data from numerous projects and has generated recommendations for the contingencies to be applied for amongst others mass, and power depending on the design phase. For instance, for a completely new and unique spacecraft in the mass range of 500-2500 kg a contingency of 20% is customary for the conceptual design phase. Some further data can be found in the work of Brown. In general though, design margins:

- are largest for completely new S/C designs
- decrease with increasing design maturity

A more basic way is to use available data on SSD or SEE to determine appropriate margins. In case we know the sample standard deviation (SSD, when averaging) or the Standard Error of Estimate (SEE; when using regression analysis), we may use the known SSD or SEE to generate an estimate for the margin to be applied. In case that the data values are normally distributed about the average, we find that 68% of all probable outcomes are in a range of +/- 1 SSD (or SEE) about the average and 95.8% of all probable outcomes are in a range of +/- 2 SSD about the average. In case of a single sided confidence bound as for example when considering spacecraft mass, we could select a margin equal to two times the SSD (or SEE) to ensure that only in 2.1% of all cases the vehicle will turn out too heavy. Notice that we do not mind if the spacecraft is much lighter as this usually can be easily corrected for. But a spacecraft being too heavy is much more difficult to correct for. Of course if we would like to be even more certain that the spacecraft will not surpass some value, we could take a margin of 3 SSD.

Example: Design margin (2) *Suppose we have determined a mass estimate (MLE) of 2000 kg with a SSD of 12% (or 12% x 2000 kg = 240 kg). To limit the probability of the final mass being higher than the originally estimated mass to say 97.9%, we should design for a vehicle mass of 2480 kg and hence our mass margin is 480 kg or 480/2000 x 100% = 24%.*

For background information on SSD, SEE and confidence bounds, see appendix E.

3.7 Some notes on data collection and data analysis

Collecting data from comparable vehicles
Generating estimation relationships for spacecraft preliminary and conceptual design purposes including preliminary budgeting may require the collection of historic data on mass, power, size, reliability, etc. of the spacecraft and its subsystems. This is especially the case when no suitable estimation relationships are available.

Appendix B provides for a handy collection of spacecraft data that can be used for deriving estimation relationships. These data have been mostly collected from open sources, like:

- Jane's spaceflight directory
- The internet, for instance http://nssdc.gsfc.nasa.gov/nmc/SpacecraftQuery.jsp

Note that collecting appropriate data can take considerable time/effort. Also there is the possibility of misinterpreting data and even of erroneous data. For instance, in case of a deep space probe that next to instruments also carries a lander on board. Question is whether the lander mass should be considered part of the payload or not? So before attempting determining an estimation relationship we should:

- Check for credibility of data source. Design reports, ESA and NASA publications, etc. are more credible than for instance an article in a local newspaper or a marketing folder. However, even when the data source is credible, errors should not be excluded.
- Check for erroneous data; even when source is credible, errors should not be excluded. A typical example is a typist that makes an error when typing in the data for instance by typing the decimal separator after the wrong digit (consider typing 1.0 instead of 10.0). To check for erroneous data, it is best to plot the data and identify clear outliers. Such outliers should then be investigated to determine the reason for why the data is peculiar and only then should one decide to omit the data or not. Make sure that one writes down the reason for omitting any of the data.
- Check if data collected is representative for the vehicle currently under design. As indicated before, launcher data should not be used to design a satellite and vice versa. Also, it is questionable whether we can mix data for nano-satellites with data for very heavy satellites, etc.
- Check if definitions for the variables used are identical. Does total mass mean launch mass or just wet S/C mass?
- Check whether subsystems are defined along identical lines; For example, (parts of) the TT&C subsystem may be included in the (communications) payload, and guidance and navigation either as part of the AOCS or the TT&C system.

Analyzing the data
Once we have collected the necessary data, we can start analyzing the data. Two important methods are:

- Analogy or system similarity based estimating: An estimating *technique* that uses the values of parameters, such as *cost*, *budget*, and *duration* or measures of scale such as size, mass, and complexity from a previous, similar *activity* as the basis for estimating the same parameter or measure for a future activity. It is frequently used to estimate a parameter when there is a limited amount of detailed information about the project (e.g., in the early *phases*). Analogous estimating is a form of *expert judgment*. Analogous estimating is most reliable when the previous activities are similar in fact and not just in appearance, and the *project team* members preparing the *estimates* have the needed expertise. Analogous estimation can be applied essentially at any level of detail in the system and for any parameter, but it has low fidelity.
- Parametric estimating refers to an estimation technique which is based on the premise is that changes in the value of a main variable (for example, payload power) are closely associated with changes in some other variable(s), like total vehicle power or vehicle cost. Parametric estimation relies on the use of relatively simple statistical analysis methods, like regression analysis, to compress large amounts of data into more easily assimilated summaries, which still provide the user with a sense of the content without overwhelming him/her. The most widely used summary statistics are regression curve that provides for a Most Likely Estimate (MLE) and standard (percent) error of estimate (SEE). Even simpler is the use of the arithmetic mean (or equally weighted mean), hereafter referred to as average and standard deviation. One valuable aspect of parametric estimating is the higher levels of accuracy that can be built into it depending on how sophisticated the original data that was built into the estimate turns out to have been. A good understanding and an ability to determine mean and variance or standard deviation of a data series is considered essential for this course. Some summary information on these terms is found in appendix D.

Both similarity based and parametric estimation require us to collect data of more or less similar designs. Hence, when designing a lander vehicle, we might consider collecting data from lander vehicles and not say orbiter vehicles and vice versa. Of course, when sufficient data is available, we might make further distinctions. Important is also to realize that the amount of data to be collected as well as the analysis effort to be spent generally is highest in case of parametric estimation. Still, the latter method has preference, because of its greater accuracy and because less expertise is needed for generating estimations (more student friendly).

To illustrate some of the methods above we will discuss some examples below. We will start by showing the usage of the arithmetic mean and standard deviation in estimation and then move on to regression analysis.

Estimation based on using arithmetic mean and standard deviation
Table 18 provides data collected from literature. We would like to use this data to estimate RCS propellant mass. Two ways can be looked into propellant mass as a function of loaded mass or as a function of dry mass. So we are looking for a relation wherein RCS propellant mass is given as a function of dry mass, i.e. it is assumed that RCS propellant mass is a fixed percentage of vehicle dry mass.

$$M_{RCS} = A \cdot M_{dry} \qquad [40]$$

Table 18: Mass distribution of selected satellites [SMAD], [SSE]

Spacecraft Name	Orbit	Loaded Mass (kg)	Propellant Mass* (kg)	Dry Mass (kg)
1. FLTSATCOM 1-5	GEO	930.9	81.4	849.6
2. FLTSATCOM 6	GEO	980.0	109.1	870.9
3. FLTSATCOM 7-8	GEO	1160.9	109.0	1041.9
4. DSCS II	GEO	530.0	54.1	475.9
5. DSCS III	GEO	1095.9	228.6	867.3
6. INTELSAT IV	GEO	669.2	136.4	532.8
7. INTELSAT V	GEO	1008.0	173.0	835.0
8. INTELSAT VI	GEO	2237.0	430.0	1807.0
9. TDRSS	GEO	2150.9	585.3	1565.7
10. GPS Blk 1	MEO	508.6	29.5	479.1
11. GPS Blk 2, 1	MEO	741.4	42.3	699.1
12. GPS Blk 2, 2	MEO	918.6	60.6	858.0
13. P80-1	LEO	1740.9	36.6	1704.4
14. DSP 15	LEO	2277.3	162.4	2114.9
15. DMSP 5D-2	LEO	833.6	19.1	814.6
16. DMSP 5D-3	LEO	1045.5	33.1	1012.3

*Propellant mass is mostly for attitude and orbit control (RCS propellant)

For the data in Table 18, we find an average RCS propellant mass of 10.5% of spacecraft dry mass with an SSD of 5-6%. When assuming a normal distribution of the data, see appendix E, this means that roughly 95% of all predicted values are within 10-12% (twice the standard deviation) of the predicted value. So for a 1000 kg dry mass, we find an RCS propellant mass of 105 kg. With a 95% probability, the actual value is a value within 92.4 – 117.6 kg. That is how accurate it is!

Now when collecting data the issue might arise on how many data points are to be needed. For instance when taking only the first data point, we have a value of 0.096, when adding the second, the mean value is 0.111 and so on. The issue is what is a statistically meaningful number? This question will be dealt with in another course. For now, it is advised to use at least 10 data points.

Another approach is to determine a weighted average by dividing the sum of propellant masses by the sum of all dry mass. This however, is not considered here and is left for further studies.

Another example of how averaging has been used to determine an estimation relationship is the relation [31]. This relation has been generated using the insurance cost data as shown in Figure 23 for the period 1980 to 2002. Again, it is obvious that the average value only tells part of the story and that from year to year large variations may occur. To also take into account these variations, again the SSD may be used. A very approximate determination gives an SSD of 5.5%, meaning that 95% of all interest rates are in the range 6.5 – 28.5% (average value ± 2 times SSD).

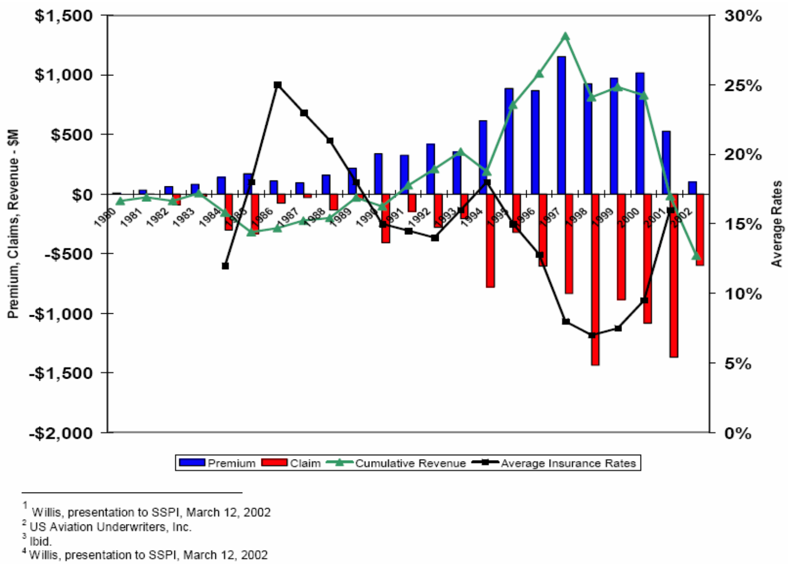

Figure 23: Insurance rates [Futron, 2002]

Estimation using regression analysis and SEE

Besides simple averaging, we might also use regression analysis to determine relationships. Regression analysis is a statistical process for estimating relationships among variables. It allows for a multitude of relationships to be considered other than just the simple averaging discussed in the previous section. For instance, we may also consider non-linear relations, but also relationships where the dependent variable is a function of a multitude of independent variables. Details on linear regression will follow in a later course. For now, we will limit ourselves to the use of for instance Microsoft Excel to determine relatively simple regression relationships. When using Excel to generate regression relationships (trend lines), the data must first be plotted in a graph. Generally the independent variable is along the x-coordinate, whereas the dependent variable is taken along the y-coordinate. Once the data are plotted, we can select the data in the figure and then add a trend line from a number of options. Options include linear, power, exponential, logarithmic and polynomial. A typical example of a trend line plotted in a data figure is given in Figure 24. Below, we first discuss the regression line itself. Following, we will discuss the data spread about this line.

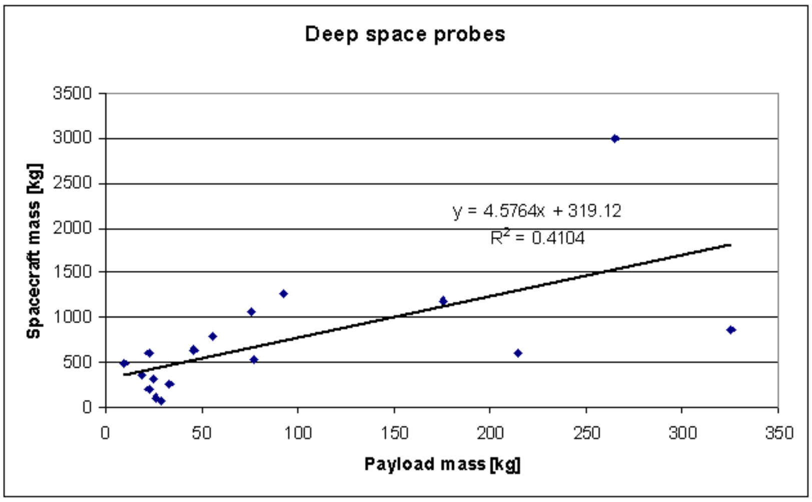

Figure 24: Gross mass of some planetary spacecraft

The relationship found is referred to as estimating relationship and in this case it is a Mass Estimation Relationship (MER). The value estimated for a given value of the independent variable x (here the payload mass) is referred to as Most Likely Estimate (MLE). Notice that other than for a linear relationship determined based on averaging, here the curve not only has a slope, but also an intercept with the y-axis different from zero.

When comparing the actual data with the regression line, it follows that there is a considerable spread of data about this line. Reasons for the spread are that in reality spacecraft mass does depend on more parameters that just payload mass. Consider for instance the effect of launch loads and size (not mass) of payloads on spacecraft mass. Now accepting the spread as being real (no data errors), it is important to have a measure for how well the regression curve fits the actual data. A figure of merit used in Excel to determine the goodness of fit is the R^2 value (R-squared value). The closer this value is to 1, the better the fit; a value of 0 indicates there is essentially no fit. No further explanation is offered. Another more readily understandable measure is the standard error of estimate (SEE), defined as:

$$SEE = \sqrt{\frac{1}{n-m} \cdot \sum_{i=1}^{n} \left(\frac{y_i}{f(x_i)} - 1 \right)^2} \qquad [41]$$

Here n is again the number of observed values, i.e. number of data points, m is number of parameters estimated, y_i is the real or actual value and $f(x_i)$ denotes the function value, i.e. the estimated value at the point i.

The value of "m" depends on the type of curve used. For a linear curve with an intercept at y = 0 we have m = 1 and in case of an intercept different from zero, we have m = 2. To further illustrate the meaning of m, consider determining a linear relationship between y (dependent variable) and x. The linear relation (y = ax + b) has two unknown constants, i.e. the slope a and the intercept b. this means that we need to use two data points (minimum) to solve for the two unknowns. This leaves only (n-m) data points for determining the SEE. Now consider that we have only two data points. This means that both data points are needed to generate the straight line. By definition then the SEE is zero.

It is mentioned that here SEE is defined in a way that the error that results can be considered a relative error and is expressed as a fraction of the estimated value. When multiplying this fraction with 100%, it can also be considered as a percentage error. The reason for considering the relative error instead of the absolute error is that the relative error can be considered more or less constant along the curve, whereas the absolute error varies. This is illustrated in Figure 25 for the dry mass of interplanetary spacecraft in relation to payload mass. The solid line in the figure represents the MER. The small diamonds are the data points. The data spread about the MER is clearly visible. Now, when considering the magnitude of the difference between the real and the predicted value, we find that this increases, with increasing payload mass. However, when considering the relative error, this seems to remain within certain limits. It is for this reason that the SSE is calculated as a relative error. For the data plotted in the figure, the SEE is ±44%.

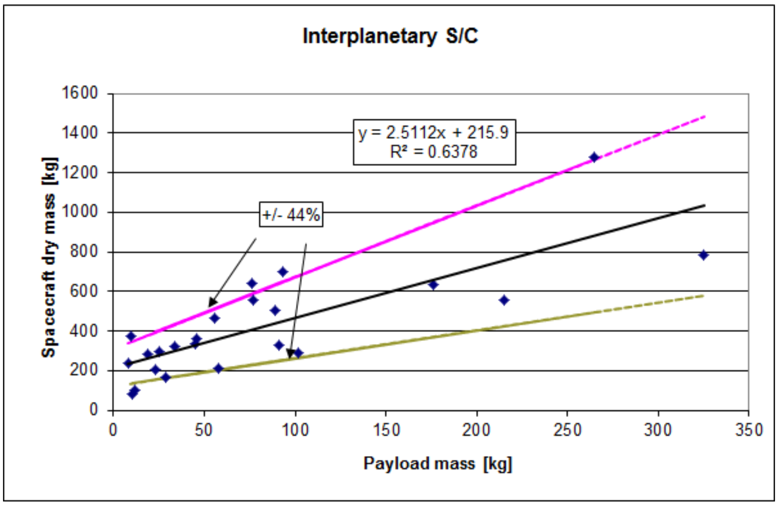

Figure 25: Interplanetary spacecraft dry mass

For improved understanding, we have also plotted two partly dotted lines in the figure showing the ±44% range about the most likely value. Notice that almost all points lie neatly within this range except for 6 points (out of 23). So roughly 75% of all data points lie between these two lines. This agrees nicely with the rule (68-95-99.7 rule) that for a normal distribution 68% of all data points are within one standard error of the mean. For further information on this rule, see appendix E.

For now, we have only considered linear regression lines. Still, also other types of regression curves may be considered, like logarithmic and power curves. For instance, when looking for a relationship between mass and size of a spacecraft, it might be wise to consider a relationship of type Mass = $(\text{Linear dimension})^3$ instead of a linear one. In cases where no such relationship is known, we can also consider relations that give a better fit to the data, but then we should not be tempted to determine estimates for values of the independent parameter beyond the range for which the curve is considered valid.

3.8 Evaluate (and if necessary iterate) design

Iterate, negotiate, and update requirements, constraints and design budgets with feedback from subsystem designers.

- What are key requirements
    - Key requirements are requirements that drive/dominate the design. They cannot be influenced by the designer
- What are key characteristics
    - Key characteristics are parameters that describe what the design looks like, have large influence on cost, development schedule, and risks, and can be influenced by the designer.

Evaluation can also be used to get answers on some or all of the following questions:
- Is launcher capable of delivering spacecraft in desired orbit?
- Does spacecraft fit in launcher?
- Is spacecraft within cost constraint?
- Is spacecraft feasible within the time allotted?
- Is development risk acceptable?
- Can we make the spacecraft sufficiently reliable
- Etc.

Key for the success of the method described in this document is in identifying the spacecraft type and selecting ‘comparable’ vehicles. For instance, it shall be clear that a space launcher is quite different from a satellite orbiting about Earth and hence we should not aim to use data taken from a space launcher to predict satellite properties and vice versa. Best is if we design for instance an orbiter, that we compare it to other orbiters and so on.

Depending on the vehicle to be designed and/or the specific mission, you may decide on other, more fitting/narrow, distinctions between spacecraft.

In addition, if we are not able to answer these questions positively, then we could ask ourselves why not and how our design should be changed to achieve the goals set.

3.9 Problems

A number of problems for exercising upon are available via Blackboard (Maple TA), whereas a few are also contained in a separate workbook available from the TU-Delft online print shop. Of this workbook also an electronic copy is available on blackboard.

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4 System level sizing

The spacecraft bus is traditionally divided into 8 subsystems as given in Table 19 and Figure 26 (see also AE1110-II), and each one has a specific job (function) to do. Systems associated with the main functions have the advantage that work on systems can be done relatively independent and allows good control over the realization of the functions. The principal advantage to a traditional approach is that this is the way most organizations and their expertise are structured. Note that navigation and orbit determination are used interchangeably, like guidance and orbit control. With the introduction of LightSats, extensive on-board computing, and autonomous operations, the traditional allocation of S/C functions is changing for newer satellite designs.

Table 19: Spacecraft subsystem [SMAD] (see also AE1110-II)

System	Principal functions	Other names
Propulsion	Adjust orbit and attitude Manage angular momentum	Reaction Control System (RCS)
Guidance, Navigation & Control (GNC)	Orbit determination and control	Orbit Control System (OCS)
Attitude Determination & Control (ADCS)	Attitude determination and control, Spacecraft pointing	Attitude Determination and Control System (ADCS) or Control System
Communications	Ground communication Spacecraft tracking	Tracking, Telemetry & Command (TT&C)
Command & Data Handling (C&DH)	Command processing Data processing/formatting	Spacecraft Computer System Spacecraft Processor
Thermal	Equipment temperature control	Environmental Control System
Power	Power generation/distribution	Electric Power System (EPS)
Structures & Mechanisms	Support structure, Booster adaptor, Other moving parts	Structure System

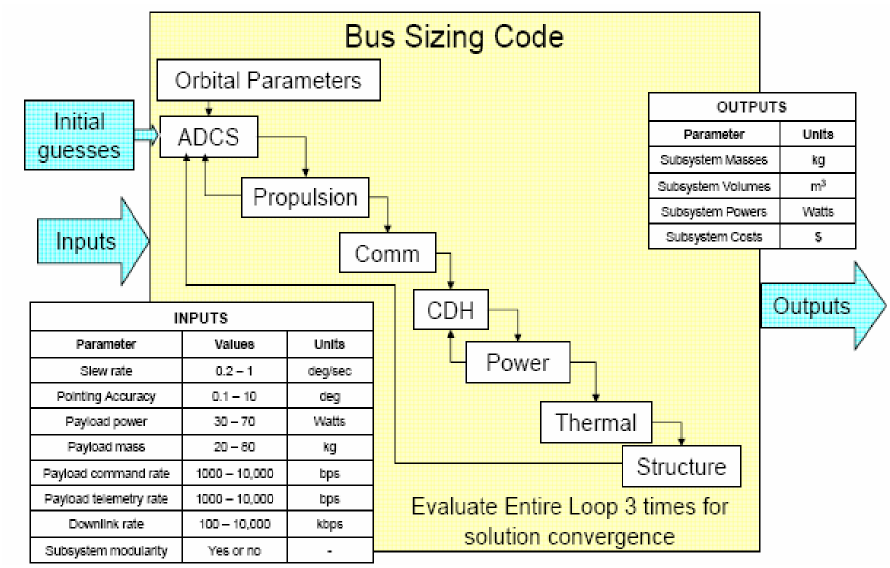


Figure 26: The spacecraft design process

In the next few sections, we will discuss the most important spacecraft systems in detail. For each system, we discuss:

- What they are for (main functions)
- Key (design) issues
- Fundamental design relationships
- How the system looks (what elements make up the system), what major design options exist and possible configuration issues

4.1 Structures and mechanisms

Some of the material offered in this section is based on the material discussed in the course Introduction to Aerospace Engineering II.

Why structures?
From [SSE] we learn that the main function of the structures system is to provide support and to ensure the overall integrity of the entire spacecraft. In more detail, this means to:

- Ensure the proper shape of the spacecraft (prevent deformation)
- Provide for hard points for mounting of equipment
- Provide interconnect to launcher and or other vehicles
- Provide handling hard points (hoisting, transportation)
- Provide protection (débris, radiation, etc.)

Key design issues
Important (key) design issues for spacecraft structures are (see [SSE]):

- Ability to withstand loads
  - Natural frequency sufficiently high to avoid resonance between launcher and spacecraft (stiffness)
  - Strength both for tension and compression

  This is important as we do not want the structure to rupture, fracture, buckle, etc.
- Open structure to allow for good accessibility
- Low mass. The ratio of structural mass versus total mass is a commonly used measure for how good the structural system is:

  $$\delta = \frac{M_s}{M_o} \quad [42]$$

  For a good design the mass of the primary structure is somewhere between 7 and 10 % of spacecraft total mass.
- Materials compatible with space environment (extreme temperatures, presence of monatomic oxygen, etc.). Some materials are unable to survive long term exposure to the extreme conditions of space.

Important design loads
The structure has to be able to withstand pressure loads as well as handling loads, on station loads and launch loads. The latter are usually most important for the structural design (See section 8.2.4 from [FSS]) and include:

- Quasi-Static (or steady state) Loads (QSL), see for a more detailed description [SSE].
- Dynamic Loads/Vibrations (induced by shocks, acoustics like the sound waves produced by the rocket engines, etc.)

An important category of loads are the launch loads. Typical launch loads can be obtained from various sources. An important source is the launcher manual. For each launcher typically a many page document is available describing the launcher's capabilities, the launch site, the launch loads, etc. A summary document containing important information on a range of launchers is the ESA launch vehicle catalogue of which a copy is available on the course's blackboard site.

An example of a load diagram is given in Figure 27. It shows the acceleration load (a quasi-steady state load) during the launcher ascent. Acceleration loads are increasing towards the burn-out of a stage as the mass of the vehicle reduces.

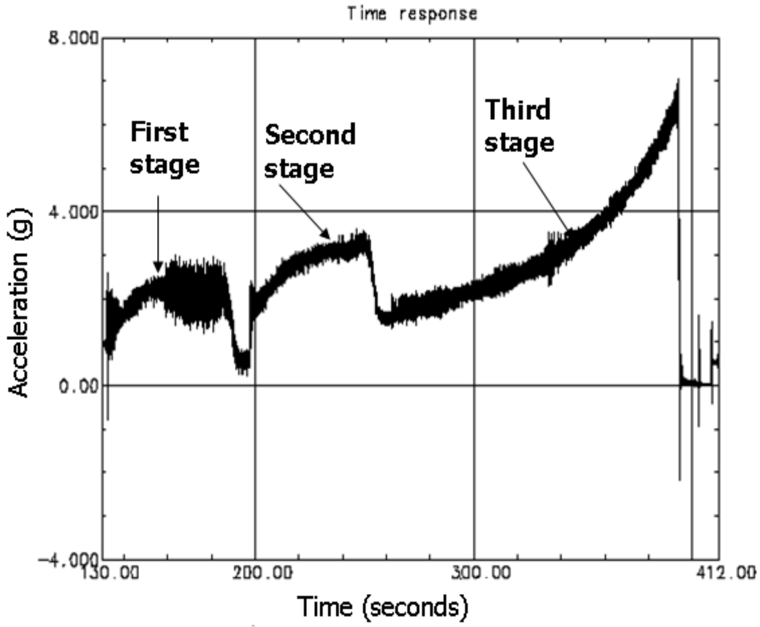


Figure 27: Typical acceleration loads during launcher ascent

Next to launch loads also other loads should be taken into account, like thrusters that are being activated on board or propellant sloshing or an unbalanced reaction wheel.

Use of safety factors

Factor of safety (FoS) or safety factor9 is a term describing the structural capacity of a system beyond the applied loads or actual loads. They are used to increase the reliability of the structure to acceptable levels, see for instance Figure 28.

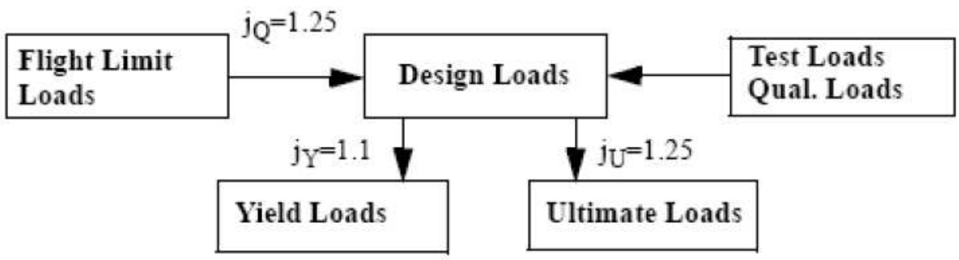


Figure 28: Factors of Safety (FoS) for metallic structures in case of verification by testing Limit load = Maximum load that can occur in service/flight (i.e. over the life of the vehicle)

- j_Q = design safety factor
- j_Y = yield load (or proof) safety factor
- j_U = ultimate load safety factor
- Design load = Qualification load
- Yield load = Design load x j_Y
- Ultimate load = Design load x j_U

FoS vary with the (type of) material used and the verification method (test or analysis only). Typical values used for different types (manned, unmanned, etc.) of spacecraft and different materials and

[9] There are two distinct uses of the Factor of Safety: One as a calculated ratio of strength (structural capacity) to actual applied load. This is a measure of the reliability of a particular design. The other use of FoS is a constant value imposed by law, standard, specification, contract or custom.

verification philosophies have been collected in amongst others [ECSS-30]. The next table is a typical such example.

Structure type and sizing case	FOSY	FOSU	FOSY for verification by analysis only	FOSU for verification by analysis only	Additional factors [a]
Metallic structures	1.1	1.25	1.25	2.0	
Composite structures, Uniform material, brittle		1.25		2.0	
Sandwich structures: - Face wrinkling - Intracell buckling - Honeycomb shear		 1.25 1.25 1.25		 2.0 2.0 2.0	 1.2 1.2 1.2
Glass structures		2.5		5.0	
Composite structures discontinuities		1.5		2.0	
Joints and Inserts		1.25		2.0	1.2
Global buckling		2.0		2.0	
[a] For application of these factors, [E54]see 4.5.17.2 d)					

Figure 29: FoS unmanned spacecraft [ECSS-30]

Study [SSE] to learn about terms like reserve factor and the Margin of Safety of a structure as measures to express the adequacy of a structure to resist the loads. From the reading you should obtain an ability to define these terms and to apply them to existing structures.

Design requirements
Study [SSE, chapter 8.2].
Spacecraft structures (examples)
Figure 30 shows the structure of the MSG (Meteosat Second Generation) satellite. It consists of a central cylinder (the backbone) to which a number of platforms are attached supported by struts. Other elements include lower closing support, sun shade, fluid tanks (pressure vessels), etc. The latter are not so much important for the total load carrying capability of the spacecraft, but still they need to carry some loads.

In general, we divide the structure in:

- ***Primary*** structure or main structure, whose purpose is to transmit loads to the base of the satellite through specifically designed components (central tube, honeycomb platform, bar truss, etc.). This structure provides the attachment points for the payload and the associated equipment. Failure of the primary structure leads to a complete collapse of the satellite.
- ***Secondary*** structure such as baffle, thermal blanket support and solar panel must only support themselves and are attached to the primary structure which guaranties the overall structural integrity. A secondary structure failure is not a problem for the structural integrity, but it could have some important impacts on the mission if it alters the thermal control, the electrical continuity, and the mechanisms or if it crosses an optical path.
- ***Tertiary*** structure or ***flexible*** appendages such as antenna reflectors and solar arrays have generally low resonant frequencies (0.5-2Hz) which interact directly on the dynamic behaviour of the satellite and require a special care for design.

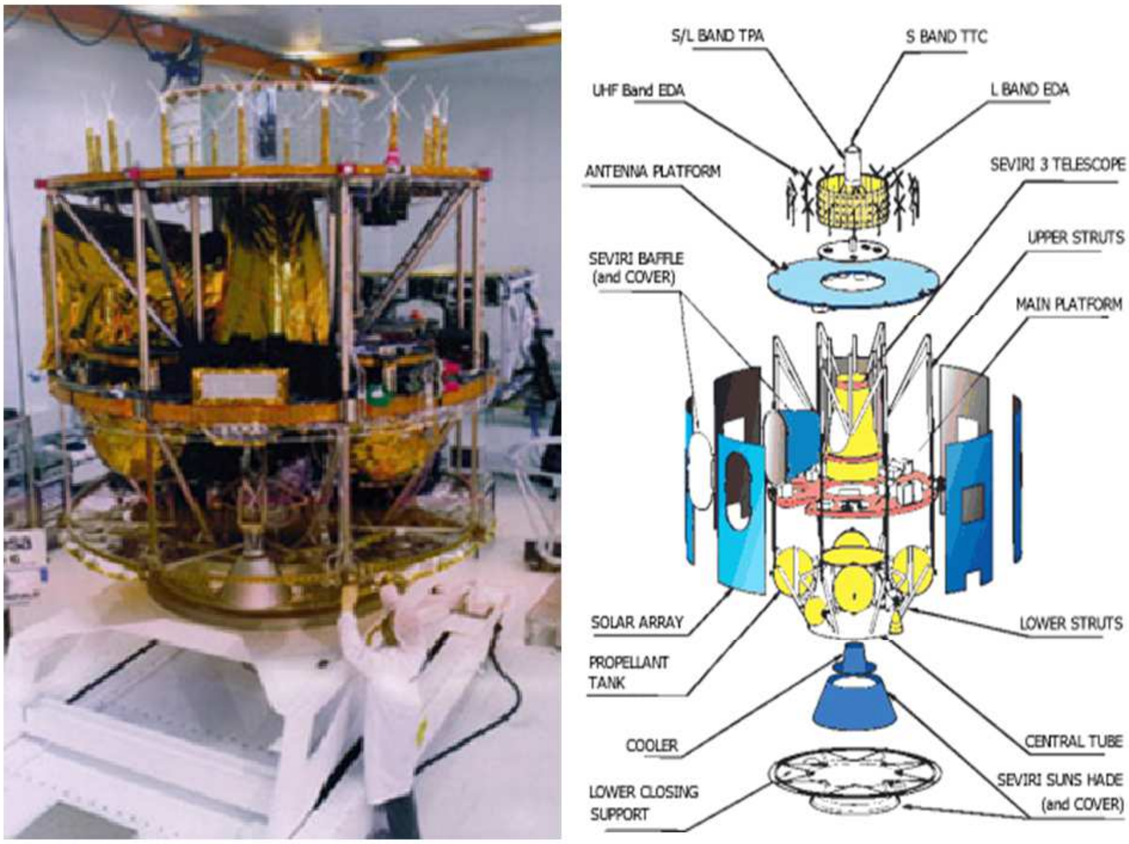


Figure 30: Meteosat Second Generation (MSG) Structure (courtesy ESA)

In the present work, we will primarily focus on the primary structure (i.e. the load carrying structure) including panel like structures (solar arrays). Figure 31 shows some more design examples of spacecraft structures with focus on the primary structure.

It shows on left a central cylinder for a large space platform (Spacebus-4000) being placed on top of a payload adapter or LVA. The CFRP Central Cylinder has a diameter of 1.2 meters, a height of 4 meters and weighs less than 90 kg. Total mass supported is up to 6 tons. Figure on right shows the primary structure of Hipparcos satellite consisting of a central cylinder to which a number of horizontal platforms are attached supported at the outer edge by a number of struts and with LVA attachment ring on bottom.

The following main structural elements are usually distinguished:

- Cylindrical primary structure with plates and beams to allow for mounting equipment/instruments
- Cylindrical/conical Launch Vehicle Adapter
- Pressure vessels (propellant tanks)
- Boxes

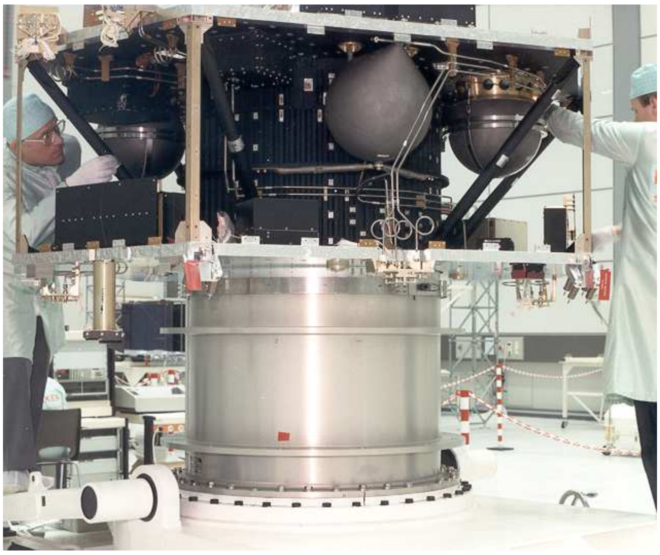


Figure 31: Spacecraft example structures

Spacecraft materials

Study the section on material selection in [SSE]. In this study you should aim to:

- identify important materials used
- describe important material characteristics
- explain how these characteristics affect strength and stiffness of a structure, see also below.

Analysis

Analysing a structure can be quite complex requiring the use of finite element methods, see [SSE]. Early in the design, though, simple methods are used to get a feel for what is important and where the major difficulties are to be expected. To this end, we will introduce in this section a simple single degree of freedom (dynamic) system. In doing so, we will use various relations that have been introduced in AE1110-II.

<u>Design for stiffness</u>

To avoid resonance, the spacecraft should be sufficiently stiff. For instance, when considering the next case, everybody understands that the larger the base, the stiffer the structure will be.

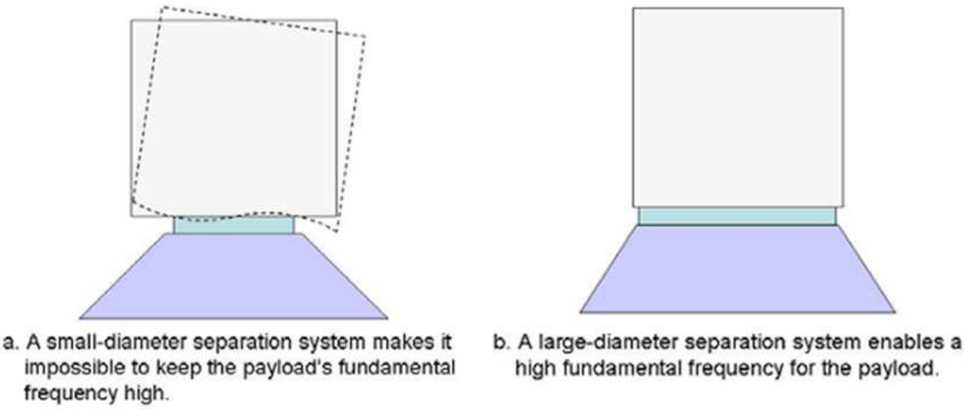


Figure 32: Effect of diameter of spacecraft separation system on stiffness

From experience, we know that large and lightweight structures generally have low stiffness. To increase the stiffness of such structures we need to add mass and/or reduce the size of the structure.

To analyze the stiffness of some structure in a simple way consider the beam approximation as is shown in the next figure (left). It may represent for instance:

- S/C with mass M mounted on launcher at height L
- S/C with mass M mounted on launch vehicle adapter of height L
- S/C with central structural cylinder of height L and with mass of S/C concentrated in point mass on top (this situation may seem somewhat unrealistic, but the results are considered to allow for a conservative design, meaning a safe and maybe too heavy, design)
- S/C bus with payload mass M mounted on top

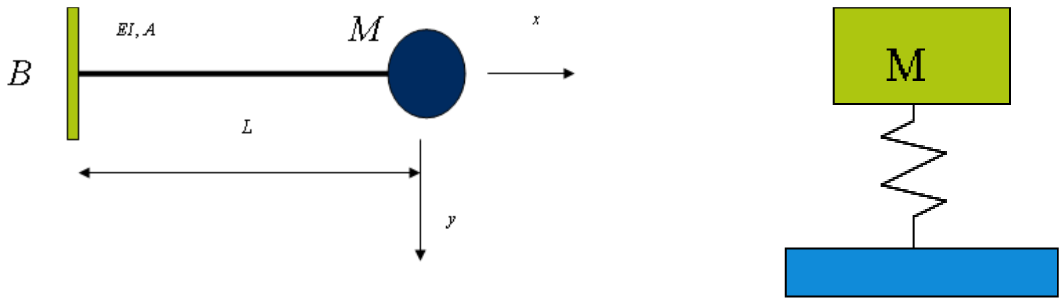


Figure 33: Beam approximation

Here:

- M = mass
- L = length of beam
- A = cross-sectional area of beam
- E = Young's modulus
- I = area moment of inertia of beam (see annex F for area moments of inertia of typical geometries)
- x = longitudinal direction
- y = lateral direction

To allow for modeling of the natural frequency, we schematize the beam system as a mass-spring system as shown in the same figure (right). From our experience we know that when compressing the spring and releasing it, the mass will start moving up and down with some frequency. From the equations of motion, a relation for the natural frequency of the mass-spring system can be derived (see course AE1110-II).

$$f_n = \frac{1}{2\pi}\sqrt{\frac{k}{M}} \quad [\mathrm{Hz}]$$
$$f_n = \sqrt{\frac{k}{M}} \quad [\mathrm{rad/s}] \qquad [43]$$

With k = spring constant. The spring constant depends on whether we consider vibrations in the lateral (perpendicular to the beam axis) or longitudinal direction:

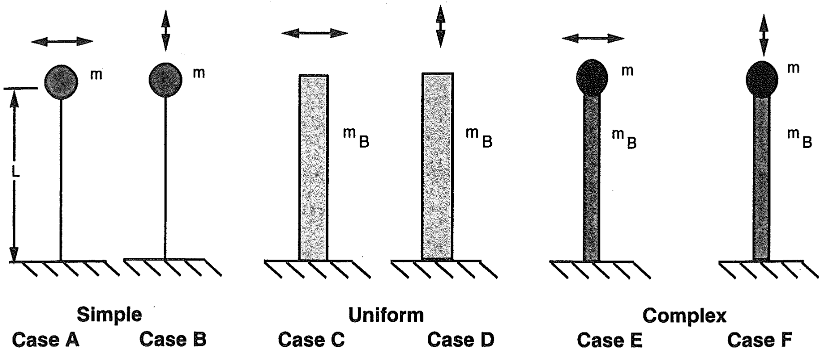
- lateral direction:

$$k_y = \frac{3EI}{L^3} \qquad [44]$$

- longitudinal direction:

$$k_x = \frac{EA}{L} \qquad [45]$$

When designing for stiffness, one should design the structure (dimensions and material selection) such that the first natural frequency (Eigen frequency) is much larger than the excitation frequency. The next figure taken from [SMAD] and also presented in AE1110-II gives some further relations for natural frequency (in [Hz]), demonstrating that the simple beam approach used here (case A and B in the figure) gives lowest natural frequency and hence also the heaviest design. Bringing more detail in the analysis (cases C to F) results in a reduced mass of the structure.



CASE A: Lateral Beam	**CASE C: Lateral Beam**	**CASE E: Lateral Beam**
$m_B = 0 \quad T_p = m$	$m_B \neq 0 \quad T_p = 0$	$m_B \neq 0 \quad T_p = m$
$\delta = 0.333\left(\frac{mL^3}{EI}\right)ng$	$\delta = 0.125\left(\frac{m_B L^3}{EI}\right)ng$	$\delta = \delta_A + \delta_C \equiv \frac{L^3}{EI}(0.333\,m + 0.125\,m_B)ng$
$f_{nat} = 0.276\sqrt{\frac{EI}{mL^3}}$	$f_{nat} = 0.560\sqrt{\frac{EI}{m_B L^3}}$	$f_{nat} = 0.276\sqrt{\frac{EI}{mL^3 + 0.236 m_B L^3}}$
CASE B: Axial Beam	**CASE D: Axial Beam**	**CASE F: Axial Beam**
$m_B = 0 \quad T_p = m$	$m_B \neq 0 \quad T_p = 0$	$m_B \neq 0 \quad T_p = m$
$\delta = \frac{mL}{AE}ng$	$\delta = 0.5\left(\frac{m_B L}{AE}\right)ng$	$\delta = \delta_B + \delta_D \equiv \frac{L}{AE}(m + 0.5\,m_B)ng$
$f_{nat} = 0.160\sqrt{\frac{AE}{mL}}$	$f_{nat} = 0.250\sqrt{\frac{AE}{m_B L}}$	$f_{nat} = 0.160\sqrt{\frac{AE}{mL + 0.333\,m_B L}}$

n = load factor
g = gravitational acceleration
m_B = mass of the beam (uniformly distributed)
T_p = tip mass
I = area momentum of inertia of the beam's cross-section
E = the modulus of the elasticity
A = cross-sectional area of the beam

Figure 34: Beam deflection δ and natural frequency f_{nat} as function of beam and tip mass

Example: Sizing for stiffness
Consider launching a 1000 kg S/C using the Ariane 5 rocket. From the ESA [LVC] we learn that the longitudinal natural frequency of the payload must be in excess of 31 Hz. No value is known for the lateral natural frequency.

Suppose we select for our spacecraft a central cylinder of diameter 2 m and length 4 m as the primary structure and Aluminum with a Young's modulus of 70 GPa as the main construction material. Selecting an effective wall thickness of 4 mm, we find for the natural frequencies of our satellite:

Longitudinal (axial) direction:
Cross-sectional area: $A = \pi D\, t = 0.025\ m^2$
Spring constant: 4.4E8 N/m
Natural-frequency of this satellite:

$$f_{nat} = 0.160\sqrt{\frac{AE}{ML}} = 0.160\sqrt{\frac{0.025m^2 \times 70GPa}{1000kg \times 4m}} = 105.8Hz$$

Lateral direction:
Area moment of inertia (see also annex F): $I = \pi r^3\, t = 0.0126\ m^4$

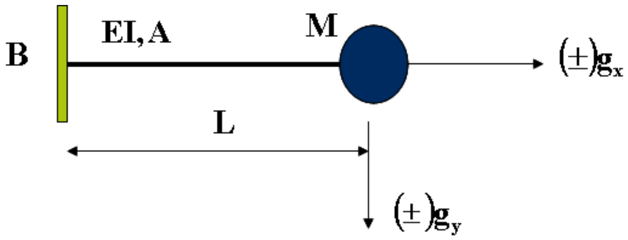
Spring constant 4.1E7 N/m
Natural frequency is 32.2 Hz or 203.1 rad/s

Comparing the calculated longitudinal natural frequency of the satellite with the value as specified based on the launcher, we find that it is substantially above. This is good, however a thickness of 4 mm is quite thick, leading to a mass of the central cylinder of about 270 kg (see for aluminum density, [SSE, table 8.8]). Compared to the total mass of the spacecraft (1000 kg) this is quite substantial. So we should seek ways to reduce spacecraft mass.

The launcher is very important to consider with respect to excitation frequencies. However, next to launcher vibrations also other sources of vibrations and the resulting excitation frequencies should also be considered. For instance:

- A thruster that is activated shortly every 10 s has an excitation frequency of 0.1 Hz.
- An unbalanced reaction wheel rotating at 6000 rpm has an excitation frequency of 100 Hz.
- Propellant sloshing may induce frequencies of order 0.7 Hz and less (from: Orbital Investigation of Propellant Dynamics in a large Rocket Booster, NASA TN D-3968

<u>Sizing for strength (see also fig. 8.14 of [SSE])</u>
When sizing for strength, we must consider stress at point B and buckling load. We use the same simple model as used earlier, but now also take into account the quasi-steady loads (QSL), represented here by the product of mass M and the load factor g (or n).



The stress at point B can be calculated using:

$$\sigma_{tot} = \frac{g_y MLc}{I} + \frac{g_x M}{A} \qquad [46]$$

The first term in the above relation gives the stress due to bending in the structure with the product of g_y and L being the bending moment and c is extreme fibre distance (maximum height in y direction) of the beam, whereas the second term gives the axial load (limit load). Multiplying the stress at point B with the cross-sectional area A gives the equivalent axial load, P_{eq}.

The equation for axial stress is:

$$\sigma_{ax} = \frac{P_{eq}}{A} \qquad [47]$$

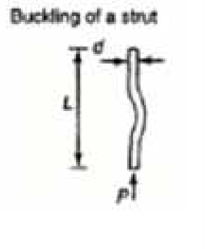
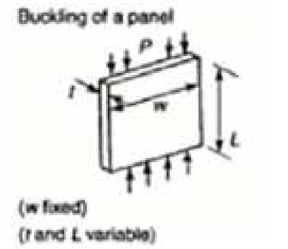
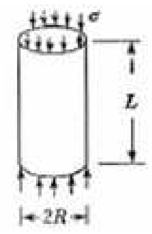
Using axial stress, we can size the cylinder for maximum ultimate and yield stress using appropriate FoS.

For buckling of the simple beam structure shown before, the critical load is given by the Euler buckling load:

$$P_e = P_{cr} = \frac{\pi^2 EI}{4L^2} \qquad [48]$$

Some common cases of interest (introduced in AE1110-II) can be found in the next table.

Table 20: Buckling relations of simple geometries

Case			
Characteristic equations $\sigma_{cr} = \frac{P_{cr}}{A}$	$P_{cr} = \frac{\pi^2 E I}{(L')^2}$ $I = \frac{\pi \cdot d^4}{64}$	$\sigma_{cr} = \text{constant} \frac{E}{(1-\nu)^2} \cdot \left(\frac{t}{w}\right)^2$	$\frac{\sigma_c}{E} = 9\left(\frac{t}{R}\right)^{1.6} + 0.16\left(\frac{t}{L}\right)^{1.3}$
Nomenclature	P_{cr} is critical buckling load, E is Young's modulus, L' is effective length of strut (column), ν is Poisson's ratio, σ_{cr} is elastic buckling stress, I is area moment of inertia		

Compare the above table also with the figure 8.14 [SSE]. Notice that the relation for critical load of a strut here is expressed in the effective length L'. The value of the latter depends on the columns end conditions. Taking L' equal to 2L (For one end fixed and the other end free to move laterally) the relation [48] results. For a further discussion on the end conditions, consult e.g. [SMAD] or http://www.efunda.com/formulae/solid_mechanics/columns/columns.cfm.

Example: Sizing for strength
Consider a spacecraft with a cylindrical central structure of length 5 m, diameter 1 m and a wall thickness of 0.5 mm. On top of this cylinder the payload is concentrated with a mass of 250 kg. Given are:

- *Structure material Young's modulus E = 70 GPa,*
- *Ultimate tensile strength: $\sigma_{ultimate}$ = 400 MPa,*
- *QSL: Maximum axial = -6 g (compression only), maximum lateral = $\pm$1.5 g,*
- *Design Factor of Safety is 1.25.*

The maximum stress in the cylindrical structure (compression or tension) can be calculated using [38]:

$$\sigma_{tot} = \frac{g_y MLc}{I} + \frac{g_x M}{A}$$

We first compute the design loads. We obtain:

- *In x-direction: Mg_x = 250*9.81*6*1.25 = 18394 [N]*
- *About z-axis(bending load): MLg_y = 250*9.81*5.0*1.5*1.25 = 22992 [Nm]*

Next we compute the cross-sectional area A and the second moment of inertia I. It follows:

- *Cross sectional area: A = π x D x t = 0.00157 m^2, and*
- *Second moment of area: I = π x r^3 x t = 0.00020 m^4*

Filling in numbers gives for the axial stress: σ = 69.2 MPa

Checking for tensile strength, we find that the axial stress occurring in the structure is well below the limit value for ultimate tensile stress, even when taking into account a safety factor of 1.25. In a real case we should also check if also the yield strength is sufficient, but here no yield strength is given.

*Checking for compressive strength, we find using the relation for critical stress for a cylinder from **Table 20**, a critical stress of 0.83 MPa. This is about a factor 100 below the compressive stress experienced by the cylinder and hence the cylinder fails if not strengthened.*

Sizing for internal pressure

Pressure vessels must be sized to withstand the internal pressure. This will be dealt with in some detail when discussing launchers. It compares well though with the sizing of a pressure cabin of an aircraft. An example problem can be found in the workbook.

Mechanisms (for self-study)

Need for mechanisms:

- For deployment of solar arrays, radiators, and booms. Some spacecraft are thus large that they would not fit in the payload bay unless specific measures (mechanisms) are employed
- To allow for pointing (receive and transmit antennae, solar panels, thrusters, etc.
- To allow for a scanning motion

In the design of a spacecraft, the selection of the required mechanisms to perform the desired tasks of the spacecraft mission becomes an important factor in the design process. Due to spacecraft being designed for a multitude of missions, a wide variety of mechanisms will be needed. For most space missions there are three main mechanisms that stand out as the most important and necessary for spacecraft functionality. These mechanisms are: the payload release mechanisms, the solar array deployment mechanisms, and the antenna deployment mechanisms. Little is known of the final size and costs of such devices, because most of these devices are constructed for the particular needs of the mission. An overview of typical spacecraft mechanisms is given in Table 21.

Table 21: Typical space mechanisms

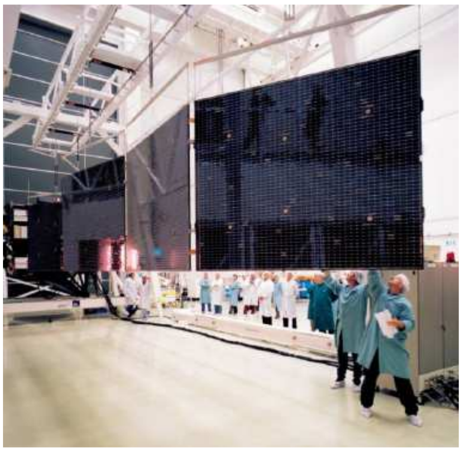
	**Solar panel deployment mechanism** To deploy the initially stowed (during launch) solar panel
	**Solar array drive mechanism** To align the spacecraft's solar panels towards the Sun

Table 21: continued

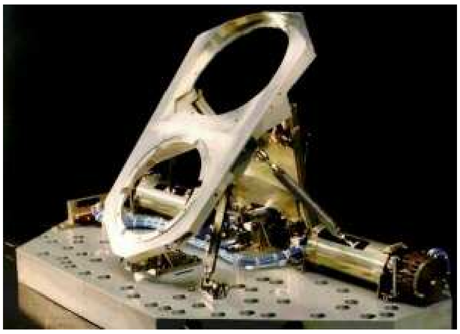
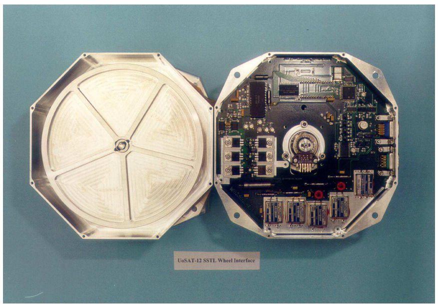
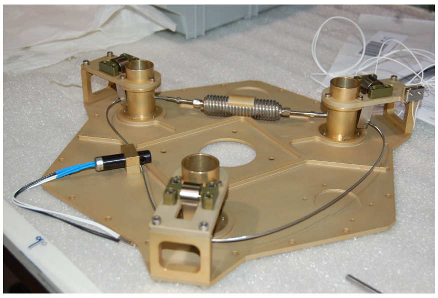
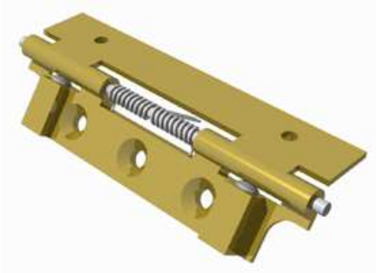
	**Pointing mechanism** For pointing of antennae, antenna dishes, thrusters, etc.
	**Reaction wheel** A type of wheel used primarily by spacecraft to change their angular momentum without using thrusters
	**Momentum Wheel** Used for gyroscopic stabilization of spacecraft. Momentum wheels have high rotational speed of around 6000 rpm.
	**Separation mechanism** Used for separation of a.o. YesSAT (ensures separation velocity of 2.1 m/s). It consists of three preloaded springs, 3 hooks that keep the springs in place, a steel cable that holds the hooks and a cable cutter that cuts the steel cable, thereby releasing the hooks and consequently the springs.

Table 21: continued

	**Inflatable solar collector** To collect a large amount of (solar) light.
	**Delfi-c^3 hinge (development model)** Used for deployment of solar panels. It features a torsion spring capable of driving the hinge from the stowed to deployed position. Deployment angle is 35 degrees.

Mechanisms are usually distinguished after the number of operations:

- One shot devices
    - Separation systems
    - Pyro valves (fluid valves that are activated by igniting a small explosive charge)
    - Deployment devices (for solar arrays, radiator surfaces, antennas)
- Continuously and intermittently operating devices
    - Solar array drive systems (for power control)
    - Electric switches (for power control)
    - Momentum and reaction wheels (for attitude control)
    - Louvers (for thermal control)
    - Control valves, regulators (for fluid flow control)
    - Antenna pointing mechanisms (for communications link control)

Another distinction is in the way the mechanisms function:

- Translating mechanisms: Telescopic booms, springs for spacecraft separation, fluid valves, etc.
- Rotating mechanisms: Hinges, reaction wheels, momentum wheels, solar array drive, pointing mechanisms, fluid valves
- Oscillating mechanisms: Scanners
- Inflation

Key elements of Space Mechanisms include:

- Actuator (Spring/stored energy devices, motor)
- Flexible joints (slip rings, hinges, bearings, etc.)

- Sensor
- Release devices

Key issues for mechanisms are:

- Friction: No lubricants allowed???
- Energy must be stored (spring) or added to keep it working

Problems
A number of problems for exercising upon are available via Blackboard (Maple TA), whereas a few are also contained in a separate workbook available from the TU-Delft online print shop. Of this workbook also an electronic copy is available on blackboard.

4.2 Thermal Control

The purpose of the thermal control system is to keep temperatures of all spacecraft components within allowable ranges throughout the mission at lowest possible expense.

Why thermal control?

- To prevent overheating and under cooling (compare human body; human body works best if core temperature is between 36.1 to 37.8 degrees Celsius (^oC)
  - o Most equipment designed to function at Earth temperatures
  - o Propellants might freeze or start boiling; batteries have no capacity left, etc.
- Large temperature differences between different locations may cause distortion of instrument and/or sensor alignment
- Large temperature differences over time may cause calibration errors

Thermal control requirements

Requirements are needed so that we know what to design for. Such requirements generally stem from history and/or from manufacturer data. For instance, the next two tables provide for an overview of typical temperature tolerances of some spacecraft components as well as allowed temperature gradients and stable temperatures.

Table 22: Typical spacecraft component temperatures [SSE]

S/C component	**T_{min} (^oC)**	**T_{max} (^oC)**	**ΔT (^oC)**
Batteries	0	+20	20
Solar arrays	-105	+110	215
Sensors (sun, Earth, magnetic field, etc.)	-30	+50	80
Thrusters	+7	+65	58
Mechanisms	0	+50	50
On board computers	-10	+50	60
Transponders/transmitters/receivers	-20	+60	80

Table 23:Typical allowed temperature gradients and temperature variations for some components

- limited Temperature Gradients
  - ΔT < 5°C across optical instrument (1.5 m)
  - $\Delta T/\Delta x$ < 2°C/m for structural element
  - ΔT < 5°C between MMH and NTO tanks
- Stable Temperatures
  - $\Delta T/\Delta t$ < 5 K/h for typical electronic unit
  - $\Delta T/\Delta t$ < 0.1 K/mn for CCD camera
  - $\Delta T/\Delta t$ < 100 μK/mn for cryogenic telescope

Other requirements stem from mass/cost/power considerations etc. For instance, the following characteristics can be obtained from literature

- Appearance: > 95 % spacecraft exterior
- Mass: 2 - 5 % of spacecraft dry mass
- Cost: 3 - 5 % of spacecraft cost
- Power: < 5 % of total spacecraft power

All above introduced values are indicative only and actual figures should be obtained from manufacturer or equipment specifications and or from other comparable spacecraft. Still such values may serve to provide direction to the design, see hereafter, but first we will introduce some basic heat transfer fundamentals.

Heat transfer fundamentals (see also physics course)
Heat transfer deals with transfer of thermal energy from a medium with high temperature to a medium with low temperature. The amount of heat transferred per unit time is usually referred to as heat transfer rate ‘Q’ and is expressed in (J/s or W). In case we consider the heat flow per unit surface area ‘A’, we refer to this as the heat flux ‘q’ and is expressed in (J/(s-m^2) or W/m^2):

$$q = \frac{Q}{A} \qquad [49]$$

Heat transfer, next to work transfer, is one of two types of energy interactions that are accounted for in the first law of thermodynamics. For a closed system you get the relation given in Equation [50]. Here Q (rate of heat transfer) and W denote the sum of all the heat and work transfer interactions experienced by the closed system.

$$Q = \frac{dE}{dt} + W \qquad [50]$$

Different modes of heat transfer exist, each governed by its own physical principle:

- Radiation heat transfer
- Conductive heat transfer
- Convective heat transfer

Hereafter, we will discuss the various modes in some more detail. To simplify matters, however, we will neglect variations in time.

Radiation
Thermal radiation is the process by which the surface of an object radiates its thermal energy in the form of electromagnetic waves. The higher the temperature of the object the more energy is radiated. The amount of heat radiated by a body is given by:

$$\dot{Q}_{emitted} = \varepsilon \cdot \sigma \cdot A \cdot T^4 = q_{emitted} \cdot A \qquad [51]$$

- ε = emissivity of a body (0< ε < 1). It depends on the material, wavelength and temperature of a body. An object is called a black body radiator if ε = 1 (for all wavelengths)
- σ = Stefan Boltzmann constant (σ = 5.67 x 10^{-8} W/m^2-K^4)
- A = radiating area of body
- T = body temperature
- q = heat flux

Conduction
Conduction takes place in stationary mediums such as solids, liquids, and gases due to a temperature gradient. Heat flows through thermally conductive materials by a process generally known as 'gradient transport'. It depends on three quantities: the conductivity of the material, the cross-sectional area of the material, and the spatial gradient of temperature. Conductive heat transfer is mathematically best described by Fourier’s law, which quantifies the conduction process as a rate equation in three dimensions. To simplify matters, the discussion hereafter will be limited to uni-directional conduction, i.e. conduction in one direction only. For stationary conditions, you get the relation given in Equation [52], where k is the coefficient of thermal conductivity [W/mK], A the cross sectional area [m^2], and l the length of conductive path [m]. Of these, the thermal conductivity coefficient depends on the material selected.

$$\dot{Q} = \frac{k \cdot A}{l} \cdot \Delta T = \frac{k \cdot A}{l} \cdot (T_2 - T_1) \qquad [52]$$

From this relationship, we learn that the larger the conductivity, and/or the temperature gradient the faster the heat flow. It is mentioned that heat transfer takes place in the direction of decreasing temperature; a negative sign in the answer than indicates that the heat flow is from location 1 to 2 instead of the other way around. The thermal conductivity is a measure of how efficiently a solid conducts heat or how fast heat travels through the material.

Convection

“Convection” is energy transfer between a solid surface and an adjacent moving gas or liquid, i.e. the transport of heat by a moving fluid (liquid or gas). It basically results from a combination of diffusion or molecular motion within the fluid and the bulk or macroscopic motion of the fluid. For convection you get the relation given in Equation [53], where h_c is the coefficient of convective heat transfer [W/m^2K], A the surface area in contact with the flow [m^2], and ΔT the temperature difference [K] between body and the surrounding medium.

$$\dot{Q} = h_c \cdot \Delta T \cdot A \qquad [53]$$

The spacecraft thermal environment

Since the spacecraft is in space and space essentially is empty, we do not have to worry about heat flowing to the spacecraft either by convection nor conduction. Some convection/conduction may occur at low altitudes (consider vehicles entering a planetary atmosphere), but this is considered out of scope. Note that inside the spacecraft conduction and convection may take place next to radiation heat transfer. For now though we will only consider the interaction of the spacecraft with the space environment. This interaction is shown in Figure 35 for a spacecraft nearby Earth.

From Figure 35 we learn that there are essentially 3 flows of heat to the spacecraft and 1 flow away from it. The incoming heat flow consists of (direct) solar radiation from the Sun (except when it experiences an eclipse), solar energy reflected from the surface in the direction of the spacecraft (again except when it is experiencing an eclipse), also referred to as albedo radiation. In addition, because also the planet itself has some temperature, the spacecraft may also receive heat from the planet (planetary of IR radiation). Not shown in the figure is that the spacecraft itself serves as a source of heat, thereby causing its temperature to rise.

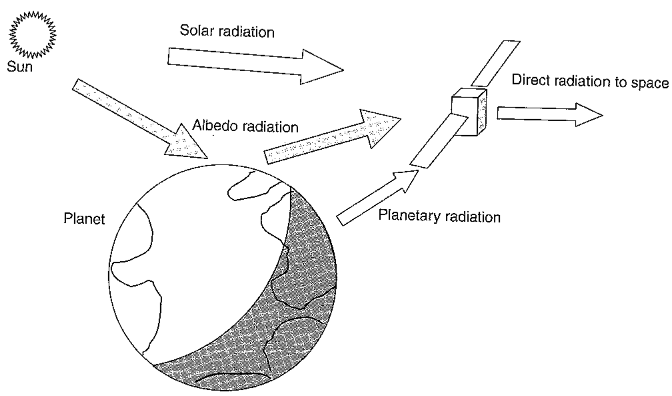


Figure 35: Typical spacecraft thermal environment [SSE]

Because space itself is a very cold environment (4 K) some heat will also be radiated away from the spacecraft to the cold environment.

Outer space is colder than the North Pole in December! But it can also be hotter than an erupting Volcano. On Earth, air helps even out the temperature. But there is no air in space. You can fry on one side while freezing on the other.

Hereafter, we will discuss the various heat flows in some more detail, before discussing how they affect spacecraft temperature

Solar intensity

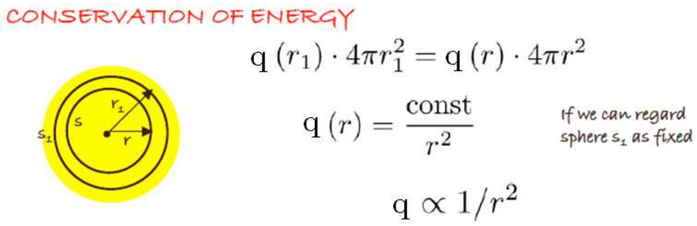
One thing we all know is that the Sun is extremely hot with an effective radiating temperature of 5870 K, where the effective radiating temperature of a celestial body is determined using relation [51] under the assumption that $\varepsilon = 1$. It follows:

$$q_{emitted} = \sigma \cdot T^4 \qquad [54]$$

and hence:

$$T_e = \sqrt[4]{\frac{q_{emitted}}{\sigma}} \qquad [55]$$

From measurements we know that the flux emitted by the Sun at its surface is about 63.28 MW/m^2. This gives an effective temperature of our Sun of about 5780 K. Using information on the radius of the Sun, we are able to determine that the total power emitted by the Sun (P) is 3.856 x 10^{26} W. From conservation of energy, under the condition that energy is emitted equally in all direction, see the sketch, it follows that the flux (also referred to as intensity) reduces with the square of the distance to the center of the radiating object (inverse square law[10]).



Hence, it follows for the solar intensity:

$$J_s = \frac{P}{\left(4 \cdot \pi \cdot d^2\right)} \qquad [56]$$

So solar intensity at some location in space can be determined once the distance of this location to the Sun is known. Solar intensity versus average Sun distance is plotted in Figure 36. It clearly shows that the further away from the Sun, the lower the solar intensity received (and the colder space will be).

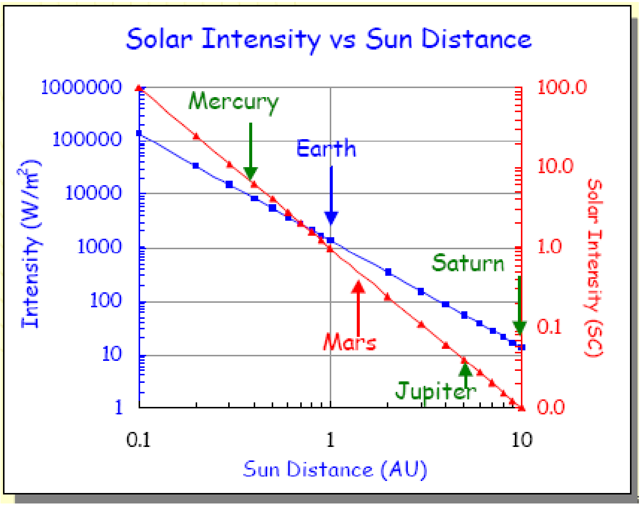


Figure 36: Variation in solar intensity with (average) Sun distance

10 From Wikipedia: In physics, an inverse-square law is any physical law stating that a specified physical quantity or strength is inversely proportional to the square of the distance from the source of that physical quantity.

To determine the power emitted by the Sun, we use Eq.[56]. Given an effective temperature of the Sun of 5780 K, it follows that the Sun emits 63.28 MW/m^2. From [Wiki], we find that the Sun has a diameter of about 1,392,000 kilometers (about 109 Earth diameters). Multiplying the calculated flux with the Sun's surface area, we find for the total power emitted by the Sun P = 3.852 x 10^{26} W, which agrees well with the value for the solar power P introduced earlier.

<u>Albedo</u>

The albedo of an object is a measure of how strongly it reflects light from light sources such as the Sun. It is of importance when considering the heat inputs to a spacecraft. The albedo flux received by a spacecraft at some distance away from the reflecting planet is given by:

$$J_a = a \cdot J_s \cdot F \qquad [57]$$

Here:

- a = albedo factor ($0 \leq a \leq 1$)
- F = visibility factor ($0 \leq F \leq \sim 1$)

Different definitions of albedo factor exist. Following the approach taken in [SSE], here albedo factor or shortly albedo is defined as the fraction of the incident solar radiation returned from a planet. This albedo factor is also referred to as Bond albedo and commonly has a value between 0 and 1. Next to Bond albedo there is also to the geometric albedo (measuring brightness when illumination comes from directly behind the observer). Bond and geometric albedo values can differ significantly, which is a common source of confusion. Typical average value of the (Bond) albedo of Earth is 0.3 ± 0.14. For an orbiting spacecraft, the albedo can vary between 0.05 (open ocean) and 0.6 (high cloud/icecap). Typical average values of the (Bond) albedo of some of the other planets in our solar system are given in the figure on the right.

Albedo of other planets

(ECSS-E-ST-10-04C)

Table F-1: Reference values for average planetary albedo

Planet	Average Albedo	Albedo Range
Mercury	0,106	0,09 – 0,45
Venus	0,65	
Mars	0,15	
Jupiter	0,52	
Saturn	0,47	
Uranus	0,51	
Neptune	0,41	
Titan	0,22	

For Pluto and the Moon:

Pluto 0.16

Moon 0.07

Figure 37: Reference values for average Albedo of some other planets in the solar system [ECSS]

F is a visibility factor, which like the albedo factor, has a value in between 0 (night side) and 1 (full sunlight, close to Earth surface). It basically depends on the distance of the spacecraft to the planet and the angle between the local vertical and the Sun's rays. For design issues, one is mainly interested in worst case hot and cold conditions. In that case the angle effect is neglected and we obtain:

$$\text{In shadow of planet}: F = 0$$

$$\text{Sun lit side of planet}: F = \left(\frac{R_{planet}}{R_{orbit}}\right)^2 \qquad [58]$$

Where R_{planet} is radius of planet and R_{orbit} is orbital radius of spacecraft about the planet. Some further details can be obtained from [SSE].

<u>Planet flux</u>

Planet flux is infrared (IR) energy radiated by a planet. It depends on the effective radiating temperature of the planet according to:

$$J_{IR} = \sigma \cdot T_{IR}^{\ 4} \qquad [59]$$

The effective radiating temperature of some planets is given in Table 24. For Earth with an effective temperature of 255 K, planet flux (at the planet surface) is equivalent to 240 W/m^2.

Table 24: Effective radiating temperature of the planets of the solar system [ECSS]

Celestial body	Effective radiating temperature [K]
Sun	5780
Mercury	600-617
Venus	227
Earth	255
Moon	120-380
Mars	210.1
Jupiter	109.5
Saturn	81.1
Uranus	58.2
Neptune	46.6
Pluto	43

For the Moon values are given for the sun lit side and for the night side independently.

Like for solar intensity, IR intensity is inversely proportional to the square of the distance to Earth center. This makes that in deep space planet IR flux is (almost) negligible.

Heat fluxes as calculated for a real mission

Figure 38 shows how the heat fluxes may vary for a real space mission, in this case a mission to the Moon. When in full sunlight close to Earth, the S/C may experience a heat flux of about 1700 W/m^2 whereas in Earth shadow, it is just about 70 W/m^2. In Moon orbit the numbers are again quite different.

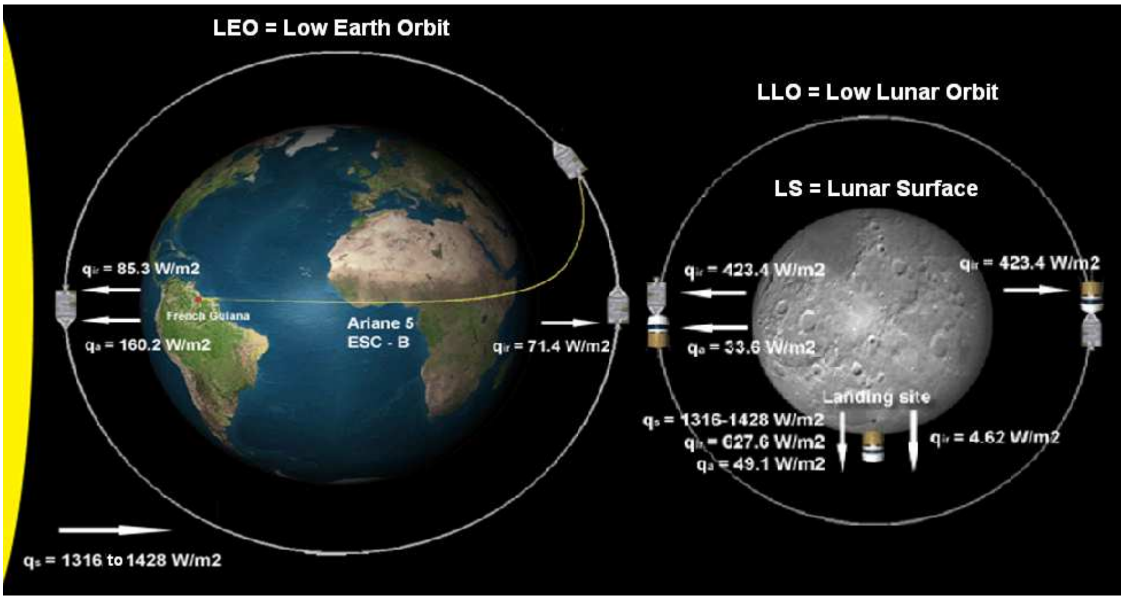


Figure 38: Overview of heat fluxes as experienced for the M3 mission

Students should prepare for calculation the heat fluxes for different spacecraft under different conditions.

Temperature change

The heat flowing to the spacecraft will cause a rise in temperature (ΔT) of the spacecraft. The temperature rise can be calculated using Eq.[60], where $\dot{Q}$ is heat flow rate (in Watt), Δt is time period considered, M is mass of body and C is heat capacity of the body.

$$\Delta T = \frac{\dot{Q} \cdot \Delta t}{M \cdot C} \qquad [60]$$

Example: spacecraft heating

Consider a spacecraft at a distance of 1 AU from the Sun. This spacecraft has a mass of 500 kg and a heat capacity of 1000 J/kg-K. The area of the spacecraft receiving the solar flux is 2 m^2. Determine for this spacecraft the rise in temperature in case the spacecraft is exposed to the Sun for a duration of 1 hour (note other heat flows may be neglected).

Solution:

Heat flow to the spacecraft is 2.8 kJ/s. This gives a temperature rise of the spacecraft:

$$\Delta T = \frac{2.8\ \text{[kJ/s]} \cdot 3600\ \text{[s]}}{\left(500\ \text{[kg]} \cdot 1000\ \text{[J/kg-K]}\right)} = 20.2\ \text{[K]}$$

The problem gets even worse, when considering that onboard of the S/C we have a number of instruments and equipment that dissipate energy and hence generate heat. Typical electrical efficiencies of most equipment are in the range of 15-30%. The remainder is lost in terms of heat.

Still, most preliminary thermal analysis are performed neglecting any changes in time. Changes in time will only be considered when detailing the design. More important is to consider equilibrium conditions, wherein the heat flowing to the spacecraft balances with the heat flowing away from the spacecraft.

Heat balance

The interplay of energy exchange between two bodies is characterized by Eq.[61], where α presents spectral absorption factor, ρ spectral reflection factor and τ spectral transmission factor. These parameters all depend on the material used, the wavelength considered and the temperature. The spectral absorption factor of some material at some wavelength is equal to its emissivity; this relation is known as Kirchhoff's law of thermal radiation. Consequently for a black body also α = 1.

$$\alpha + \tau + \rho = 1 \qquad [61]$$

In a practical situation a spacecraft receives thermal energy from the Sun. When in the vicinity of a planet, the spacecraft may also receive reflected sun light and IR radiation from the planet itself. However, the spacecraft also loses considerable energy by emitting infrared heat, as shown in [SSE, Figure 11.1]. It will experience thermal equilibrium once the flow of energy to the spacecraft (including the heat internally generated on board) balances with the heat flow from the spacecraft to outer space. It follows:

$$\dot{Q}_{in} = \dot{Q}_{out} = \dot{Q}_{absorbed} + \sum P_{dissipated} = \dot{Q}_{emitted} \qquad [62]$$

The amount of heat absorbed by the spacecraft is given by the following equations:

$$\dot{Q}_{absorbed} = \alpha_s \cdot J_s \cdot A_i + \alpha_s \cdot J_a \cdot A_i + \alpha_{IR} \cdot J_{IR} \cdot A_i \qquad [63]$$

$$\dot{Q}_{absorbed} = \alpha_s \cdot J_s \cdot A_i + \alpha_s \cdot J_a \cdot A_i + \varepsilon_{IR} \cdot J_{IR} \cdot A_i \quad \left(\alpha_{IR} = \varepsilon_{IR}\right) \qquad [64]$$

Where:

- ε is emissivity

- α = absorptivity
- J = intensity
- Subscripts s, a and IR refer to solar radiation, albedo and IR radiation, respectively
- A_i = projected area receiving respectively solar, albedo and planetary radiation. Respective areas vary depending on the vehicle orientation to the oncoming radiation. Subscript "I" refers to incoming radiation.

Note that in relation [63] different absorption factors apply for solar radiation and Earth IR. This is because the radiation from the Sun and the Earth IR encompass different frequencies. In relation [64], the absorptivity in the IR range is set equal to the emissivity in the IR range. This is because according Kirchhoff, absorptivity and emissivity of a material at identical temperature are essentially equal. Now in spacecraft thermal engineering, we like to get rid of the subscripts "IR" and "s" and hence the thermal jargon uses:

- o α for absorption in solar spectrum
- o ε for absorption <u>and</u> emission in infrared spectrum

Another important heat source is the heat internally generated in the spacecraft by the various devices , like batteries, power converters, etc., represented by ($\sum P_{dissipated}$). The summation sign here signifies that heat may be dissipated in multiple devices.

The heat flowing away from the spacecraft can be determined using:

$$\dot{Q}_{emitted} = \varepsilon_{IR} \cdot \sigma \cdot A_e \cdot T^4 \qquad [65]$$

Here the subscript "e" denotes the emitting surface, which may be quite different from the surface receiving radiation, see for instance the example below.

Example: Equilibrium temperature flat plate
A flat plate at 1 AU from the Sun is illuminated by the Sun with the Sun's rays normally incident to the flat plate. In equilibrium, it follows that the energy absorbed must equal the energy emitted (see also [SSE section 11.3]):

$$\alpha_s \cdot A_a \cdot J_s = \varepsilon_{IR} \cdot \sigma \cdot A_e \cdot T^4 \quad \rightarrow \quad T^4 = \frac{\alpha_s \cdot A_a \cdot J_s}{\varepsilon_{IR} \cdot \sigma \cdot A_e}$$

- *A_a = area receiving/absorbing sunlight, A_e is radiating area = 2 A_a*
- *Black on both sides: ($\alpha_s/\varepsilon_{IR}$ = 0.9/0.85)*
- *J_s = 1371 W/m^2*

Filling in numbers provides an equilibrium temperature for the plate of 63^{o}C.

The thermal control subsystem
The thermal control subsystem consists of all the hardware (and software) needed to maintain the temperatures of all spacecraft bus components, and those payload suites without their own thermal control provisions, within acceptable limits during ground test, launch and on orbit operations.

For instance, the equilibrium temperature for the flat plate type calculated in the foregoing example may be lowered by selecting a larger area for radiating heat away. Another option might be to select materials with a different ratio between solar absorptance and IR emissivity, see for instance the next table.

Table 25: Solar absorptance and hemispherical emissivity of typical space materials

Material	Solar absorptance (BOL)	Hemispherical emissivity
Black paint	0.96	0.75 to 0.88
Aluminized teflon foil 5 mil	0.14	0.78
Sivered teflon foil 5 mil	0.09	0.80
Aluminized kapton foil 2 mil	0.42	0.72
Aluminized kapton foil metal side	0.12	0.05
White paint	0.17 – 0.38	0.82
OSR without glue gaps	0.09	0.76
Solar cell Si	0.75	0.82
Solar cell GaAs	0.91	0.81
CFRP	0.92	0.82

From the table it should be clear that solar absorptance can be quite different from IR emissivity. This is because these values hold for a different wavelength band. When at the same wavelength or band of wave lengths, like when taking both absorptance and emissivity in the IR range, than absorptance and emissivity will have equal value (see Kirchhoff's law).

Figure 39 shows a real life satellite covered with thermal control blankets; most part of the satellite is covered with brown/gold shiny colored aluminized Kapton and some part is covered with black painted Kapton. Some mirror-like surfaces are also visible. These are radiator areas covered with Second Surface Mirror tape or OSR mirrors and are especially devoted to radiate heat away from the spacecraft. From the figure it must be clear that thermal control determines more than 95 % of the spacecraft exterior.

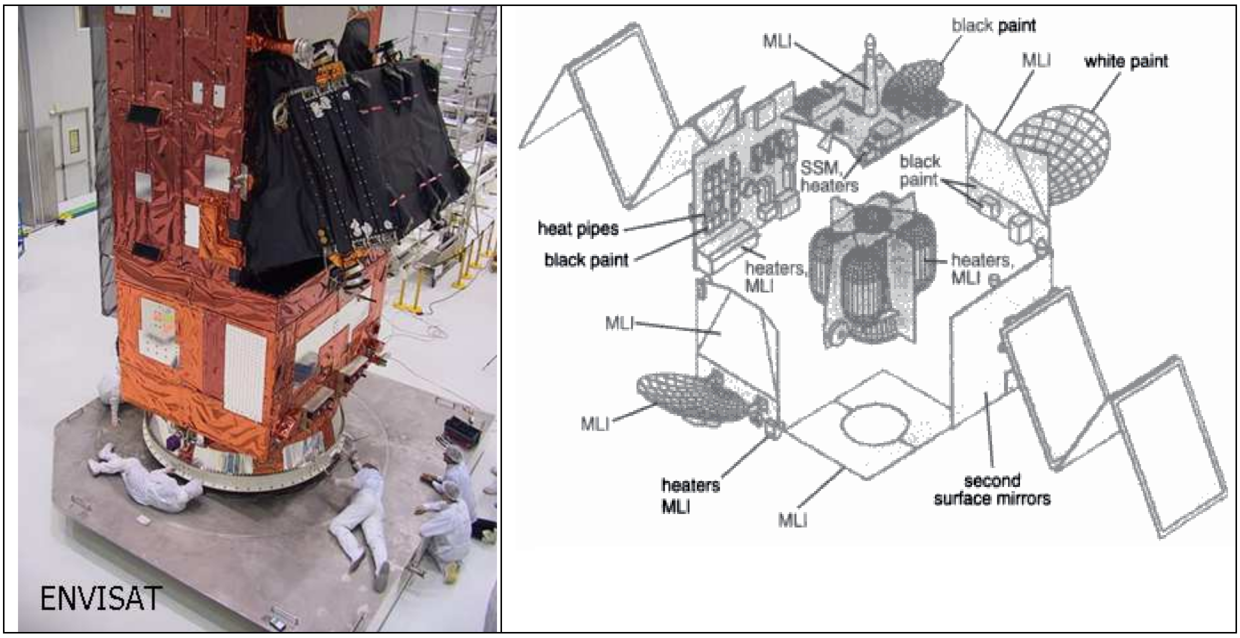


Figure 39: Thermal design of a satellite

Components of the thermal control subsystem include:

- Paints that modify the emissivity and/or absorptance of a surface
- Multi-layer insulation (MLI), see Figure 40; Multiple layers of thin foils, see figure, that allow to achieve low emissivity in the range of 0.003-.03.
- Heaters: To provide local heating for instance to prevent propellants from freezing or a drastic reduction in battery capacity

- Heat pipes: To transport heat from surfaces of high temperature to surfaces with lower temperature
- Second Surface Mirrors or Optical Surface Reflectors (OSR), see Table 25.

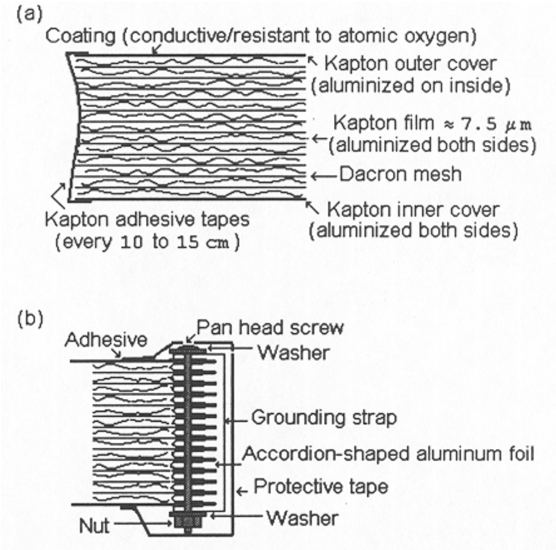


Figure 40: MLI construction: a) typical lay-up and b) electrical grounding (courtesy Dutchspace)

- Radiators: Essentially consisting of black painted pipes facing open space through which a hot fluid runs. The black surface will radiate heat into space and the fluid will be cooled. To enlarge the radiating area per pipe, fins may be attached to the pipes. Radiators may be part of a pumped loop or part of a heat pipe system (condenser section).

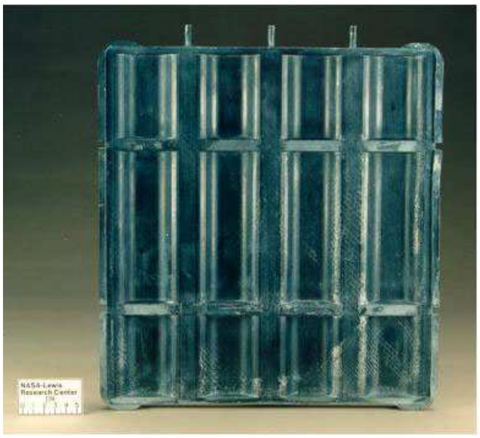


Figure 41: Radiator panel (courtesy NASA)

Problems
A number of problems for exercising upon are available via Blackboard (Maple TA), whereas a few are also contained in a separate workbook available from the TU-Delft online print shop. Of this workbook also an electronic copy is available on blackboard.

4.3 Electrical Power Generation

The electrical power subsystem provides the spacecraft with the necessary electrical power over the duration of the mission. Key requirements for the Electric Power System (EPS) are hence power level and duration of power delivery.

Why electrical power
Most spacecraft need some means of electrical power provision to power the payloads, like:

- Radar transmitters and receivers
- Radio transmitters & receivers
- Visual (VIS), Infra-Red (IR), Ultra-Violet (UV) and roentgen (X-ray) cameras

and on-board equipment, like:

- Computer, data recorder
- Telemetry and -command radio transmitters & receivers
- Valves, pressure regulators
- Heaters
- Sensors (gyros, accelerometers, sun/star/Earth sensors, etc.), actuators (reaction wheels, momentum wheels, etc.)
- Electrical motors

Electrical Power subsystem requirements
An important requirement for the electrical power subsystem is how much power, i.e. energy per unit of time, is to be provided. To estimate the amount of power to be generated on board of the spacecraft, some data is needed. Unfortunately in the early design stages very little data is available and one generally uses historical data, Table 1. Table 26 provides an overview of installed power per mission type [ESA bulletin 87].

Table 26: Overview of installed power per mission type

Mission	Orbit, Attitude	Installed Power (W) min. max
Science: – Astronomy – Deep space	 HEO, Fixed point Sun pointing (mostly) 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

Historical data shows that electrical power usage has increased with about a factor 10 every 10 years. Even so, power levels are still quite modest (compare for instance with power usage at home; a simple light bulb might already require 100 W).

The foregoing table can now be used to come up with a first very global estimate. An alternative approach which allows for a more accurate estimate can be obtained in case the amount of power needed by the payload is known. It is actually the payload for which we have to design the spacecraft, so hence it should also determine for a large extent the power needs. The approach envisaged is based on collecting power data for comparable spacecraft and determining the percentage contribution of the payload and the various subsystems. For instance, Figure 42 shows the average power distribution as determined based on a large number of GEO telecommunications satellites.

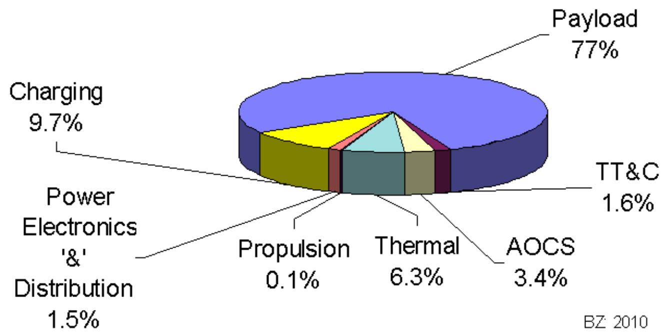


Figure 42: Power distribution of GEO telecommunications satellites (average percentages)

So given some payload power, total power and hence also the power required by the various subsystems can be determined, see also an earlier section on budgeting.

A third method can be used once we know more or the spacecraft and the equipment used on board (note that we need to be somewhat further in the design chain). It is based on an inventory of all the equipment requiring electrical power and a determination of their power usage. The equipment and their power usage are generally collected in a power budget, i.e. a list of all electrical apparatus on board (the loads) with per operating mode the electrical properties and the duty cycle that allows us to keep track of things. A typical such power budget is shown in Table 27.

Table 27: Typical (early) power budget for LEO observatory spacecraft

	Maneuvering	Operations
Observation instruments	-	150.0
Spacecraft subsystems		
Thermal control	52.6	52.6
Orbit and attitude control	75.8	62.4
On-board computer	20.0	45.0
Communication	92.5	92.5
Subtotal	240.9	402.5
Margin (10%)	24.1	40.5
Electrical power subsystem	262.0	262.0
Total	527.0	705.0

The table gives power values for two different working modes, distributed over the observation instruments and the various spacecraft subsystems. As data tend to change over the project, a margin of 10% has been taken into account thereby allowing for some growth. However, depending on how certain we are of the data margins of up to 50% can be applied. The lack of detail is typical for an early design since at that stage in the design details on the various equipment is still lacking and hence budgeting is performed based on historical data (obtained for comparable spacecraft).

Not included in the table is information on the time period during which the spacecraft is performing maneuvers or (normal) operations. This is important to know, as energy usage follows from the product of power of some item and the duration over which this item is active. An upper limit here is given by the life (nowadays in the range of a few months up to 15 year) of the spacecraft. Of course we also strive to maximize the time for normal operations, for instance 95% of the total time available. Also not shown in the table is that some equipment might work intermittently instead of continuously. This of course also influences the (average) power needed. For instance, a radar device working in a pulsed mode requires much more power when transmitting then when listening (receiving) the

reflected signal. This shows that we should make a distinction between average power and peak power to be delivered by the system.
The budget example shown is a simple budget showing little detail. Later in the design budgets will become more detailed and more accurate. This will be left for you to explore later in the study for instance in a design project.

Once we know the power level, the duration over which this power should be delivered and the total energy required, we can start with the design of the spacecraft electrical power subsystem (EPS), i.e. the set of hard- and software that provides the spacecraft with the necessary electrical power. Here design means that we strive to provide the required electrical power, while limiting the resources needed. Currently, the following resources need to be considered:

- It makes up 20 - 40% of spacecraft dry mass, see appendix D
- It is visibly present (solar panels), thereby limiting the space available for other equipment
- It accounts for 10 - 15% of all serious on-orbit spacecraft (S/C) bus failures, see for more detailed information appendix D

The ideal EPS of course delivers power at almost no drawbacks, meaning low cost, minimum mass, no risk, etc.

Types of EPS
Different types of EPS for spacecraft exist. We mention the use of photovoltaic cells to convert sunlight into electric power, the use of batteries, fuel cells and even nuclear generators. An overview of typical EPS used is given in Table 28.

Table 28: Overview of S/C power generation systems

Envisat	Electrical power generation system consists of a solar array with 8 batteries providing eclipse power. The Solar Array consists of 14 panels one by five meters, which generate a total of 6.6 kilowatts of electrical power end of life (5 yr in space)
GRACE	Electrical energy is generated using a solar array of Globalstar silicon cells, placed on the top and side exterior surface of the satellite and providing 160 W. Excess energy is stored in a battery of NiH2 common pressure vessel cells for use during eclipse periods and for providing peak power. The power bus delivered unregulated power to all users at the respective user interface.
Meteosat	Power is provided by body mounted solar array providing about 240 W over a mission life of 5 years. As the S/C rotates about cylinder axis only part of array is effective at any one time and hence the total array area is much larger than for a planar panel always directed towards the Sun.
Delfi C^3	Power is generated by photovoltaic array of GaAs cells distributed over 4 wings providing an average power of 2-3 W in full sun light. Total area is 0.08 m^2. Powerless during eclipse periods.
Mars Express	Power is provided for by two photovoltaic solar wings with a total area of 11.42 m^2 providing 650 W at maximum distance from the Sun + batteries for eclipse periods
Venus Express	Solar generator provides 650 W at max. distance from the Sun + batteries for eclipse periods; Array area is 5.7 m^2
Voyager	Power is taken from 3 radio-isotope thermo-electric generators (RTGs) that provide 470 W (BOL) over a 10 year life
Apollo Lunar Lander	4-5 28-32 V, 415 Ah Silver-Zinc (AgZn) batteries with a mass of 61.2 kg each.
Apollo Moon buggy	2 AgZn batteries with a total capacity of 121 Ah each for a mission duration of about 4 hrs
Ariane 5 rocket	6 AgZn batteries of which 4 provide 40 Ah each for a mission duration of 65 minutes
Space Shuttle	3 Alkaline fuel cells providing 12 kW each for a duration of maximum 2500 hr.
Apollo SM	Power is taken from fuel cells providing 1.5 kW for up to 12 days

An important distinction we can make is in systems for which the power output depends on the distance to the Sun and its visibility, like for photovoltaic systems, and those that operate independently, like batteries, fuel cells and RTGs. The former essentially can operate for indefinite time since they use an <u>external</u> energy source, i.e. the energy source is not carried on board; whereas the latter are equipped with an internal energy source that provides for the required energy. The latter usually have limited operation time as otherwise they simply become too heavy. In [SSE, figure 10.1] a figure is presented illustrating how for space missions power level and mission time are related to the selection of some type of power source. An example illustrating how the energy requirement determines the mass of an internal energy source is provided later in this text.

PhotoVoltaic (PV) systems

Most long—life spacecraft use photovoltaic power since solar energy is available for free. In such systems solar energy is converted into electric energy using the photovoltaic effect. Characteristic system data of some spacecraft photovoltaic systems are given in Table 29.

Table 29: Some important characteristics of photovoltaic systems [Sarsfield]

Mission	Spacecraft	Power System Mass (kg)	Solar Array Material	Solar Array Area (m2)	Solar Array Efficiency (W/m2)	Solar Array Mount	Beginning of Life Power (W)	Average Power (W)	End of Life Power (W)
Clementine		32.0	GaAs/Ge	2.30	156.5	Deployed	360	n/a	n/a
Discovery	NEAR	64.4	GaAs/Ge	8.90	211.2	Deployed	1,880	1,600	1,500
	Mars Pathfinder	23.0	GaAs	4.00	62.5	Body Mount	250	190	—
Explorer	SMEX-SWAS	59.0	GaAs	2.50	240.0	Deployed	600	270	525
	SMEX-TRACE	32.0	GaAs	1.30	203.8	Deployed	265	90	200
	MIDEX-MAP	41.0	GaAs/Ge	3.10	193.5	Deployed	600	320	400
New Millennium	Deep Space 1	108.0	GaInp2/ GaAs/Ge	9.00	288.9	Deployed	2,600	n/a	n/a
	Earth Observer 1	36.0	GaAs	4.50	140.0	Deployed	630	250	600
SSTI	Lewis	n/a	GaAs/Ge	—	—	Deployed	370	n/a	n/a
	Clark	11.0	GaAs	3.60	111.1	Deployed	400	165	350
Surveyor	Mars Global Surveyor	73.6	GaAs + Si	12.00	56.4	Deployed	677	645	624
	Mars Surveyor '98—Lander	40.5	GaAs	3.70	181.9	Deployed	673	n/a	307
	Mars Surveyor '98—Orbiter	46.3	GeAs	7.40	202.7	Deployed	1,500	n/a	515
Baseline	RADCAL	19.0	Si	0.81	55.6	Body Mount	45	25	18

Table 29: Continued

Mission	Spacecraft	System specific power (W/kg)	Battery Type
Clementine		11.3	NiH2
Discovery	NEAR	29.2	Super NiCd
	Mars Pathfinder	10.9	AgZn
Explorer	SMEX-SWAS	10.2	Super NiCd
	SMEX-TRACE	8.3	Super NiCd
	MIDEX-MAP	14.6	NiH2
New Millennium	Deep Space 1	24.1	NiH2
	Earth Observer 1	17.5	Super NiCd
SSTI	Lewis	n/a	NiH2
	Clark	36.4	NiH2
Surveyor	Mars Global Surveyor	9.2	NiH2
	Mars Surveyor '98—Lander	16.6	NiH2
	Mars Surveyor '98—Orbiter	32.4	NiH2
Baseline	RADCAL	2.4	NiCd

From the table we learn that:

- Array delivers between 50-300 W/m^2. On average power density (power per unit of array area) is 100 W/m^2
- Specific power of photo-voltaic based EPS is in range 1.0 W/kg - 25 W/kg based on End Of Life (EOL) power (10 W/kg is a reasonable value to produce a first mass estimate based on power to be delivered).
- Power output reduces in time (from a few percent up to 20%). This is due to ageing of the solar cells.
- Two types of systems: body mounted and deployable systems
- Most PV systems use a battery system as a secondary power source. This system is to provide for power during eclipse periods, i.e. periods wherein the view of the Sun is obscured by Earth, Moon or some other planetary body, see Figure 43.

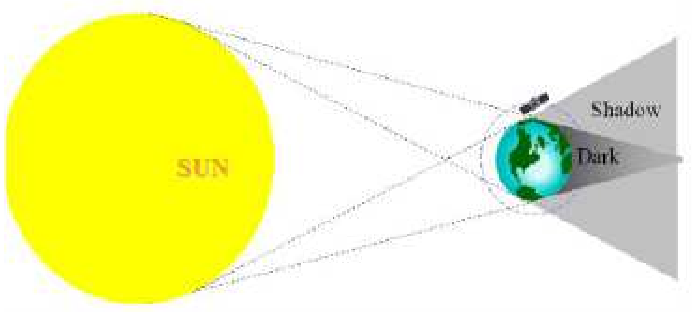


Figure 43: S/C experiencing solar eclipse during its rotation about some planet

PV power essentials
The main constituents of a photovoltaic based power system are:

- Primary power source, i.e. the solar array
- Secondary power source, like a battery, that provides for power during eclipse periods or when the power peaks
- Power distribution lines/cables that transport the source to the units requiring power (the S/C loads)
- Power controls that ensure the proper amounts of power are delivered and protect the various units against power spikes, etc.

A schematic overview of a PV system is given in Figure 44, a typical power subsystem mass breakdown is given in Figure 45.

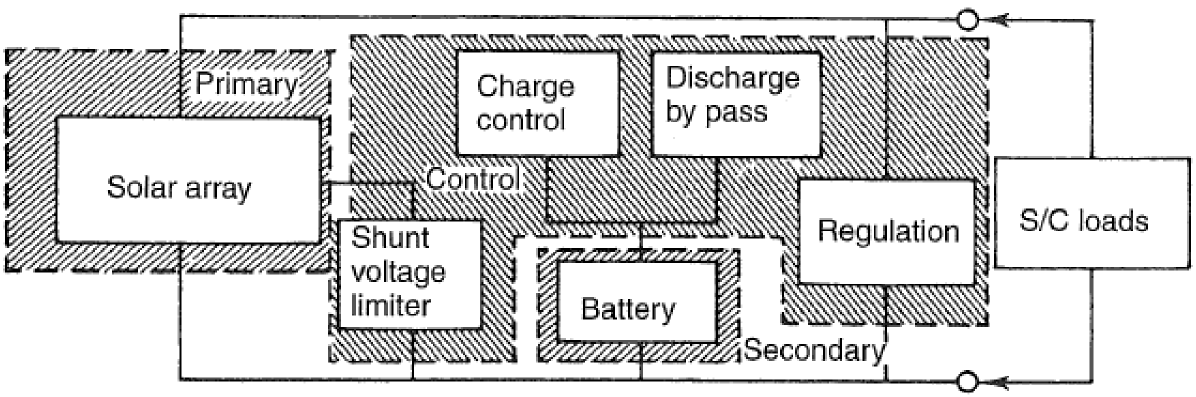


Figure 44: Schematic of photo-voltaic based EPS

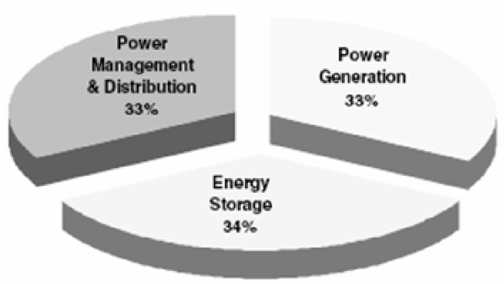


Figure 45: Typical power subsystem mass breakdown

Here the mass of the EPS is given as:

$$M_{EPS} = M_{PV} = M_{SA} + M_{BAT} + M_{PMD} = P_{total} \ / \ \left(P_{sp}\right)_{system} \qquad [66]$$

System specific power @ 1 AU is in range 2-40 W_e/kg, see Table 29 for more info.

In the next few sections, we will discuss in some more detail the sizing of a PV system based on the power that is to be delivered by the system.

Required power

The power (P) needed from the solar array (SA) can be determined using:

$$P_{SA}\ t_d = \frac{P_d\ t_d}{\eta_d} + \frac{P_e\ t_e}{\eta_e} \qquad [67]$$

Here t is time with the subscript d referring to the daytime period and e to the duration of the eclipse period, P_d is the power needed during day time and P_e during eclipse and η gives the path efficiency of the power system during day time and night time respectively. Note that the power requirement during daytime and night time may differ as during night time one might operate with less active equipment to conserve energy. Day time path efficiency typically is 80% and represents the efficiency of the control and distribution system. When considering night time, path efficiency decreases as in that case also the efficiency of the storage system (battery) and its management systems need to be taken into account. A typical value for preliminary design considerations is an efficiency of 60%.

Example: Sizing for solar array power
Consider a spacecraft in a highly elliptical orbit about Mars. The spacecraft's orbit has an apocenter of 10,107 km and a pericenter of 298 km giving an orbital period of 6.7 hours (consider verifying this value using the material taught in AE1110-II). Eclipses experienced by the spacecraft will last maximum 75 minutes (or 1.25 hr) per orbit about the Red Planet. Given a total power required by the equipment and payloads of 450 W, an efficiency of the power management and distribution system of 80% during daytime conditions and 60% during night time (=eclipse) conditions it follows for the power requirement:

- *Daytime: 450W/0.8 = 562.5 W*
- *Eclipse: 450W/0.6 = 750 W*

It follows for the total energy needed during one orbit:

$562.5\ \text{W} \cdot \left(6.7 - 1.25\right)\ \text{h} + 750\ \text{W} \cdot 1.25\ \text{h} = 4003.1\ \text{Wh}$ = *(14.4 MJ).*

Given that the array only produces power during daytime conditions, it follows for the total power to be provided by the array:

$4003Wh / \left(6.7hr - 1.25hr\right) = 4003/5.45 = \boxed{734.5\ \text{W}}$

Here the 6.7 hr – 1.25 hr represents the day time available per orbit.

Figure 46 provides some detailed info on the efficiency of PV systems.

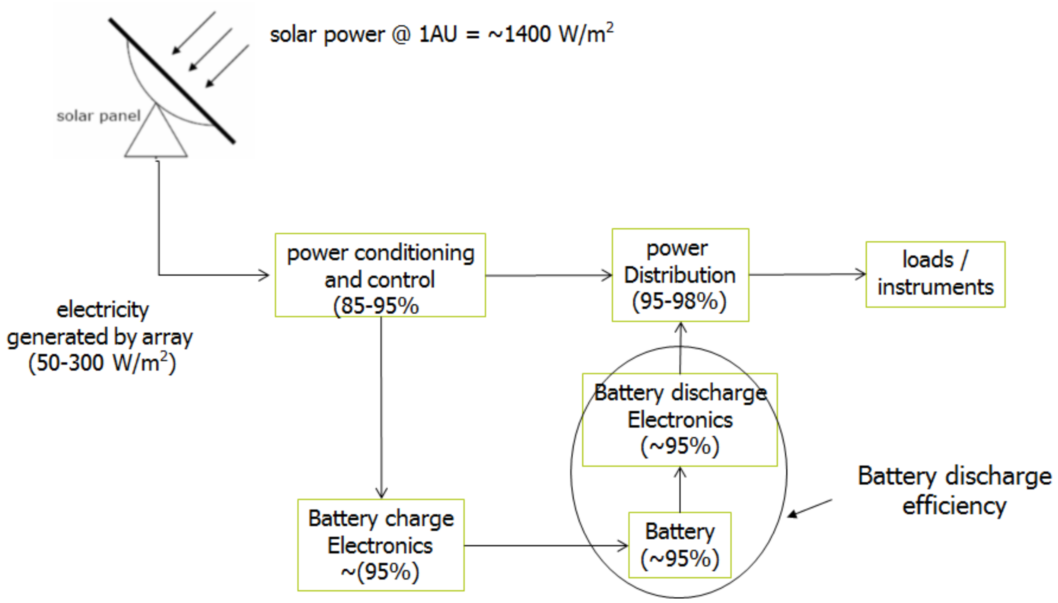


Figure 46: Example of PV system component efficiencies

Day time efficiency is given by:

$$\eta_d = \eta_{\text{power conditioning}} \cdot \eta_{\text{distribution}} \qquad [68]$$

Night time efficiency is given by:

$$\eta_e = \eta_{\text{power conditioning}} \cdot \eta_{\text{charge electronics}} \cdot \eta_{\text{battery}} \cdot \eta_{\text{discharge electronics}} \cdot \eta_{\text{distribution}} \qquad [69]$$

In addition, we define battery discharge efficiency as:

$$\eta_{BAT} = \eta_{\text{battery}} \cdot \eta_{\text{discharge electronics}} \qquad [70]$$

Array sizing

From the electrical power to be delivered by the array, the array can be sized and its mass determined. The mass and size of the solar array can be determined using:

$$M_{SA} = \frac{P_{SA}}{(P_{sp})_{SA}} \qquad [71]$$

$$A_{SA} = \frac{P_{SA}}{(P_{\delta})_{SA}} \qquad [72]$$

Here $(P_{sp})_{SA}$ is specific power of array (power/unit of solar array mass) and $(P_{\delta})_{SA}$ is power density of array (expressed in units of power per unit of array area).

Both specific power and power density depend on the array type, see Table 30.

Table 30: Space solar array types and their characteristics (values at 1 AU, normal incident radiation and BOL) [Bailey]

Technology All values valid at 1 AU	Specific power [W/kg] (BOL) @ cell efficiency	Cost [$K/W]	Area per power [m^2/kW]
High-efficiency silicon (HES) rigid panel	58.5 @ 19%	0.5–1.5	4.45
HES flexible array	114 @ 19%	1.0–2.0	5.12
Triple junction (TJ) GaAs rigid	70 @ 26.8%	0.5–1.5	3.12
TJ GaAs ultraflex	115 @ 26.8%	1.0–2.0	3.62
CIGS thin film[a]	275 @ 11%	0.1–0.3	7.37
Amorphous-Si MJ/thin film[a]	353 @ 14%	0.05–0.3	5.73

[a]Represents projected values. These arrays are unavailable commercially

From the table, it is clear that rigid panels gives lower specific power then flexible panels. Also panels equipped with Si (silicon) cells have lower specific power then panels equipped with GaAs (Gallium Arsenide) cells. We also notice that values given in the table are valid for BOL (Begin Of Life) conditions and under normal incidences. BOL values are because the performance of arrays degrades over time. Typical values for life degradation are:

$$\begin{aligned}&\text{Si array: } 4\% \text{ per year}\\&\text{GaAs array: } 1-15\% \text{ per year}\end{aligned} \quad [73]$$

Hence arrays should be designed such that at End Of Life (EOL) still sufficient power is delivered by the array.
The effect of solar incidence angle (θ), i.e. angle between the direction of the incident solar light and the normal onto the panel surface is determined by considering its effect on the power output of the array. Suppose that the solar array has a constant incidence angle, it follows:

$$\left(P_{sp}\right)_{SA,\theta} = \left(P_{sp}\right)_{SA,\theta=90} \cdot \sin\theta \quad [74]$$

Finally, we consider the change in intensity of the solar radiation arriving at the spacecraft with distance to the Sun. For this we use:

$$P_{sp,d} = \frac{P_{sp}}{d^2} \ [\mathrm{W_e/kg}] \quad [75]$$

$$P_{\delta,d} = \frac{P_{\delta}}{d^2} \ [\mathrm{W/m^2}] \quad [76]$$

With d is distance to the sun in AU. Notice that the above relations have also been introduced earlier in the course, but in a slightly modified form.

Battery sizing

Batteries are used on PV equipped spacecraft to provide for the necessary power during eclipse conditions and or for peak power. Hereafter, we will only deal with the design for eclipse conditions. Because of the recurring nature of eclipses for spacecraft, most spacecraft are equipped with rechargeable (secondary) batteries. To allow for sizing of the battery we first should know the battery power to be delivered and the battery discharge period, i.e. the duration of the battery discharge or power draw from the battery. Typically the power to be delivered by the battery is taken equal to the power needed during eclipse, see SA sizing. The total energy (usually expressed in Watt-hour, Wh) to be delivered by the battery follows from:

$$E_{BAT} = \frac{P_{BAT} \cdot t_{discharge}}{DOD \cdot \eta_{BAT}} \quad [77]$$

Here η_{BAT} is battery discharge efficiency (defined earlier) and DOD is battery Depth of Discharge. Battery discharge efficiency typically is in the range 90%, whereas DOD depends on the number of charge-discharge cycles a battery undergoes. Some typical data are provided in Table 31. Note that all data are for spacecraft in LEO. In GEO, much higher values (80%) can be attained for DOD as eclipses are much less frequent.

Table 31: Sample spacecraft battery configurations

Spacecraft	Launched	Battery Characteristics
LANDSAT-7	April 1999	2 Batteries (17 Cells/Battery, 50 Ah Ni/H_2) LEO, 5 years (less than 30,000 cycles) DOD 17%, 0 to 10°C
EOS Terra	December 1999	2 Batteries (54 cells/battery, 50 Ah NiH_2) LEO, 5 years, < 30,000 cycles DOD 30% -5 to 10°C
TDRS-H	June 2000	1 Battery (3 8-Cell Packs and 1 5-Cell Pack/Battery, 110 Ah Ni/H_2) GEO, 15 years, DOD 73% assuming 3 failed cells, 5°C
EOS PM-1 Aqua	May 2002	1 Battery (24 Cells/Battery, 160 Ah Ni/H2) LEO, 6 years , <35,000 cycles DOD 30%, 0 to 10°C
POES L ,M	September 2000 June 2002	3 Batteries per spacecraft (17 Cells/Battery, 40 Ah Ni/Cd) LEO/Polar, 2 years (Design), 3 years (Goal), DOD 0 21%, 5°C
HST	2003 Battery Change-out Servicing Mission 4	6 Batteries (22 Cells/Battery, 80 Ah Ni/H2) LEO, 5 years (less than 32,000 cycles), DOD < less 10%, -5 to 5°C

From: DL Britton and TB Miller, Battery Fundamentals and Operations, NASA Glenn Research Center, April 2000

Mass and size of the battery system can be determined using:

$$M_{BAT} = \frac{E_{BAT}}{(E_{sp})_{BAT}} \quad [78]$$

$$V_{BAT} = \frac{E_{BAT}}{(E_{\delta})_{BAT}} \quad [79]$$

Here $(E_{sp})_{BAT}$ is specific power of battery (energy/unit of battery mass) and $(E_{\delta})_{BAT}$ is power density of battery (expressed in units of power per unit of battery volume). Spacecraft batteries must have acceptable volumetric (Wh/l) and specific energy (Wh/kg) at a useable depth of discharge (DOD) and also good cycle life. Typical values for specific energy and energy density of space grade batteries can be obtained from various sources. Some typical values are shown in Table 32.

Sizing of Power management and distribution system

As a first approximation, we can use a simple rule based on the result earlier shown in Figure 45.

$$M_{PMD} = 0.33 * M_{EPS} \quad [80]$$

Here M_{EPS} stands for the total mass of the EPS. The underlying assumption being that the mass of the Power Management and Distribution (PMD) system is a fixed percentage of the total system mass.

[Saleh] provides for a slightly more detailed model that allows for distinguishing between the mass or the controls (PCU) and the distribution system (dist) separately:

$$M_{PCU} = 0.0045 * P_{BOL} \quad [81]$$

$$M_{dist} = 0.15 * M_{EPS} \quad [82]$$

Here the mass of the PCU is taken to be a fixed percentage of the power Begin Of Life, because at Begin Of Life generally a lot of excess power is produced.

Table 32: Characteristics of space-grade secondary batteries

SECONDARY BATTERIES

Sample Battery Characteristics and Performance

Technology	Use	No of Batteries & Cells	Ah Rated/actual	Operating Voltage	Specific Energy, Wh/kg	Energy Density, Wh/l	Operating Temp. Range, °C	Design life, Years	Cycle life to Date	Manufacturer
Ag-Zn	Cell	1	40/58	1.5	130	248	-20 to 25			BST
	Pathfinder Lander	1/18	40/58	27	85	190	-20 to 25	2	100	Yardney
Ni-Cd	Standard 50 Ah	1	50/62	1.25	31	111	-20 to 25	3		Gates
	Landsat	3/22	50 /60	22-36	27	61	-20 to 26	3	25K	MDAC
	TOPEX	3/22	50/60	22-36	27	61	-10 to 30	3 to 5	40K	MDAC
Super Ni-Cd	9 Ah Cell	1	9/12	1.25	31	93	-20 to 30			EPI
	50 Ah Cell	1	50/63	1.25	32	100	-20 to 30			EPI
	Sampex Battery	1 /22	9/12	28	28	72	-20 to 30	5	58K	EPI
	Image	1/ 22	21/24	28	33	71	-20 to 30	5	14K	
IPV $Ni-H_2$	IPV Cell	1	98/83	1.25	48	71	-10 to 30		10K	EPI
	Space Station	6/76	81/93	48	24	8.5	-10 to 30	6.5	11K	Boeing
	HST	6/22	80/85	28	8	4	-10 to 30	5	65K	EPI
	Landsat 7	2/17	50 / 61.7	24			-10 to 30	5	>50K	LMAC
CPV $Ni-H_2$	CPV Cell	2	16/17.5	2.50	43.4	77	-10 to 30	10		EPI
	MIDEX MAP	1/11	16/17.5	28	36	21	-10 to 30	5	50K	
	Odyssey	2/11	16/17.5	28	36	21.1	-3 to 8	10 to 14	1K	LMAC
	Mars 98	1/11	16/17.5	29	37	41	5 to10	3		LMAC
	MGS	2/16	20/23	20	35	25	5 to 10	5 Mars Yr	50K	LMAC
	EOS Terra	2/54	50/	67		21	-5 to 10	5		
	Stardust	1/11	16/17.5	28	36	21	-5 to 11	7	1135 days	LMAC
SPV $Ni-H_2$	SAR 10065	1/12	50/60	28	54.6	59.3	-10 to 30	10		JCI/EPI
	Clementine	1/22	15/18	28	54.8	78	-10 to 30	1	200 cycles	JCI/NRL
	Iridium	1/22	60/70	28	53.4	67.7	-20 to 30	3 - 5	50K	JCI/ EPI
Li-Ion	Cell	1	8.6/10	4.0	133	321	-20 to 30			Yardney
	MER-Rover	2/8	16-20	28	90	250	-20 to 30	3	n/a	Yardney

Ag-Zn=Silver Zinc, Ni-Cd=Nickel Cadmium, IPV=Individual Pressure Vessel, CPV=Common Pressure Vessel, SPV=Single Pressure Vessel, Ni-H2=Nickel Hydrogen, Li-ion=Lithium Ion

From: R Surampudi, R Bugga, MC Smart, SR Narayanan HA Frank and G Halpert, Overview of Energy Storage Technologies for Space Applications, Jet Propulsion Laboratory, Pasadena, CA 91109

<u>Other systems: Fuel cell systems and RTGs</u>

For some space vehicles photovoltaic systems are not a good solution. Alternative systems are for instance RTGs (some kind of nuclear reactor based on natural decay of radioisotope materials) or fuel cell based systems, see Figure 47. These systems have the disadvantage that the energy needed needs to be carried on board, but have the advantage that they can also work when in eclipse and hence do not need a secondary power source as required by a solar array system. A more extensive description of fuel cell systems and RTGs can be obtained from [SSE].

Figure 47: Apollo fuel cell powerplant (l) and Ulysses equipped with RTGs (r)

RTG's have been applied on Voyager, Pioneer and Viking deep-space probes, but also on some Nimbus and Transit satellites. Currently they are used on Cassini, New Horizons and the Galileo spacecraft. First US RTGs, developed under the US SNAP (Systems for Nuclear Auxiliary Power) program, produced about 2.7 watts of electric power. The most recently designed system, the General Purpose Heat Source RTG (GPHS-RTG), generates about 290 watts of electric power at BOL and 250 W at EOL. It weighs about 55 kg of which about 11 kg is fuel (about 7.5 kg of isotope fuel, remainder is impurities). Thermal power is 4234 W.
Fuel cell systems have been used on the Gemini spacecraft and the Apollo service module. More recently they have been used on the Space Shuttle. The Apollo fuel cell system consists of three 31-cell hydrogen oxygen fuel cell stacks which provide 28 volts, two cryogenic oxygen and two cryogenic hydrogen tanks. The Space Shuttle operates 3 fuel cell power-plants, each supplying its own isolated, simultaneously operating 28-volt dc bus. The power-plant section of each system consists of 96 cells contained in three sub-stacks.

Dimensioning parameters for both RTG and fuel cell system is peak power level and total energy need. For most design purposes, one aims to keep the required power output steady. If too many power is produced, one aims to shunt the excess power away.

To estimate the mass of a fuel cell system a very first approach is to consider the system as consisting of two elements, being the dry fuel cell system, consisting of the fuel cell power-plant, the feed system and the controls, and the reactants. Notice that for now the reactant storage system is not included, see later in this section.

$$M_{\text{fuel cell system}} = \left(M_{\text{fuel cell system}}\right)_{\text{dry}} + M_{\text{reactants}} \qquad [83]$$

Of these, the dry fuel cell system mass is estimated based on the (maximum) power output (P) to be generated by the system and the fuel cell system specific power (P_{sp}) according to:

$$\left(M_{\text{fuel cell system}}\right)_{\text{dry}} = \frac{P}{P_{sp}} \qquad [84]$$

Reactant mass is estimated based on the total energy (E) to be delivered by the fuel cell system and the reactant consumption rate (C_{rate}):

$$M_{\text{reactants}} = E \cdot C_{\text{rate}} \qquad [85]$$

From [SSE] Tables 10.3 and 10.5, it can be found that the specific power of a fuel cell power-plant (excluding tankage) is in the range 25-300 W_e/kg, where the higher values apply to more modern fuel cell systems. Next to the system itself, we must also to take into account the reactants needed to provide for the required energy. For instance, for an hydrogen-oxygen fuel cell hydrogen-oxygen consumption (mass ratio 1 kg of hydrogen reacts with 8 kg of hydrogen) typically is ~0.5 kg/kWh.

Example: Sizing of fuel cell system
Consider a fuel cell system that is required to deliver 3360 kWh at an average power level of 10 kW (operational life of 336 hours or 14 days) we find for the mass of the fuel cell:

$M_{fc} = P/P_{sp}$ *= 10000 (Watt) / 100 (Watt/kg) = 100 kg*

Here the specific power of fuel cell power-plant is taken equal to 100 W/kg. For the mass of the reactants follows:

3360(kWh) x 0.5(kg/kWh) = 1680 kg

Of which about 186.5 kg is hydrogen and 1493.5 kg is oxygen.

In relation [83] the mass of the reactant storage system actually should have been added to determine the actual system mass. Now once the reactant mass is known, the mass of the required tankage system can be estimated. For this one is referred to the lecture material dealing with launcher design.

For RTGs, the sizing is mostly based on the total power to be delivered and RTG specific power (in W/kg). Depending on the propellant though the power delivered may decrease to a large extend. This is because of the reduction in the amount of radioactive isotope in the reactor. For power P at time t follows:

$$P = P_0 \cdot e^{\left(\frac{-0.693}{\tau_{1/2}}t\right)} \qquad [86]$$

Here P_0 is initial power, and $\tau_{1/2}$ is "half-life" (the time it takes for P to be 1/2 of P_0). From [SSE] the following data for some isotope fuels is obtained:

Table 33: Characteristic data of some radio-isotope fuels [SSE]

Fuel	Symbol	Atomic mass	Half life (Year)	Specific power (W_t/g)	Specific cost (\$/W_t) 1)
Plutonium	Pu	238	90	0.55	3000
Polonium	Po	210	0.38	141	570

Cost figures are based on FY 1999 cost.

From the foregoing equation, it follows that the BOL power to be designed for is higher than the power that is actually used. Note that cost figures are given per Watt of thermal power. In practice, only a few percent of thermal power produced is converted to electric power.

For a derivation of the RTG power equation (equation [86]), it is considered that the power produced decreases at a rate proportional to its value (linear decay model):

$$\frac{dP}{dt} = -\alpha P \xrightarrow{\text{separation of variables}} \frac{dP}{P} = -\alpha dt$$

Integration gives $\ln\left(\frac{P}{P_o}\right) = -\alpha t$

Where P_o is available power at time t = 0.

After rearranging follows: $P = P_o \cdot e^{-\alpha t}$

Now using the information that at half-life the available power has reduced with a factor 0.5, it follows:

$$P = \frac{1}{2}P_o \cdot e^{-\alpha\tau_{0.5}} \Rightarrow -\alpha\tau_{0.5} = \ln\left(\frac{1}{2}\right) \Rightarrow -\alpha\tau_{0.5} = -0.693$$

$$\alpha = \frac{0.693}{\tau_{0.5}}$$

Substitution of the relation for the half time in the relation for P results in equation [86].

<u>Other systems: Batteries as the primary source of energy (primary batteries)</u>

Most launchers use batteries to provide for the required energy. Such batteries come in different sizes and performances. It shall be obvious that large batteries are capable of delivering more energy than small ones. The amount of energy that a battery can deliver per unit of mass or per unit of volume is given by the batteries specific energy (E_{sp}) and energy density (E_δ):

$$M_{bat} = \frac{E}{E_{sp}} \;;\; V_{bat} = E/E_\delta \qquad [87]$$

Some typical values of specific energy and energy density for two important primary battery types are given in the next table.

Table 34: Characteristics of some primary batteries

Parameter	Unit	Silver-Zinc	Lithium Thyonil Chloride
Specific energy	Wh/kg	55-286	300-550
Energy density	Wh/l	80-415	600-1000

Batteries as secondary energy source (for storing energy; secondary batteries)
Rechargeable or secondary batteries are amongst other used on spacecraft equipped with a photovoltaic system. The secondary batteries than provide the required power during the eclipse periods. An overview of secondary batteries and their characteristics can be obtained from [SSE].

The amount of energy to be stored in the battery can be determined from Eq.[67] (eclipse only). For instance given a total power requirement of the loads of 100 W and eclipse duration of 0.5 hour, we find that the total energy needed is 50 Wh. Taking into account a loss of 20% in the distribution system, the battery needs to deliver 60 Wh. In reality, we select batteries capable of providing much more energy. This is because batteries have to undergo many charge-discharge cycles. To allow for many cycles, batteries can only be discharged for a certain amount. How deep a battery can be discharged is given by the Depth of Discharge (DOD). For instance, a DOD of 40% means that the battery's depth of discharge is 40%. It also means that 60% of the full charge of the battery is remaining in the battery. To account for this, we typically select a battery system capable of providing more energy than actually required. For a DOD of 40%, we need to select a battery with a capacity 2.5 (= 1/0.4) times larger than actually needed. So we should select a battery system capable of delivering 150 Wh. For more details, see the appropriate sections in [SSE].

For rechargeable batteries, we need to make a distinction between the energy that flows to the battery when charging and the energy provided to the loads when discharging. The difference between the two is referred to as the battery (system) efficiency. A typical value is in the range 80-90%. For our example, taking an efficiency of 80%, it would mean that the total energy that flows to the battery is a factor 1.25 larger than what flows from the battery. So it follows that for charging we need 75 Wh. It are the battery system efficiency and the efficiency of the power distribution system together that determine the value of η_e in Eq.[67].

Configuration issues
Typical issues that need to be considered include (see also Figure 48):

- Solar panels exposed to the Sun (no or limited shadowing): May need some device to point the arrays towards the Sun and keep them pointed towards the Sun. For GEO satellites, solar arrays usually are mounted onto North and South panel of satellite, which allows for full 360 degree rotation of the panel.
- Body mounted versus wing mounted panels: Body mounted usually limited to low power applications. It allows for a stiffer design or low mass. But temperatures go up, which tend to lower the cell efficiency.
    - Spin control: Not all panels used at the same time
- Single versus multiple wing: Two wings allows for a symmetric design, thereby facilitating the positioning of the CoM.

In addition, we need to consider that power sources generate a lot of heat. To allow radiating this heat into cold space, batteries or fuel cells shall be placed close to or on a cool surface of the satellite.

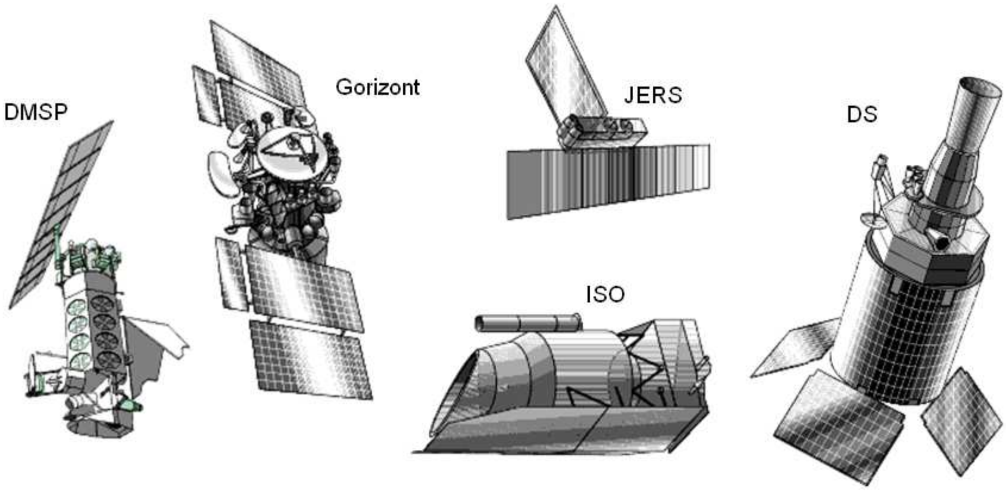


Figure 48: Typical configurations for solar cells

Example: Configuration effects on solar array area needed
Consider a spacecraft that needs to provide 1000 W of power. For this it uses solar array panels that under nominal conditions (normal incident solar radiation) provide for 100 W/m^2. In this example, we will discuss a number of configuration options and their effect on solar array area
The first option is to equip the spacecraft with a single wing that can be rotated about two axes so that solar light is always normally incident on to the panel (compare DMSP or JERS in the above figure). In that case the total solar array area needed is 10 m^2.
The second option is to use two wings that both can be rotated about two axes (see Gorizont). The same area results, but not each wing only has an area of 5 m^2.
A third option is to use body-fixed panels like as is shown for ISO in the above figure. In that case, we need to consider that pointing of the array is determined by the pointing of the spacecraft itself. For ISO the pointing is determined by the telescope and hence we need to take into account that pointing for the array is not optimum and that this may vary with the season. Depending on the season, the Sun's apparent position in the sky may be up to 23.5 deg above the equatorial plane or below. To correct for this angle, we need to increase the solar array area with a factor 1/cos(23.5 deg) = 1.09. So in this case we would need a total solar array area of 10.9 m^2.
A fourth option is to use body-fixed panels on a spinning satellite (compare the Meteosat series satellites). We would essentially need 10.9 m^2, but since the panels are spinning around, it means that only the area projected onto the plane perpendicular to the solar radiation is effective. So to allow sufficient power to be obtained, we would need in total π times 10.9 m^2 = 34.26 m^2 of solar array area.

For self-study: Consider a freely tumbling cubical satellite covered on all sides with solar cells. Determine the fraction of the effective solar array area in relation to the total solar array area. Hint: Determine the area of the largest cross section of the vehicle and determine the fraction of this area to the total surface area.

Problems
A number of problems for exercising upon are available via Blackboard (Maple TA), whereas a few are also contained in a separate workbook available from the TU-Delft online print shop. Of this workbook also an electronic copy is available on blackboard.

4.4 Propulsion

The propulsion system is the whole of hard- and software needed to propel a spacecraft. The propulsive force produced is referred to as the thrust force or simply thrust.

Why propulsion?

To deal with this question, we need to study past missions. For example, from the ESA Mars Express web pages, we learn that the ESA Mars Express vehicle is launched into space using the four-stage Soyuz/Fregat launcher. The Fregat upper stage places the spacecraft on a Mars-bound trajectory. Mars Express on-board propulsion is used for orbit corrections and to slow the spacecraft down for Mars orbit insertion. For this, the velocity of the spacecraft is to be reduced by 2880 kilometers per hour in 30 minutes. The onboard propulsion is also to provide attitude control during the transfer flight and when in orbit about Mars. Likewise, the propulsion system of the U.S. $3.4 billion Cassini spacecraft launched October 15, 1997, from Cape Canaveral, Florida provides propulsion for major changes to Cassini's trajectory en route to Saturn & Titan. In more detail, the system provides for:

- Mid course corrections and plane change
- Capture at Saturn
- Station keeping at Saturn
- Attitude control during all above phases
    - Compensate for disturbance torques
    - Provide for attitude maneuvers

It has a total mission characteristic velocity capability of 1.6 km/s. In addition, it is capable of being fired 17 times en route to Saturn, and will be ignited approximately 150 more times before the end of the mission.

In general, we find that propulsion is needed to …:

- accelerate and/or decelerate a vehicle (orbit insertion, launch, de-orbit, breaking maneuver, landing maneuver)
- maneuver in space (e.g. to change orbit and or to change the orbit plane)
- counteract disturbing forces (gravity, drag, etc) to ensure station keeping, i.e. orbit control
- provide attitude control
- Other

In practice, we find that almost all spacecraft are equipped with some means of propulsion. Some exceptions exist that either do not require a propulsion system or no suitable propulsion system is available yet. The latter is for instance the case for nano- and pico-satellites for which the currently available propulsion systems are way too large and heavy.

Some spacecraft may even be equipped with a primary (or main) and secondary propulsion system. The purpose of the primary propulsion system is to provide thrust needed to launch a spacecraft into space and/or to change orbits for instance for interplanetary travel. Hence, Primary propulsion systems are sometimes also referred to as orbit control systems. Secondary systems are used for small (in terms of characteristic velocity of Δv) maneuvers and for attitude control and steering. Secondary systems are also referred to as reaction control systems, abbreviated RCS.

Key requirements

Key requirements generally relate to allowable or required acceleration/deceleration levels, Δv capability, magnitude of disturbing forces and torques, mission or more specific travel/flight duration, and maintaining a suitable level of availability of the payload. For illustration, the figure 2.4 from [SSE] provides typical tolerance levels to sustained acceleration levels for astronauts, whereas Figure 49 provides an overview of typical velocity changes required to accomplish a Moon mission including landing and return to Earth.

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	

Figure 49: Overview of typical velocity changes required to accomplish some maneuver

Requirements with regard to the first three parameters are generally outcome of trajectory (for launchers), orbit and attitude control analysis (see e.g. material from course AE1110-II). For generating requirements relating to the compensation of disturbing forces the next few paragraphs provide methods to determine/estimate the main disturbing forces on a spacecraft. In general such forces need to be compensated for by propulsive means. For this, the compensating force should be equal to (not taking into account any margins) or larger than the disturbance force. In the latter case it allows for short propulsive burst to correct for the distortions that result from the disturbing forces over a longer period of time. We will discuss gravity, (aerodynamic) drag and the force resulting from solar radiation. Notice that in our discussion, we are more interested in maximum values then in how forces vary with mainly interested in maximum values

<u>Gravity</u>

Launcher propulsion

Rocket motion has been treated in some detail in AE1110-II. From this treatment, we learn that the thrust produced by the propulsion system should be able to accelerate (launcher) or decelerate (lander) the vehicle. Typical values for thrust load (also referred to as thrust to weight ratio) at start of flight as reported range from:

- For launchers: 1.2 – 2.5
- Sounding rockets: 5-35

Other important propulsion related parameters are propellant mass ratio, and burn time. In general we strive for a low propellant mass ratio and a short burn time. The latter is to limit gravitational loss, see again AE1110-II, whereas the former is to maximize the payload mass.

Space propulsion

Current spacecraft typically have a thrust to weight ratio in the range of 0.1 and less. This is because thrust is in a direction perpendicular to gravity instead of opposing gravity as for launchers during the initial phases of the ascent. Higher values are of course feasible, but generally require a tougher spacecraft structure and hence resulting in a reduced payload mass.

For space propulsion essentially two approaches exist:

- Impulsive shot: Short burn time while thrusting at right angle to local gravitational field; g ~0
- Low thrust (more or less continuous) propulsion

An important difference is in the burn time and the associated gravity loss. In general, one aims for short burn times as to limit gravity loss just like for space launchers. For longer burn times, gravity loss for spacecraft can be significant as unburned propellant is being accelerated and moved to a higher altitude during the mission. For instance, using the figure below, we find for the transfer from LEO (28.5 deg)-GEO that high thrust requires a Δv capability of 4220 m/s, whereas the optimized low-thrust value is 5900 m/s. This is about a factor 1.4 higher.

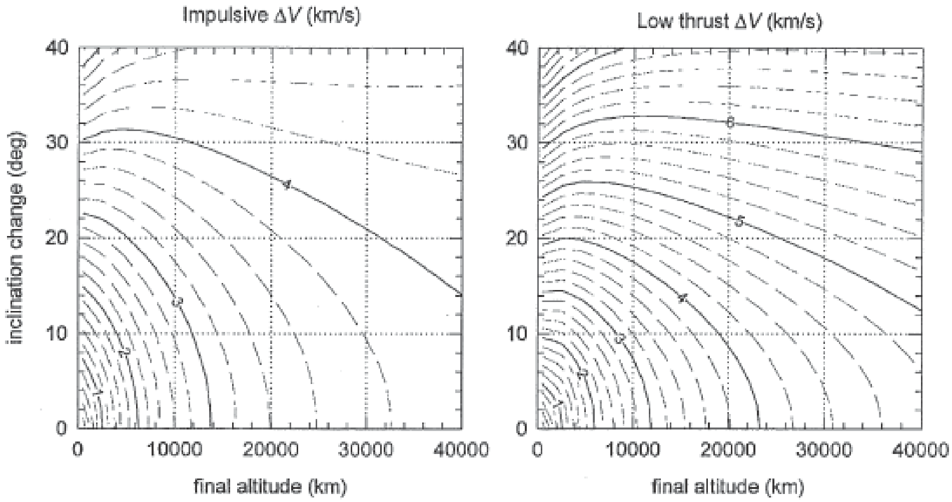


Figure 50: Contour maps of Δv for altitude and inclination change (Initial altitude is 400 km) [Sanchez]

Figure 51 shows the effect of the thrust level on the velocity increment for a Mars mission. It clearly demonstrates that with decreasing thrust level the required Δv capability increases.

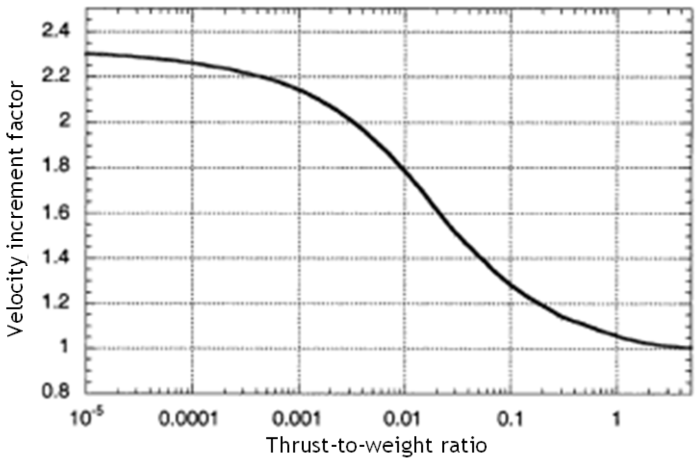


Figure 51: Effect of thrust to weight ratio on mission characteristic velocity (Mars mission) [Turner]

The reason for considering low thrust options with the associated gravity loss is because some low thrust options allow for significantly higher exhaust velocities (up to 100 km/s) then the high thrust options thereby allowing for a much reduced propellant load and hence also increased payload mass.

Drag

(Aerodynamic) drag should be taken into account when orbiting a planet with an atmosphere. The drag force exerted on a satellite moving through some atmosphere can be calculated using:

$$F_a = \frac{1}{2} \cdot \rho \cdot v^2 \cdot S \cdot C_D \qquad [88]$$

- F_a is aerodynamic drag
- ρ is the atmospheric density (depends on the altitude; for values for Earth see appendix H)
- V is orbital velocity (for typical values of circular velocity, see again appendix H)
- S is the frontal projected area
- C_D is the aerodynamic drag coefficient; for Free Molecular Flow[11] its value ranges from 2 to 4

[11] **Free molecular flow** describes the fluid dynamics of gas where the mean free path of the molecules is larger than the size of the chamber or of the object (in this work the spacecraft) considered. It is in contrast to

Values of the drag coefficient for specific spacecraft can be taken from for instance SMAD, table 8-3. A simple method to calculate the drag coefficient of simple shapes in free molecular flow can be obtained from NASA SP 8058, see also material provided on launcher design. Since a body in free molecular flow does not disturb the flow, a complicated shape can be resolved into simple parts, and the contributions of each of these parts can be added together to obtain the coefficients for the entire S/C.

Example: Drag estimation
Consider a satellite in circular orbit at an altitude of 500 km with a frontal projected area of 5 m^2, and a drag coefficient of 2. From appendix H, we find an orbital velocity of 7.613 km/s and a mean (averaged over time) density of 4.89 x 10^{-13} kg/m^3. It follows a drag force on the satellite of:

$$F_a = \frac{1}{2} \cdot 4.89 \times 10^{-13} \cdot 7613^2 \cdot 5 \cdot 2 = 142 \mu N$$

This essentially shows that even at low altitudes, drag force is rather small.

For self-study: What would be the drag force experienced by the foregoing satellite when in orbit at an altitude of 500 km about Mars? The big problem here may be finding the mass density of the Martian atmosphere at the altitude given. Consider also how the different composition of the Martian atmosphere may affect the resulting drag (consider changes in viscosity and hence Reynolds).

For preliminary design purposes, we are mainly interested in the maximum value of the drag force as experienced by the craft and not so much in how it varies with attitude. This is because it is mostly the maximum value that determines the required thrust to be installed on the vehicle. For this reason also we do not take into account all the variations in mass density of the atmosphere, but consider e.g. maximum mass density only.

<u>Solar radiation</u>
The force exerted on the spacecraft by solar radiation can be calculated using:

$$F_s = (1 + \rho) \cdot P_s \cdot S \quad [89]$$

$$P_s = \frac{J_s}{c} \quad [90]$$

- F_s is incident radiation force due to solar radiation
- ρ is reflectivity ($0 < \rho < 1$)
- S is frontal (projected) area
- P_s is solar pressure
- c is velocity of light (c = 3 x 10^8 m/s)

Example: Solar pressure force estimation
For a spacecraft of surface area S (perpendicular to the direction of the solar radiation) = 5 m^2, ρ = 0.5 at 1 AU (I_s = 1400 w/m^2) we obtain a force of 35 microNewton (μN).

continuum flow, where the gas (here air) is considered a continuum, i.e. there are no significant gaps between the air molecules that delay its reaction to external disturbances.

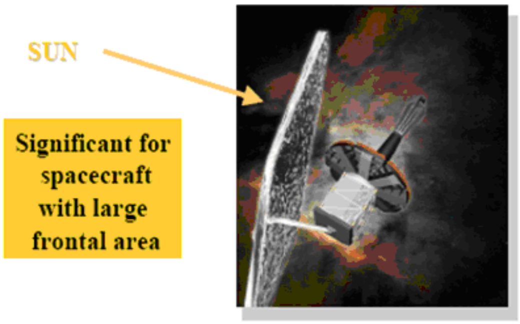


Figure 52: Force induced by solar radiation

The force due to solar radiation is directed along the line connecting Sun and satellite (Sun-Satellite line) and points away from the Sun. Like for aerodynamic drag, we are mainly interested in the maximum value of this force over the mission duration and not so much in how it varies in time.

Propulsion fundamentals
Different options exist to generate a propulsive force including:

- Rocket propulsion: The thrust is generated by expelling mass from within the spacecraft in a direction opposite to the direction of travel/motion.
- Non-rocket propulsion: Next to rocket systems, a whole array of non-rocket systems exist that can be considered for use on spacecraft, provided they can be qualified for flight in time. These include air-breathing propulsion for Earth-to-orbit launchers, but also solar sailing, tether propulsion, and magnetic sails for in-space propulsion.

Since most (99% and more) space propulsion systems use rocket propulsion as the main means of propulsion, we limit our treatment from now on to rocket systems only.

How does it work?
A rocket system expels mass at a high velocity in a direction opposite to the direction of motion. The matter expelled is referred to as expellant or more commonly as propellant. To accelerate the expellant the rocket must exert some force on it (action). From Newton's second law it follows a reaction force works on the rocket equal to the force acting on the expellant, but in opposite direction (action is reaction). This force is referred to as rocket thrust or shortly thrust. Its magnitude depends on the mass expelled per unit time (m), commonly referred as the mass flow rate, times the velocity with which it is expelled (w), also referred to as exhaust velocity:

$$F_T = m \cdot w \qquad [91]$$

Example: Rocket exhaust velocity
A rocket is producing a thrust of 30 kN over a 100 s time span. During the time span the mass of the rocket decreases linearly with 900 kg from 1000 kg to100 kg (see also relation [62]). Mass flow rate in that case is 9 kg/s. Using above relation, we find an exhaust velocity of 30000 N/9 kg/s = 3333 m/s.

So thrust depends on the mass flow rate and the velocity at which the material is expelled. Of these, mass flow rate generally is determined by the propellant (feed) system i.e. the capacity of the pump/pressurization system (see for more details the material covered in the part of the course entitled "Launcher Design and Sizing") and the exhaust velocity by the technology used to accelerate the propellant to a high exhaust velocity. Typical acceleration technologies include:

- Cold gas propulsion, wherein a high pressure inert gas is accelerated to a high velocity by allowing the gas to expand in a nozzle (a special shaped flow channel).
- Thermo-chemical propulsion: Chemical energy is used to heat up a propellant which is than accelerated to a high velocity in a nozzle. As propellants are generally used a combination of fuel and oxidizer that react to free up the required chemical energy. Compared to cold gas propulsion,

this has the advantage of adding thermal energy to the flow, thereby providing for the capability of a much higher exhaust velocity

- Thermal propulsion (arcjet, resistojet, thermo-nuclear): A hot propellant is accelerated to a high exhaust velocity in a nozzle. Main difference with chemical propulsion is that the energy for heating the propellant does not stem from a chemical reaction but from a nuclear reactor or from an electric generator. Compared to chemical propulsion it allows for selecting light (in terms of molar mass) propellants, thereby increasing the attainable exhaust velocity.
- Ion propulsion, wherein electrical energy is used to accelerate ions to a high exhaust velocity. Electrostatic ion thrusters use the Coulomb force and accelerate the ions in the direction of the electric field. Electromagnetic ion thrusters use the Lorentz force to accelerate the ions. Some neutralizer is needed to neutralize the beam and prevent the vehicle from charging.
- Plasma propulsion, which uses the Lorentz force (a force resulting from the interaction between a magnetic field and an electric current) to accelerate a plasma, thereby generating thrust.

The latter two are also referred to as electric propulsion as both require electric power for their operation. In that sense also the arcjet and the resistojet are referred to as electric propulsion.

The attainable exhaust velocity can be obtained from for instance [SSE, Figure 6.2]. It follows:

- Cold gas propulsion: up to 600-800 m/s
- Chemical propulsion: up to 4.5 km/s
- Thermal propulsion: up to 10 km/s
- Ion systems: up to 100-200 km/s
- Plasma systems: up to 100 km/s

In more detail, we find that the values also depend on which propellant is chosen and their mixture ratio. For now we will leave the details for later.

The propulsion system should operate long enough to allow for the required, velocity change to be accomplished. Once the technologies to determine some exhaust velocity have been selected, the total amount of propellant needed solely depends on the mission characteristic velocity (from mission analysis) and the vehicle mass ratio (obtained from comparable vehicles) and can be calculated using the rocket equation, see Eq.[5]. Remember that to minimize propellant mass it is important to maximize the exhaust velocity.

Once propellant mass is known (e.g using the rocket equation) and a thrust value set (for some exhaust velocity), the burn or operation time of the propulsion system can be determined using:

$$t_b = \frac{M_p}{m} \qquad [92]$$

Another important figure of merit for propulsion systems is the specific impulse (I_{sp}). It essentially gives the ratio of momentum delivered by the engine divided by the total propellant weight:

$$I_{sp} = \frac{\int F_T \cdot dt}{M_p \cdot g_o} \xrightarrow{\text{for } F_T \text{ = constant in time}} I_{sp} = \frac{F_T \cdot t_b}{M_p \cdot g_o} \qquad [93]$$

Here t_b gives the time over which the engine operates, M_p is propellant mass and g_o is Earth gravitational acceleration at sea level. A high value of I_{sp} than indicates that for the same amount of total impulse (or momentum) delivered, propellant consumption is low and vice versa. This becomes more obvious when combining Eq. [92] and Eq.[93]:

$$I_{sp} = \frac{m \cdot w \cdot t_b}{m \cdot t_b \cdot g_o} = \frac{w}{g_o} \qquad [94]$$

This equation essentially shows that a high exhaust velocity is identical to a high specific impulse, the only difference being a factor of about 10. The next figure gives typical ranges of I_{sp} for a number of different propulsion systems. It shows that chemical systems have low specific impulse and electric systems (ion and plasma) high specific impulse.

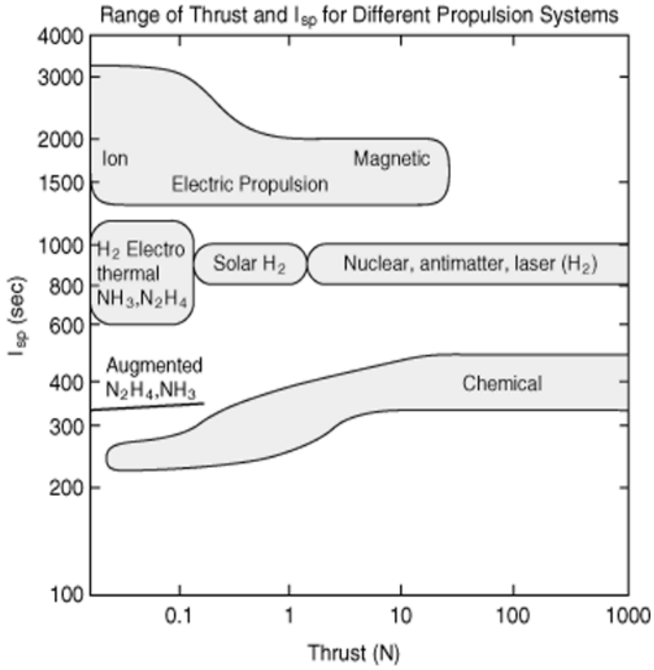


Figure 53: Range of thrust and I_{sp} for different propulsion systems

When comparing the data from Figure 53 with the data from the earlier referred to Figure 6.2 from SSE, we find that velocity ranges indicated do differ slightly. This is explained by different investigators using different data and/or data of an older date.

From [SSE, Figure 6.2] and Figure 53, we also learn that thermonuclear and chemical systems are capable of delivering high thrust, allowing for launcher propulsion, whereas the other systems are limited to low-thrust applications. This is related to the high power levels needed to operate these thrusters, i.e. the required power. How much power is needed can be determined from the jet power P_j:

$$P_j = \frac{1}{2} \cdot F_T \cdot w = \frac{1}{2} \cdot m \cdot w^2 \quad [95]$$

Using:

$$P = \eta_T \cdot P_j \quad [96]$$

With P is input power, and η_T is thrust efficiency ($0 < \eta_T < 1$), i.e. a parameter indicating how efficient the power input to the thruster is converted into jet power. Using the above relations it can be shown that even at low thrust levels high power levels result.

Example: Thruster input power
A thruster producing a thrust of 1 N with an exhaust velocity of 3000 m/s (or specific impulse of about 300 seconds) produces a jet power of 1500 W. In case of a thrust efficiency of 50% it follows an input power needed of 3 kW.

Values of thrust efficiency for the various technologies can be obtained from literature and range from about 10-30% for plasma thrusters, 50-60% for ion thrusters and up to 80-90% for thermal rockets.

The power level is important for non-chemical rocket systems as it provides a measure for how much power should be delivered by some power source during the operation of the rocket and hence may affect the dimensioning and sizing of the power source.

In [SSE, Figure 6.2] also lines of constant specific power, here beam power per unit of vehicle mass, are given. It clearly shows that nuclear and chemical systems both are capable of providing high power per unit mass, whereas for the other systems this is limited to values below 500 W/kg.

Rocket propulsion system elements and configuration

The propulsion system of a spacecraft generally consists of a primary (for large maneuvers) and a secondary system (for small maneuvers and/or attitude control) or reaction control system (RCS). These systems typically differ in the thrust levels used. In addition, they may even use different propellants to allow for a more optimum design. The primary system may be integrated with the secondary system to form a single system or may be a separate propulsive stage, i.e. a propulsive module or kick stage that can be dropped once the propellant is used. This gives as advantage that the mass of the actual vehicle decreases thereby lowering the effect of disturbance forces and mass moment of inertia of the vehicle. The latter allows for reducing the capabilities of the vehicle's attitude control system. An RCS system typically includes many small thrusters providing small amounts of thrust in any desired direction or combination of directions. An RCS is also capable of providing torque to allow control of rotation (pitch, yaw, and roll). Sometimes an RCS is equipped with different small thrusters providing different thrust levels for e.g. East-West and North-South station keeping[12] of satellites.

Some specific propulsion systems and their main characteristics are given in Table 35.

Table 35: Primary and secondary propulsion system characteristics [Sarsfield]

Mission	Spacecraft	Launch Vehicle	Upper Stage	Stabilization Type	Number of Thrusters	Fuel Type
Clementine		Titan II	Star-37	3-axis	12	Hydrazine
Discovery	NEAR	Delta II 7925		3-axis	11	Hydrazine
	Mars Pathfinder	Delta II 7925	Star-48B	Spin	8	Hydrazine
Explorer	SMEX-SWAS	Pegasus XL	n/a	3-axis	n/a	n/a
	SMEX-TRACE	Pegasus XL	n/a	3-axis	n/a	n/a
	MIDEX-MAP	Delta 7325	Star-48	3-axis	6	Hydrazine
New Millennium	Deep Space 1	Delta 7326	Star-37	3-axis	8	Hydrazine
	Earth Observer 1	Delta 7320	n/a	3-axis	4	Hydrazine
SSTI	Lewis	LMLV-1		3-axis	8	Hydrazine
	Clark	LMLV-1	n/a	3-axis	2	Hydrazine
Surveyor	Mars Global Surveyor	Delta II 7925	PAM-D	3-axis	12	Hydrazine
	Mars Surveyor '98—Lander	Delta 7425	Star-48	3-axis	8	Hydrazine
	Mars Surveyor '98—Orbiter	Delta 7425	Star-48	3-axis	8	Hydrazine
Baseline	RADCAL	Scout	n/a	Grav. Grad.	n/a	n/a

From this table follows:

- Launch vehicle forms part of the transportation system needed to get the craft at its destination
- In some cases a separate upper stage or kick stage (for instance PAM-D) or a kick motor is used. The latter is usually integrated in the satellite. Typical propellants used are mostly solids (Star[13] series of motors) that allow for a low cost and simple design thereby allowing for short thrust times (reduced gravity loss).

[12] http://en.wikipedia.org/wiki/Orbital_station-keeping#Station-keeping_in_geostationary_orbit.5B1.5D

[13] Thiokol's Star family of space motors provides propulsion for spacecraft and launch vehicle upper stages. The Star number for each motor indicates its approximate principal diameter in inches.

- Not all spacecraft (for instance SMEX and RADCAL) equipped with a propulsion system
- Number of thrusters for the RCS ranges from 2-12. Typically 4 thrusters needed per axis in case of pure torques. A lower number of thrusters may indicate that other means are used on board of the spacecraft to provide for attitude control.
- RCS uses mostly hydrazine propellant. This is because it is a monopropellant allowing for a relatively simple system. Also hydrazine offers good storage life and is self-igniting (hypergolic). For long-life spacecraft we may also use a storable bipropellant, like mono-methyl hydrazine as fuel and nitrogen tetroxide as oxidizer, as an alternative, see later in this text. This allows for reducing propellant mass.

Different propulsion systems may be used on board of spacecraft. Still the main elements of the various systems are essentially the same. As the main elements of any rocket propulsion system we distinguish, see Figure 54:

- one or more thrusters/engines/rocket[14] motors, i.e. the thrust generation system
- propellant, which makes up the mass to be expelled
- propellant system (compare fuel system for aircraft):
- power-plant or power source that provides for the power (energy) needed for propulsion; For chemical rockets such a power-plant is absent as the propellants themselves also act as the power source, however, for various other types of rockets, like ion rockets and plasma rockets we do need to take into account the presence of a separate power source

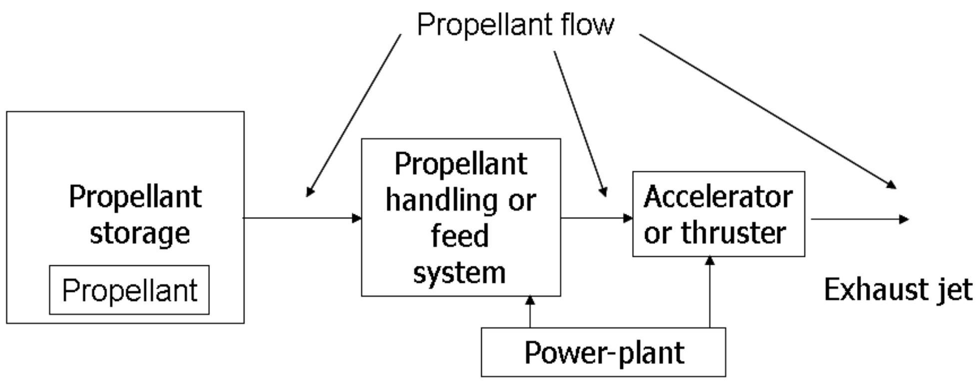


Figure 54: Rocket propulsion system elements

The thrust generation system generally consists of one or more thrusters see for instance the Table 35. Some typical thrusters are shown in Figure 55.For main propulsion purposes; we typically find a single thruster suffices (per axis). However, to increase reliability, one sometimes tends to implement multiple thrusters. It also allows for using smaller thrusters and hence a better usage of the available volume in the spacecraft.

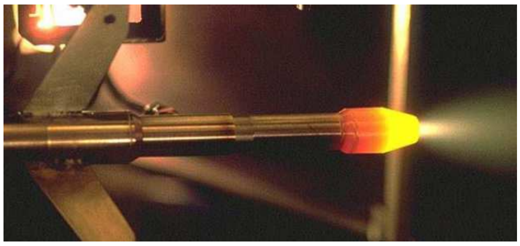
[14] **Engines/motors:** The larger of a spacecraft's propulsive devices, perhaps producing a force of several hundred Newton, used to provide the large torques necessary to maintain stability during a solid rocket motor burn, or they may be the rockets used for orbit insertion. Usually the word motor is reserved for solid chemical propulsive devices and engines for liquid chemical propulsive devices.
Thrusters: A set of small propulsive devices, typically generating between less than 1 N and 10 N, used to provide the delta-V required for interplanetary trajectory correction maneuvers, orbit trim maneuvers, reaction wheel de-saturation maneuvers, or routine three-axis stabilization or spin control.
Thruster and engine are sometimes also referred to as motor.

A) Cold gas thruster

B) Monopropellant thruster

C) Solid propellant kick motor

D) Ion thruster



E) Arcjet

Figure 55: Typical chemical and non-chemical thrusters

The feed system typically consists of a propellant storage system that stores the propellant and a propellant handling system that ensures the proper flow of propellant to the thruster. Typical components include fluid tanks for storage, piping or propellant tubing to distribute the propellants to the proper thruster and various valves and regulators to control the propellant flow.

For all means of propulsion some power source is needed providing for the required power. For instance, ion and plasma rockets require an electrical power source and nuclear rockets require a nuclear power source. Figure 56 shows a schematic of an electrical propulsion system with the electrical power source highlighted in the figure. Also chemical rockets require some power source, however, for a chemical rocket no separate power source is needed as the power stems from a chemical reaction between the various propellant constituents.

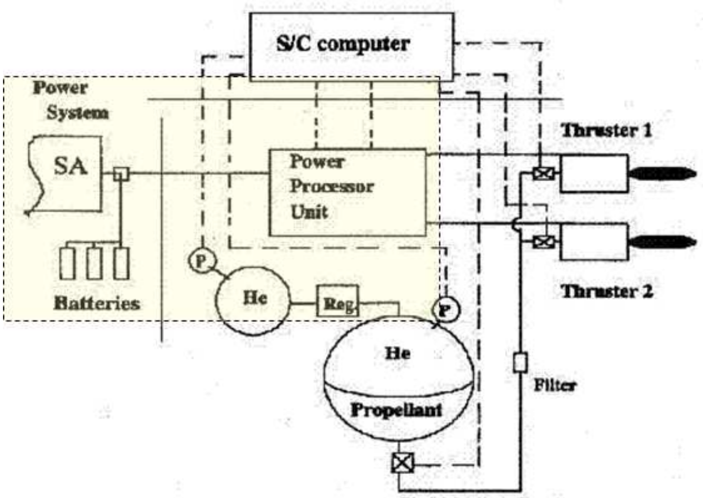


Figure 56: Schematic of electrical propulsion system

To show the various elements that make up the propulsion system and how they are related with each other, usually a propulsion system schematic is used of the type shown in Figure 56 and Figure 57. The latter represents a chemical propulsion system. From such figures we can make out the number of thrusters, the number of tanks, how the thrusters are connected to the tanks, the number and types of valves used, how the propellants are fed to the thrusters and so on. Notice the absence in the figure of a separate power source.

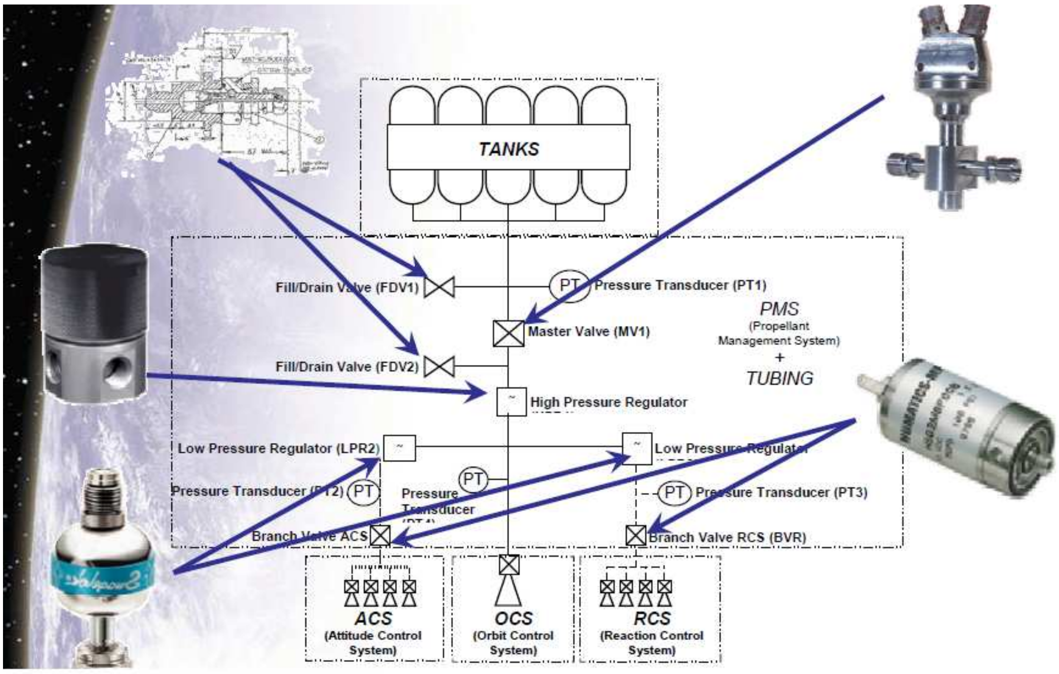


Figure 57: Schematic of a typical spacecraft RCS

Figure 58 shows the propulsion system of the Cassini spacecraft. It consists of a combined (primary and secondary) system. Propulsion for major changes to Cassini's trajectory is provided by one of two main engines. These powerful engines use mono-methyl hydrazine as fuel and nitrogen tetroxide as oxidizer. Sixteen smaller thrusters use hydrazine to control Cassini's orientation and to make small adjustments to the spacecraft's flight path. The fuel (MMH) and oxidizer (NTO) are each stored in their own tank. The hydrazine for the smaller thrusters is stored in the monopropellant tank. The propellants are forced from the tanks to the thrusters by high pressure Helium stored in four Helium high pressure pressurant tanks. A pressure regulator regulates the pressure down to a value acceptable for the piping and the propellant tanks. Finally a range of filters, valves and pressure sensors allow for a proper distribution of the propellants to the various thrusters.

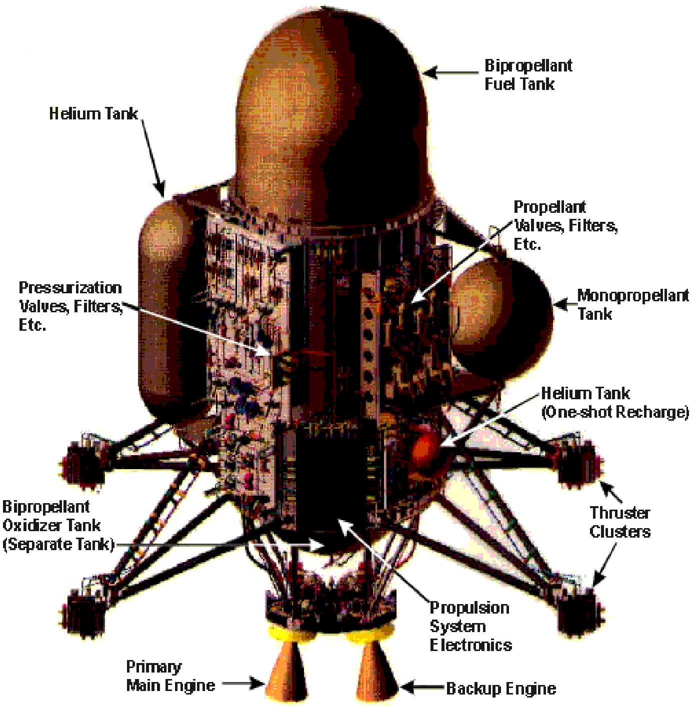
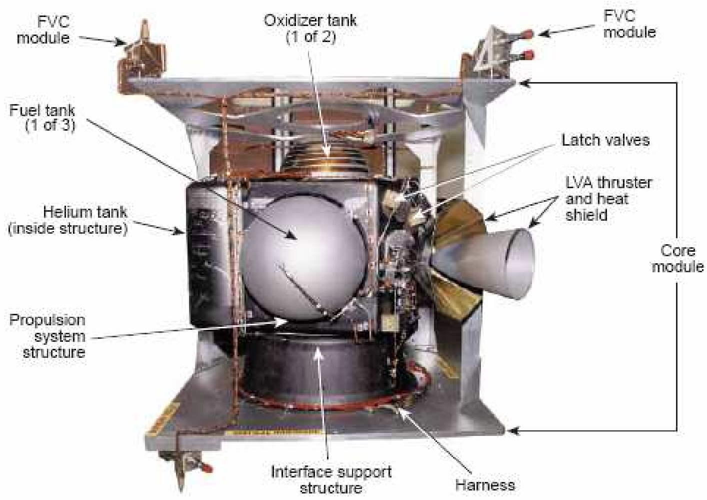


Figure 58: Propulsion system of Cassini spacecraft (Courtesy NASA)

Figure 59 shows the propulsion system of the Near Earth Asteroid Rendezvous - Shoemaker spacecraft (NEAR Shoemaker), renamed in honor of Gene Shoemaker, which was designed to study the near Earth asteroid Eros from close orbit over a period of a year. The craft is three-axis stabilized and uses a single 450 N bipropellant (hydrazine and nitrogen-tetroxide) main thruster, and four 21 N and seven 3.5 N hydrazine thrusters for propulsion, for a total delta-V potential of 1450 m/s. Attitude control is achieved using the hydrazine thrusters and 4 reaction wheels. The propulsion system carries 209 kilograms of hydrazine and 109 kilograms of nitrogen-tetroxide oxidizer in two oxidizer and three fuel tanks.



NEAR propulsion system configuration.

NEAR mass summary.

Item	Mass (kg)
Major assemblies	
Structure	33.1
Helium tank assembly	10.1
Oxygen tank assembly	11.9
Fuel tank assembly	23.4
LVA assembly	9.9
FVC modules	10.3
Valves, electrical, thermal	19.3
Total dry mass	118.0
Helium	1.6
Usable N_2O_4/N_2H_4	315.1
Residual propellant	3.0
Total wet mass	437.7

FVC is fine velocity control
LVA is large velocity actuator

Figure 59: NEAR propulsion system module lay-out and mass characteristics

From the foregoing two figures it follows that the storage system makes up a significant part of the propulsion system especially when a high mission characteristic velocity is required. Other important configuration issues include:

- Exhaust of thrusters must be pointed away from the spacecraft. This is to prevent damage of the spacecraft by the hot exhaust
- Main propulsion: Work line of thrust shall be pointed through the CoM of the spacecraft as to not produce a disturbance torque
- To allow for pure control, 4 thrusters are needed about each axis.
- Multiple thrusters can be used to increase system reliability
- In case non-pure torques are allowed, we can do with less thrusters.
- Tanks shall be installed close to the CoM as not to provide disturbing torques (sloshing) and influence MMOI.
- A heat shield may be incorporated to limit thermal radiation from the thruster to the spacecraft
- Some thrusters are mounted outside the vehicle. This is to limit heat flowing from the thruster to the spacecraft
- Some thrusters are mounted under an angle. This is to allow for reducing the number of thrusters and/or to prevent the nozzle exhaust from touching sensitive areas.

Sizing and dimensioning

As a first approach thrust can be determined based on historical data for comparable vehicles. In later phases, more detailed analysis may follow including the calculation of disturbance forces and gravitational losses. Typically the thrust force must be sufficient to compensate for disturbing forces and/or to overcome gravity and to allow for acceleration or deceleration. Sometimes the disturbing force is thus small that we need to resort to pulsed thrusting. In that case we allow the disturbance to grow over some period of time and then use a short thrust period to compensate for the disturbance. This allows for using thrusters with thrust levels attainable in practice. For instance, for attitude control several 1000 (or more, depending on the mission) operating cycles (on/off) may be needed, meaning that the thrusters are started and stopped several 1000 times.

Likewise, propellant load can be estimated using historical data for comparable vehicles. However, as indicated earlier, see chapter 3, for primary propulsion purposes a more accurate result can be obtained using the rocket equation. For this though mission characteristic velocity, vehicle mass ratio and the specific impulse (or exhaust velocity) delivered by the propulsion system need to be known.

Instead of the vehicle mass ratio, see chapter 3, also the final (empty) or initial vehicle mass may be considered known. In that case and considering that propellant mass M_p is given by the difference between initial mass M_o and final mass M_f:

$$M_p = M_o - M_f \quad [97]$$

We can derive the following equations for propellant mass:

$$\begin{aligned} M_p &= \left(1 - e^{-\Delta v / w}\right) M_o \\ M_p &= \left(e^{\Delta v / w} - 1\right) M_f \end{aligned} \quad [98]$$

These equations relate propellant mass to initial and final mass, respectively and again stress the importance of a high exhaust velocity for minimizing propellant mass.

Note that here we have replaced final mass M_f by empty mass M_e. This is because propellant may also be spent in multiple burns rather than in a single burn. In that case it is better to use final mass, where final mass refers to the mass after the burn rather than the empty mass, i.e. the mass of the vehicle when all propellant is spent.

When calculating propellant load, one should realize that specific impulse depends on the propellants selected, see for instance [SSE, table 6.1] and that for some systems different propellants may be used for different functions. In general, the propulsion engineer is required to assess different propellants for use on some spacecraft for some mission. This in general includes many aspects, but in the early stages of the design the assessment is mostly limited to mass and size considerations and ease of use.

Example: Propellant mass estimation
You are designing a spacecraft with a Δv capability of 2.5 km/s. Total mass at start of the manoeuvre is 1000 kg. Determine for this spacecraft the propellant mass to be carried on board.

Solution 1: Selecting a chemical rocket with a rocket exhaust velocity of 3000 m/s, we find a mass ratio R of 2.30. Given a total mass of start of 1000 kg, this will mean that the mass at the end of the manoeuvre is reduced to 1000/2.30 = 435 kg or just about 565 kg of propellant is expelled at a velocity of 3000 m/s to attain a velocity change of the vehicle of 2.5 km/s. Selecting a thrust level of 1000 N (which gives an initial acceleration of just 1 m/s^2 or about 0.1 g_o) than allows us to calculate a propellant mass flow rate. Using Equation [91] *we find a mass flow rate of 1000 N / 3000 m/s = 0.333 kg/s. Given the total mass of propellant of 565 kg than follows for the operation time of the thruster 565 kg / 0.333 kg/s =~1697 sec or about 28-29 minutes.*

*Solution 2: As an alternative, we could also select an ion propulsion system with say an exhaust velocity of 30 km/s. With the rocket equation we find a mass ratio of 1.11, which gives an empty mass of about 905 kg and hence a propellant mass of just 95 kg (as compared to 565 kg for the chemical rocket motor). For an arbitrary thrust level of 1000 N, it follows a mass flow rate of 0.0333 kg/s and an operation time of about 280-290 minutes. However, if we now estimate the beam power, it follows a power level of 0.5 (0.0333) * (30000)2 = 15 GW. This is too much for most current EPS with 15-30 kW currently being an upper limit. To reduce the beam power, the thrust level needs to be reduced. For instance, for a thrust level of 1 N we find a beam power of 15 kW. As a consequence though, operation time of the thruster is increased to more than 4500 hrs (or just about half a year). So to reduce the power required by the ion propulsion system, we select a low thrust value, but in that case we must take into account that the thrust time increases. Moreover, even at this reduced thrust level, we find (using Eq.**[19]**) that the mass of the power source (0.04 kg/W x 15000 W = 600 kg) more than offsets the reduction achieved in propellant mass due to the high exhaust velocity.* *In the above example we have for now neglected the effect of thrust level on the required Δv capability. This will be dealt with in later courses dealing with low-thrust trajectories. We have also neglected that power conversion efficiency is not equal to 100%. Finally, we mention that power could also be reduced by selecting a lower exhaust velocity. This though increases the propellant mass, but reduces the mass of the power source.*

Next to thrust and propellant load, also the dry mass, i.e. the hardware mass, and the size of the propulsion system need to be determined. Some simple methods suitable for the early design stages are dealt with in the next few sub-sections.

<u>RCS dry mass estimation</u>

System dry mass (in kg) of chemical RCS can be estimated based on known propellant mass (M_p) using the following simple scaling rules:

RCS type	Estimating relationship	
Cold gas	$M_{rcs} = 0.99 \cdot M_p + 6.71 \;\; ; \;\; SEE \; = \; 42\,\%$ 5 data points; Propellant mass in range 2-40 kg	[99]
Monopropellant	$M_{rcs} = 0.178 \cdot M_p + 7.69 \;\; ; \;\; SEE \; = \; 8.1\,\%$ 15 data points; Propellant mass in range 30 – 300 kg	[100]
Bipropellant	$M_{rcs} = 0.0348 \cdot M_p + 58.15 \;\; ; \;\; SEE \; = 6.0\%$ 10 data points; Propellant mass in range 700-1800 kg	[101]

All mass values in kg.

For non-chemical RCS systems, no such rules are available yet due to lack of statistical data. As a first approach, however, one could apply the relationship [100] and then add the mass of the (electrical or thermal) power supply. The mass of the latter could be estimated using:

$$M_W = \alpha_W \, P_W \qquad \text{(compare FSS, eq. 6-26)} \qquad [102]$$

With:
α_W = specific mass of power supply (in kg/W); 1/ α_W = specific power.
P_W = power output of power supply ($P_W = P_j/\eta$ where P_j follows using Eq.[95])

Typical specific power values are (see also SSE propulsion web pages):

- Thermal power-plants
  - Radio-isotope: 25-170 W_t/kg
  - Nuclear-thermal: 300-4000 kW_t/kg
  - Solar collector-receiver at 1 AU: 200-2000 W_t/kg

- Electric power-plants:
    - Photo-voltaic array: 10-40 W_e/kg (compare Eq.[19])
    - Photo-voltaic system (incl. batteries): 7-12 W_e/kg
    - Nuclear-electric: 2,5-100 W_e/kg

Referring to the foregoing example, we applied Eq.[19] to come up with a mass estimate of the power-plant. However, in case also batteries are needed (in case of eclipses), the mass of the power-plant could further increase.

For a more in depth discussion of the various energy sources, their working principle and the vehicle design implications, you are referred to the section on electrical power generation earlier in this lecture series.

Kick motor dry mass estimation
Solid propellant kick motor dry mass can be estimated using:

$$\left(M_{KM}\right)_{Dry}[kg] = 0.071 \cdot \left(M_p[kg]\right) + 18.97 \qquad [103]$$

The above relationship is based on 9 data points in the range 300 to 9500 kg and has an SEE of 16.5%.

Liquid chemical systems are usually integrated in either a kick stage and/or the RCS. So no special estimation relationships are provided here.

Kick stage dry mass estimation
Kick stage[15] dry mass can be estimated using:

$$\left(M_{stage}\right)_{Dry} = 10 - 25\% \text{ of } \left(M_p\right)_{stage} \qquad [104]$$

Compare this relation for instance with the data given for the NEAR propulsion module in Figure ***59***. The data shows a dry mass of 118 kg and a total propellant load of 318.1 kg, which gives a dry mass to propellant mass ratio of 37%. It shows that the NEAR propulsion module is a relatively heavy module. This is mainly attributed to the relatively small propellant load.

Size/volume estimation
For chemical systems, it is the size of the propellant storage that determines to a large extent the size/volume of the propulsion system. Based on the known propellant load and using the propellant information from SSE or from Braeunig, the propellant volume can be estimated. Tank or storage volume typically is a factor 1.1 - 2.0 larger. To take into account the other items (piping, thrusters) it is advised to take 10-20% of the tank volume.

For non-chemical systems the same rules apply, but we need to take into account the volume/size of the power source. For electrical systems, one is referred to the section on electrical power generation. For thermal power systems no such relationships are available due to lack of information.

[15] A kick stage differs from a kick motor in that a kick stage may consist of not only the kick motor, but also of avionics, an RCS, a separation system, interfaces with other stages and/or the payload (like payload ring, electric connectors, etc.). Read for instance the specification of the Inertial Upper Stage (IUS) on http://www.braeunig.us/space/specs/ius.htm

Example: Propulsion system volume
Consider a rocket system carrying on board a hydrogen-oxygen propellant load of 1500 kg. From SSE, we learn that this propellant has a mean density of about 280 kg/m^3. This hence gives a propellant volume of 5.4 m^3. Using a factor of 1.1, we obtain a tank volume of 5.9 m^3. To take into account all other items, we add a further 15% (a figure somewhat arbitrarily chosen) of propellant volume, meaning that the total volume of the propulsion system is estimated at 6.7 m^3.

Note when selecting hydrazine with nitric acid as an alternative propellant, the total tank volume becomes 1.37 m^3. To this we should add some 0.2 m^3 to take into account thrusters and piping. However, this does not take into account yet the additional propellant needed to make up for the reduced specific impulse of the hydrazine – nitric acid combination.

Minimizing system mass
The mass of the propulsion systems follows from the sum of propellant mass and propulsion system dry mass. From the rocket equation it follows that to minimize propellant mass we need to maximize the velocity with which the propellant is expelled. But does that also mean that in that case propulsion system mass is minimal or is it possible that by minimizing the propellant mass we increase propulsion system dry mass and hence cancel out the gain made by increasing the exhaust velocity.

This topic will be dealt with in more detail in a later course (AE2203).

Problems
A number of problems for exercising upon are available via Blackboard (Maple TA), whereas a few are also contained in a separate workbook available from the TU-Delft online print shop. Of this workbook also an electronic copy is available on blackboard.

4.5 Attitude Determination and Control

The motion of a spacecraft in space is specified by its *position*, *velocity*, *orientation* and *rotational rate*. The first two quantities describe the *translational* motion *of* the centre of mass of the spacecraft and are the subject of celestial mechanics or orbit mechanics, while the latter two quantities describe the *rotational* motion of the spacecraft body *about* the centre of mass and are the subject of this lecture.

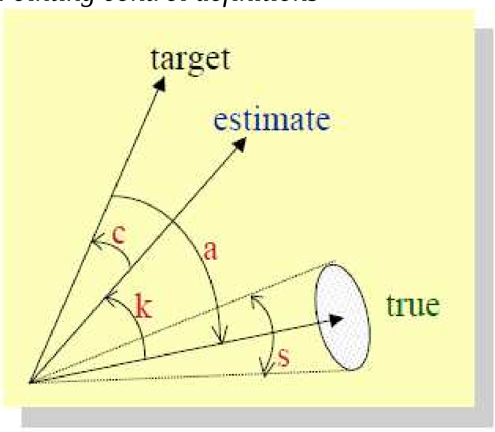
Some definitions

- The *attitude* of a spacecraft is its orientation in space.
- 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.

Why attitude determination and control?
Attitude determination and control is necessary to measure and control the orientation of the satellite, its instruments and appendages throughout the mission life. In more details it means that the ADCS system:

- Orients and reorients the satellite, its instruments and appendages (point sensors, align thrusters) as needed. For instance, the ADCS may provide control inputs to the Solar Array Drive Mechanisms (SADM), which change the orientation of the solar arrays.
- Stabilizes the satellite (maintain desired orientation and sensor pointing angles) by minimizing the effects of disturbance torques (external and internal), thereby preventing "blurring" of images acquired.

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

Figure 60: Pointing control definitions (from AE1110-II)

Axis definitions
To allow describing the angular motion of a spacecraft two axis systems are needed. One is the local orbit reference frame (indicating local vertical and horizontal) and the other is a spacecraft axis system. Both systems are orthogonal systems. The spacecraft axis system is used on board spacecraft amongst others to:

- Identify location of equipment on board of the spacecraft
- Identify viewing directions relative to the spacecraft

The origin of the orthogonal spacecraft axis system generally is taken in one of the outer corners of the spacecraft as this allows for using positive coordinates for most equipment. Some designers use the CoM as the origin of the spacecraft axis system, but the CoM is very susceptive to change. Another option would be to select the spacecraft (body) geometric center, which is relatively better defined than the CoM.

Knowing where an item is located in the S/C or how it is pointed relative to the S/C body axis system does not tell yet in what direction the item is pointing in space. For this, we need to know the orientation of the body axis frame relative to e.g. the local orbital reference frame. The latter is shown in the next figure.

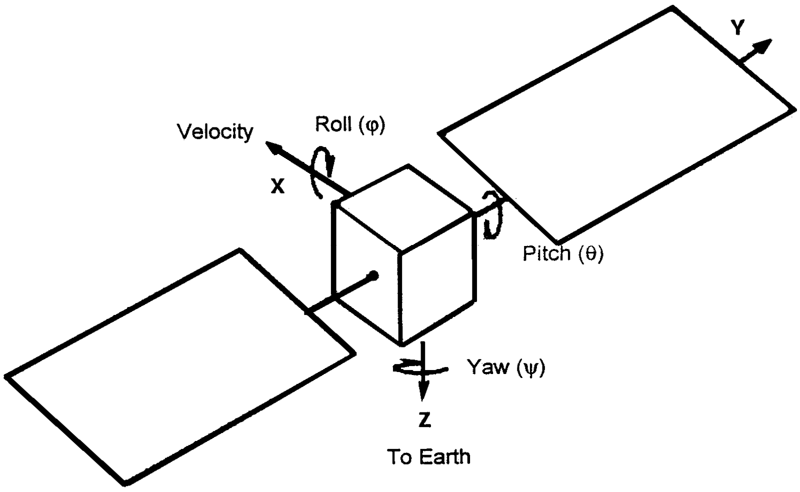


Figure 61: Definition of local reference frame

- Z-axis is collinear with the line connecting Earth's center and the satellite. It defines the yaw axis and is usually taken positive in the direction of earth.
- Y-axis is perpendicular to the orbital plane. It defines the pitch axis.
- X-axis completes the set of orthogonal axes. It lies in the orbital plane and defines the roll axis. It is taken positive in the direction of the flight velocity, but does not coincide exactly with the velocity vector due to the eccentricity of the orbit.

Key requirements:
- Pointing direction (for instance Earth pointing or Sun pointing)
- Pointing accuracy (control) and pointing knowledge, see Table 36

Table 36: ADCS pointing characteristics [Sarsfield]

Mission	Spacecraft	Bus Pointing Accuracy (degrees)	Bus Pointing Knowledge (degrees)	Stabilization Type
Clementine		0.0500	0.030	3-axis
Discovery	NEAR	0.1000	0.003	3-axis
	Mars Pathfinder	1.0000	n/a	Spin
Explorer	SMEX-SWAS	0.0008	—	3-axis
	SMEX-TRACE	0.0060	—	3-axis
	MIDEX-MAP	0.0300	—	3-axis
New Millennium	Deep Space 1	0.2000	n/a	3-axis
	Earth Observer 1	0.0090	n/a	3-axis
SSTI	Lewis	—	0.004	3-axis
	Clark	2.0000	0.020	3-axis
Surveyor	Mars Global Surveyor	0.5700	0.180	3-axis
	Mars Surveyor '98—Lander	n/a	n/a	3-axis
	Mars Surveyor '98—Orbiter	1.1000	n/a	3-axis
Baseline	**RADCAL**	**10.0000**	**5.000**	**Grav. Grad.**

Figure 62 shows that for scientific missions orientation accuracy over the period 1970-2010 has increased with about a factor 100. This leads to a factor 3 increase in accuracy per decade. Note: 1 arcsecond is 1/3600th of a degree.

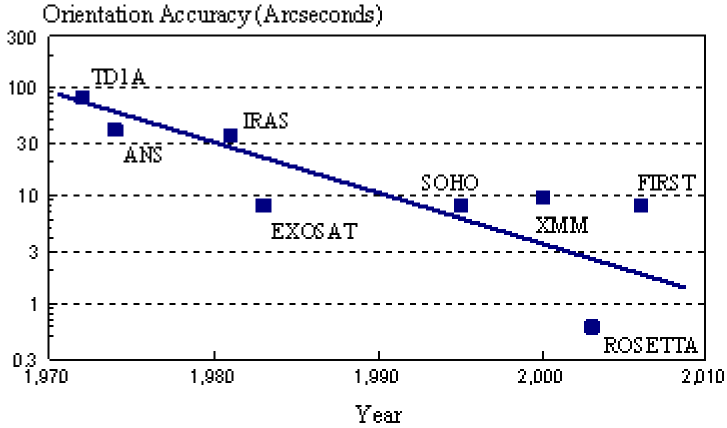


Figure 62: Trend in orientation accuracy for ESA scientific missions

ADCS requirements depend on the type of instruments used, the required pointing direction and the pointing stability. For instance, a deep space probe equipped with a large antenna to communicate with the ground station on Earth may have a beam width, i.e. angle measured in a horizontal plane, between the directions at which the intensity of an electromagnetic beam is sufficient for communications, of less than 1 degree. To ensure that the ground station on earth can receive the message, the beam should be pointed towards Earth and it should remain that way over the duration of the communications exchange.

Example analysis for ADCS requirements generation
Problem (1): Consider observing Earth from an altitude of 500 km using a nadir looking camera, which is capable of taking photos, which each cover an area of 10 by 10 km on ground. The camera is equipped with a mechanism that ensures that the center location of the image remains fixed provided that the platform is perfectly stable. To ensure that the center location of the photo is within 10 m from the desired center location the required pointing accuracy (angle) is ….

Solution (1):The value is determined as follows: 10 m from an altitude of 500 km indicates that we may be off 0.00115 degree or 0.00115/(1/60) = 0.069 arcmin.

Problems (2): Consider some disturbance torque acting on the satellite, causing the satellite to rotate. Because of this rotation, the image taken (see problem 1) is blurred. Suppose that the spacecraft is allowed to rotate over 0.01 arcsec over the period that a photo is taken. What is the spacecraft pointing stability required?

*Solution (2) First we determine the time it takes for a single photo to be taken. At 500 km altitude the circular velocity is 7.613 km/s (see appendix H). Earth radius is 6378 km. Hence, it follows a ground velocity of 7.613 km/s/(6378.1km + 500km) * 6378.1km = 7.06 km/s. So in 1 second the sub-satellite point travels 7.06 km over ground. Since a photo covers 10 km in along track direction it means that every 10km /7.06km/s = 1.42 seconds a photo needs to be taken to allow for a perfect fit of successive photos (without any overlap and/or gap in between successive photos). Over this period, the pointing accuracy should be better than 0.01 arcsec to prevent the center location to move too far off. So the pointing stability of the satellite should be better than 0.01 arcsec/1.42 s = 0.007 arcsec/sec. Of course the required pointing stability reduces when the angle over which the satellite is allowed to rotate increases and or the time required for the photo reduces, etc.*

Different modes of operation may be distinguished for the attitude determination and control system each with their own requirements. Typical modes of operation include:

- Launch mode
- De-tumble mode: Reduce rotation rates to near zero (from separation)
- Attitude acquisition: Find Sun, Earth, Stars etc. by sweeping
- Normal mode: Normal operation such as pointing for science
- Delta V or thrust mode: Attitude control to enable thrusting
- Communication mode: May require rotating the spacecraft or antenna to allow for communications.
- Safe mode: Response to a fault, stable state in which to wait for commands

For each mode different requirements may result for the ADCS.

Fundamentals of attitude control

In this section we will discuss some of the fundamentals of attitude control. These fundamentals are needed to allow determining the effect of both disturbance and control torques acting on a satellite on the satellite motion. To this end, we will consider the rotational motion of a spacecraft w.r.t. the body axes system. This is an orthogonal axes system fixed to the body with origin in CoM, x-y plane is ground plane of the spacecraft with x-axis preferably in direction of flight, z-axis perpendicular to ground plane).

<u>Fundamentals of rotational motion (simplified)</u>

Here we will limit ourselves to a rigid spacecraft/body rotating about one axis. Later in this lecture series the rotational motion is analyzed in more detail. For an object with a fixed mass that is rotating about a fixed symmetry axis, angular momentum is expressed as the product of the moment of inertia of the object and its angular velocity vector:

$$H = I \cdot \omega \qquad [105]$$

- I is mass moment of inertia (MMOI) of the object
- ω is angular velocity.

Angular momentum is important in physics because it is a conserved quantity: the angular momentum of an isolated system stays constant unless an external torque acts on it.

$$T = I \cdot \alpha \qquad [106]$$

Here α is angular acceleration.

For constant acceleration maneuver we find:

$$\Delta\theta = \omega \cdot t = \frac{1}{2} \cdot \alpha \cdot t^2 + \omega_o \cdot t \qquad [107]$$

$$\omega = \alpha \cdot t + \omega_o \qquad [108]$$

Here $\Delta\theta$ is angle over which the S/C rotates in time t, t is time that the torque acts on the S/C, ω_o is initial angular velocity.

Since in space there is essentially no friction $\Rightarrow$ Satellite keeps on rotating. This requires that in space we not only need to initiate motion, but we also need to actively stop the motion. Moreover every disturbance will cause the satellite to rotate and since motion is not damped even a small disturbance can lead to a large pointing error. Many small disturbances in space see hereafter

Example (1): Angular velocity
Consider a satellite that experiences a constant disturbance torque about one of its principal axis of 10^{-4} Nm. MMOI about this axis is 1000 kg-m^2. If this disturbance torque is not counteracted than the spacecraft experiences an angular acceleration of 10^{-7} rad/s^2. After just one day, the vehicle rotates with an angular velocity of 0.00864 rad/s or ~0.5 deg/s. This means that it will rotate 180 degrees in 360 seconds.

Example (2): Rotation angle
From the readings of a spacecraft sensor it follows a constant acceleration about the y-axis of the S/C body frame of 0.02 mrad/s^2. If unattended, we find that the vehicle rotates over an angle of 0.036 rad (2.06 degrees) in 60 seconds or 206 degrees (almost a full revolution) in 10 minutes.

Disturbances
Disturbances to the attitude are due to the gravity gradient, solar radiation pressure, aerodynamic drag etc. Most of these disturbances do vary in time, depending on the position of the spacecraft in orbit, the spacecraft's attitude, the solar intensity as well as the strength of the remnant magnetic field of the spacecraft. For a very first design though, we neglect all these details and focus on determining maximum values. Here lies also the focus of this course. Hence, in this text we present only simple methods that allow for estimating the magnitude of the torque contributions of some important disturbances. Exercises aimed at calculating the disturbance torques are provided for in blackboard.

External torques
In this section simple methods are described that allow for determining a first estimate of the major external torques acting on a spacecraft. More details can be obtained from [SSE, sections 9.4.2 to 9.4.5]. See also [SSE, section 9.2.2] to learn about the region where certain torques are dominant.

Gravity gradient torque
"Tidal" force due to gravitational field variation. This disturbance torque especially plays a role for long extended bodies. Gravity gradient torque tends to align the axis of minimum Mass Moment Of Inertia along the local vertical, as shown in Figure 63. This compares well with a floater on the water that always turns back in the up-right attitude. The disturbance is 0 for a symmetric spacecraft.

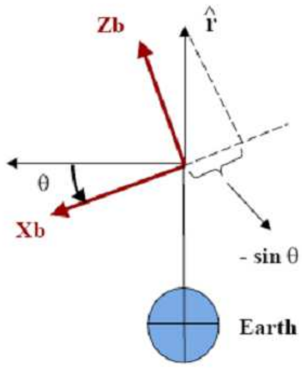


Figure 63: Vehicle attitude in relation with local vertical

For small deviations from the local vertical, it can be shown that the torque depends on the orbital rate (orbital radius), the difference in MMOI about its principal axis and the angle with which the vehicle deviates from the vehicle as given in Equation [109]:

$$T \cong 3 \cdot n^2 \begin{bmatrix} (I_{zz} - I_{yy}) \cdot \phi \\ (I_{zz} - I_{xx}) \cdot \theta \\ 0 \end{bmatrix} \qquad [109]$$

Here n is mean motion as defined by [relation 4.17, SSE]:

$$n = \sqrt{\mu / a^3} \qquad [110]$$

With a is semi-major axis and μ is gravitational parameter of body about which the motion takes place.

Example: Gravity gradient torque estimation
*For S/C with a maximum difference in MMOI of 1000 kg-m^2 and an orbital rate (mean motion) of 0.0011 radians/second we find (1 degree angle): T= 3 * (0.001)^2 * 1000 * π/180 = 5.2 * 10^{-5} Nm.*

Aerodynamic torque ("weathervane" effect)
Torque induced by unbalance in aerodynamic pressure on different sides (relative to CoM) of spacecraft.

$$\underline{T} = \underline{r} \times \underline{F}_a \qquad [111]$$

$$F_a = \frac{1}{2} \cdot \rho \cdot V^2 \cdot S \cdot C_D \qquad [112]$$

- Vector r is the vector from body C_M to aerodynamic C_P
- Vector F_a is the aerodynamic drag vector in body coordinates

Solar radiation torque
Torque induced by unbalance in solar radiation pressure on different sides (relative to CoM) of spacecraft.

$$\underline{T} = \underline{r} \times \underline{F}_s \qquad [113]$$

$$F_s = (1 + K) \cdot P_s \cdot S \qquad [114]$$

- Vector r is the vector from body C_m to optical center of pressure
- Vector F_s is the solar radiation pressure in body frame coordinates

Magnetic torque

$$\underline{T} = \underline{M} \times \underline{B} \qquad [115]$$

- Vector T is the magnetic torque which is typically 3 x 10^{-6} Nm
- Vector M is spacecraft residual dipole in A-m^2 and has a typical value of 0.1 A m^2
- Vector B is magnetic field vector in spacecraft coordinates in TESLA. A typical value for the Earth magnetic field at about 200 km altitude is 3 x 10^{-5} TESLA. For other planets, see [SSE; Table 2.7]. Sometimes Gauss is used as unit for the magnetic field strength: 1 Gauss is x 10^{-4} TESLA.

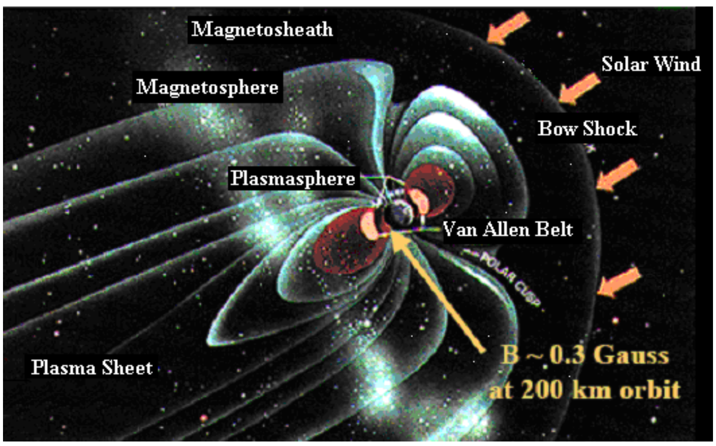


Figure 64: Artist view of Earth's magnetosphere

Internal torques (due to e.g. thrust misalignment and or operating machinery, etc.)

Mass expulsion torque (see also [SSE, section 9.4.6]:

$$\underline{T} = \underline{r} \times \underline{F} \qquad [116]$$

Notes:
- May be deliberate (Jets, Gas venting) or accidental (Leaks)
- Wide Range of r, F possible; torques can dominate others
- Also due to jettisoning of parts (covers, canisters)

Moving parts due to for instance antenna, solar array, scanner motion or to deployable booms and appendages:
- Momentum exchange between moving parts has no effect on System H, but will affect attitude control loops

Types of attitude control
The attitude control of a spacecraft can be considered being either actively controlled (meaning that a controller calculates necessary control torques and acting on the satellite to adjust its attitude to a desired position) or passively controlled (meaning that the satellite uses external torques that occurs due to its interaction with the environment and thus they cannot be avoided, in this case the disturbances being used for forcing the attitude of the satellite).
Another distinction is after the type of stabilization technique used. Four important types are:

- 3-Axis Stabilization: With three-axis stabilization, satellites have small spinning wheels, called reaction wheels or momentum wheels that rotate so as to keep the satellite in the desired orientation in relation to the Earth and the Sun. If satellite sensors detect that the satellite is moving away from the proper orientation, the spinning wheels speed up or slow down to return the satellite to its correct position. Some spacecraft may also use small propulsion-system thrusters to continually nudge the spacecraft back and forth to keep it within a range of allowed positions. Voyagers 1 and 2 stay in position using 3-axis stabilization. An advantage of 3-axis stabilization is that optical instruments and antennas can point at desired targets without having to perform "despin" maneuvers.

- Gravity gradient stabilization. The principle is to use the Earth's gravitational field to keep the spacecraft aligned in the desired orientation. The gravity of the Earth decreases according the inverse square law and by extending the long axis perpendicular to the orbit, the "lower" part of the orbiting structure will be more attracted to the Earth. The effect is that the satellite will tend to align its axis of minimum moment of inertia vertically (like a floater for fishing).

- Magnetic stabilization: The principle is to use the Earth's magnetic field and uncontrolled magnets to obtain some means of stabilization (like a compass needle always pointing to the local North). It allows for low resolution attitude control in an Earth orbit.

- Spin Stabilization: With spin stabilization, the entire spacecraft rotates around its own vertical axis, spinning like a top. This keeps the spacecraft's orientation in space under control. The advantage of spin stabilization is that it is a very simple way to keep the spacecraft pointed in a certain direction. The spinning spacecraft resists perturbing forces, which tend to be small in space, just like a gyroscope. Spin-stabilized satellites most often have a cylinder shape and rotate at one revolution every second. A disadvantage to this type of stabilization is that the satellite cannot use large solar arrays to obtain power from the Sun. Another disadvantage of spin stabilization is that the instruments or antennas also must perform "despin" manoeuvres so that antennas or optical instruments point at their desired targets. Spin stabilization was used for NASA's Pioneer 10 and 11 spacecraft, the Lunar Prospector, and the Galileo Jupiter Orbiter.

The latter three are all considered passive means of stabilization, whereas the first one necessitates the use of active systems, but also is considered to allow for most accurate control. Some typical control accuracies are:

- Gravity gradient control, or magnetic field control: Coarse control (> 5°)
- Spin stabilization:
    - Low accuracy pointing: 1 - 5°
    - Fine pointing: > 0.1 -1°
- 3-axis control:
    - Medium accuracy pointing: > 0.1 -1°
    - High accuracy pointing: < 0.1°

How well the attitude is controlled affects amongst others the design of the EPS, and the communications subsystem of the spacecraft, but also the other systems are affected.

System elements

The ADCS system consists of elements that allow for attitude determination (sensors) and for attitude control (actuators). Below these elements are discussed in some detail.

<u>Attitude sensors</u>

Various sensors exist that allow for attitude determination. We mention:

- Sun sensor: A device that senses the direction to the Sun. This can be as simple as some solar cells and shades, or as complex as a steerable telescope, depending on mission requirements.
- Earth (horizon) sensor: An optical instrument that detects light from the 'limb' (the circular outer edge) of the Earth's atmosphere, i.e., at the horizon. It can be a scanning or a staring instrument. Infrared is often used, which can function even on the dark side of the Earth. It provides orientation with respect to the earth about two orthogonal axes.
- Star sensor: An optical device measuring the direction to one or more stars, using a photocell or camera to observe the star. There are 57 bright navigational stars in common use. One of the most used is Sirius (the brightest). However, for more complex missions entire star-field databases are used to identify orientation. Star trackers, which require high sensitivity, may become confused by sunlight reflected from the exhaust gases emitted by thrusters.
- Magnetometer: An instrument used to measure the strength and/or direction of the Earth magnetic field. Using detailed information about Earth’s magnetic field at a given location it is possible to determine the attitude of the spacecraft. Magnetometers are usually mounted far away from the spacecraft body, for instance at both ends of the solar panel assemblies to isolate them from the spacecraft's magnetic fields.
- Rate gyro: A device used to detect and measure angular rates of change. When a rotation occurs, the momentum stored in the gyrating elements causes an out-of-plane bending force (called Coriolis force) that is representative for the rotation rate.

The working principles of these sensors are discussed in some detail in SSE, section 9.5. Typical sensor performances and some limitations to their use are given in the next table taken from [SMAD].

Table 37: Typical sensor performances [SMAD]

Sensor	Typical Performance Range	Mass (kg)	Power (W)	Characteristics and Applicability
Horizon sensors Scanner Fixed Head (static)	 0.1° to 1.0° < 0.1° to 0.25°	 2 to 5 2.5 to 3.5	 5 to 10 0.3 to 5	Horizon uncertainties Typically operates in IR Scanners: Wide field of view Can accommodate Fixed Heads: Single altitude, single No moving parts
Sun sensors	0.005° to 0.3°	0.5 to 2	0.1 to 3	Field of view up to 120°
Star sensors (scanners & mappers)	1 arc sec to 1 arc min	2 to 7	5 to 20	Typical field of view ±6°
Gyros (mechanical)	Drift = 0.03°/hr to 1°/hr	2	10	Normal use involves periodically resetting the reference position.
Magnetometer	0.5° to 3°	0.6 to 1.2	<<1	Attitude measured relative to Earth's local magnetic field. Magnetic field uncertainties and variability dominate accuracy. Usable only below ~6.000 km.
GPS	~0.1°	~5	~15	Requires one receiver and multiple antennas.

<u>Attitude control actuators</u>

Typical actuators include:

- Reaction wheels, see also [SSE, section 9.4.7]: Reaction Wheels are the most common actuators currently used in space. Reaction wheels are devices consisting of a wheel which rotates about a fixed axis with a built in motor. By speeding up or slowing down the wheel the ADCS is able to produce a torque about the axis of rotation of that wheel and so cause the spacecraft to rotate about that axis. As such, they belong to the class of momentum storage torquers, see SSE, section 9.4.7. In principle 3 wheels (for each axis 1) are needed to allow for full 3-axis control. Most assemblies though comprise four reaction wheels in a skewed configuration, which provides for 1 wheel to act as back-up for any of the other three wheels. Reaction wheels will eventually reach an rpm limit (~3000-6000 rpm) at which time they must be de-saturated. The torque delivered by a reaction wheel can be determined using [106], whereas the total momentum stored in the wheel is given by [105]
- Magnetorquers: Magnetorquers are essentially (electro-) magnets that can be used for attitude control and/or to de-saturate reaction wheels. As control actuators they allow for attitude control accuracy of the order of a few degrees. The torque produced is given by:

$$\underline{T}_m = \underline{a}NIA \times \underline{B} = \underline{D} \times \underline{B} \qquad [117]$$

  With a is a unit vector along the axis of the torquer, N is number of loops in the coil, A is area enclosed by a single loop, I is current in the coil, B is Earth's magnetic flux density (see ae1110-II) and D is dipole moment.
- Thrusters: Thrusters can be used to control attitude but at the cost of consuming fuel or rather propellants[16], see SSE, section 9.4. Nowadays thrusters are used to perform attitude changes that cannot be accomplished using the reaction wheels and/or to de-saturate the wheels if magnetorquers cannot be applied or are insufficient. The torque produced by a thruster pair is given by:

[16] Since in space there is no oxidizer available that can react with the fuel carried within, a S/C has to carry with it its own oxidizer. The combination of fuel and oxidizer generally is referred to as propellant.

$$\underline{T}_{thrust} = 2\,\underline{F}_T \cdot \underline{L} \qquad [118]$$

Here F is thrust force of a single thruster and L is thrust arm. A pair of thrusters is used thereby allowing for the thrusters to work in opposite directions to produce a torque, but while preventing the existence of a net force on the spacecraft.

Example: Reaction wheel rotational rate
To counteract any disturbance torque we need to apply a control torque. Suppose we use a reaction wheel producing a torque of 10^{-4} Nm, we find that the total angular momentum that must be stored by this actuator over a one day period is 10^{-4} x 3600 x 24 = 8.64 Nms. When using a wheel with a MMOI of 0.2 kg-m^2, it follows a wheel rotational velocity of 2500 deg/s or 600 rpm.

Example: System sizing
A spacecraft experiences a constant disturbance torque of 1 mNm about one of its principal axis. Given is an MMOI about this axis of 3145 kgm^2. This leads to an angular acceleration of 3.2 x 10^{-4} mrad/s^2 unless we counteract the disturbance torque.
Suppose that to counteract the disturbance torque, we use a pair of thrusters each with a distance of 2.000 m to the CoM of the spacecraft. The required magnitude of the thruster force (F) perpendicular to the moment arm (r) follows from:

$$T = 2Fr = T_{disturbance}$$

We find a thrust of 0.025 mN or 25 μN. However, this low a thrust level is almost impossible to realize with current existing thrusters. One way out is to use thrusters with a higher thrust level and then only thrusting over small periods of time.
For instance, in case of thrusting over 10% of the time, the thrust level increases with a factor 10 and becomes 0.25 mN (still very small, but more realistic). Suppose now that we allow for the vehicle a pointing error of +/- 1 deg. It follows for the time it takes for the vehicle to rotate over 2 deg = ~0.035 rad (twice the pointing error) under the influence of the disturbance torque: t = 14.8 seconds. So every 14.8 seconds the pointing error needs to be reduced. Given this short duration it is better that we use reaction wheels in case of thrusters. Still for now, we continue using thrusters.

In case the thrusters work in bursts of duration 100 ms, hence every 14.8 s two thrusters (in case of pure control) need to be activated for 0.1 second, we find that the required thrust level is given by 14.8 s x 1 mNm = 2 x F x 2.000 m x 100 ms => F = 0.037 N or 37 mN. Assuming that the thrusters are canted under an angle of 30 degrees to avoid jet impingement on any of the spacecraft surfaces, we need to install a thrust level of 64 mN per thruster.

For a mission duration of 10 years, we find that the thrusters should be capable of 10 x 365 day x 24 hr x 3600s/14.8 s = 21.3 million thrust cycles with a total operation time of 2.13 million seconds or about 592 hours. In case we use thrusters with an effective exhaust velocity of 2000 m/s, we obtain a mass flow rate per thruster of 32 mg/s or a total propellant consumption of 136.3 kg. RCS dry mass is estimated using relation [100]. I t follows an RCS dry mass of 0.178 (136.3) + 7.69 = 32.0 kg.

Consider now replacing the thrusters by reaction wheels (RW). Given that we aim for the RW to be de-saturated not more than once a day, we have to select a wheel capable of storing 1 mNm x 24 x 3600 sec = 86.4 Nms. Using relation [119], we find that a single RW capable of storing 86.4 Nms has a mass of 11.7 kg. Given that the RW still has to be de-saturated and that we select thrusters for de-saturation, we find that the mass of the RW has to be added to the RCS mass of 136.3 kg + 32.0 kg. So in this case it would be nice if thrusters could do the job by themselves.

The above case with the RW would be quite different if the disturbance torque is not constant, but cyclic where during part of the cycle the disturbance torque works in a positive direction and part of the cycle in a negative direction. In that case the RW will accelerate during part of the cycle and decelerate during the other part. Over the whole cycle, the net effect on the RW in that case is zero and hence no de-saturation maneuvers have to be planned and the whole RCS may be skipped.

ADCS software

Next to hardware elements, the ADCS system also includes a software component. The ADCS software contains algorithms[17] related to sensor and/or actuator data processing, attitude determination, attitude control needed, and the generation of attitude commands. It may reside on the onboard computer although sometimes also a dedicated computer is used of course linked to the onboard computer. Typical ADCS software elements are shown in Figure 65 next to some elements that are associated with position determination and orbit calculation and orbit prediction (see section on navigation).

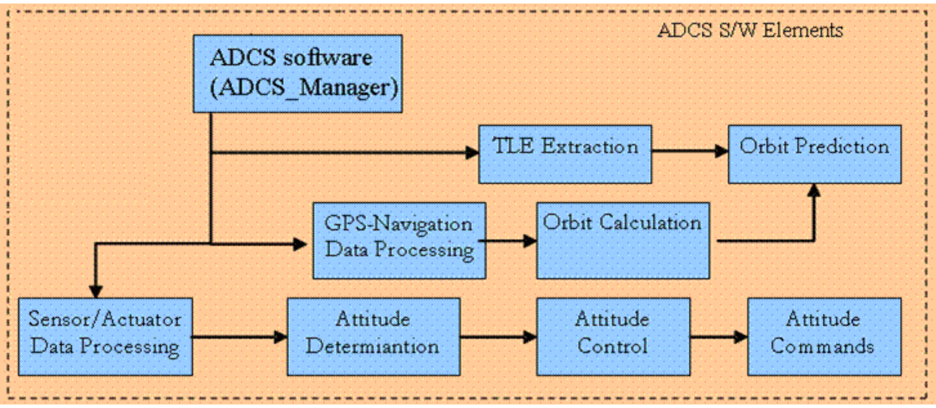


Figure 65: AOCS software (TLE= Two Line Elements)

Drawing the system

Figure 66 shows a representation of a typical ADCS. This kind of diagram is referred to as a block diagram. Advantage is that this diagram is very simple to be made, while it still provides a good overview of the various elements making up the system and their numbers. This particular diagram shows an ADCS consisting of a number of sensors as well as two types of actuators. Using the sensors we can determine the direction of Earth and the Sun. Given the time of the year it is possible to compute the attitude of the spacecraft. The gyros provide detailed information on changes in angular orientation. The 4 wheels (1 back-up wheel) allow for attitude control with the magnetorquers (1 for each axis) allowing for wheel desaturation.

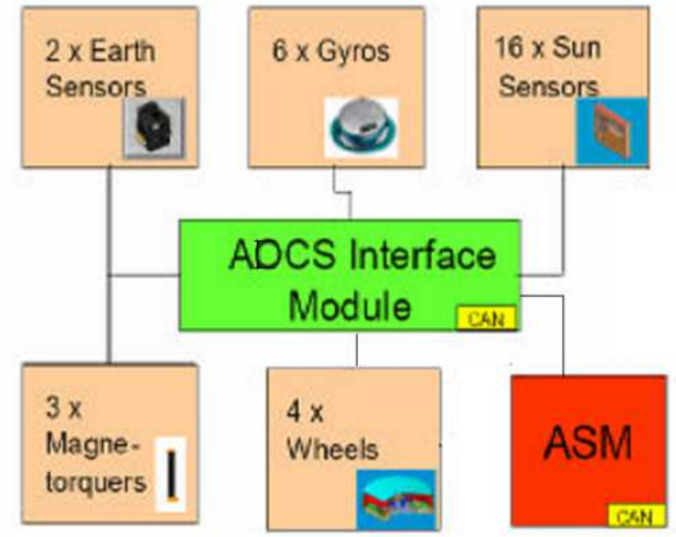


Figure 66: ADCS block diagram (ASM = Attitude Safety Module)

[17] An algorithm is a finite sequence of instructions, an explicit, step-by-step procedure for solving a problem.

The Figure 67 gives a much more detailed view of an ADCS. It shows:
- a cold gas propulsion system (for attitude control (10 mN) and orbit change (40 mN) maneuvers)
- a set of three magnetic torque rods (for attitude control in support of the cold gas system)
- two star cameras to providing the inertial attitude (attitude relative to some inertial frame)
- interfaces to a GPS receiver (to provide on-board orbital position)
- a course Earth -Sun sensor to provide attitude measurements with respect to Earth and Sun
- a three-axis Inertial Reference Unit used to measure angular rates
- a three-axis magnetometer mounted in the S-band antenna boom

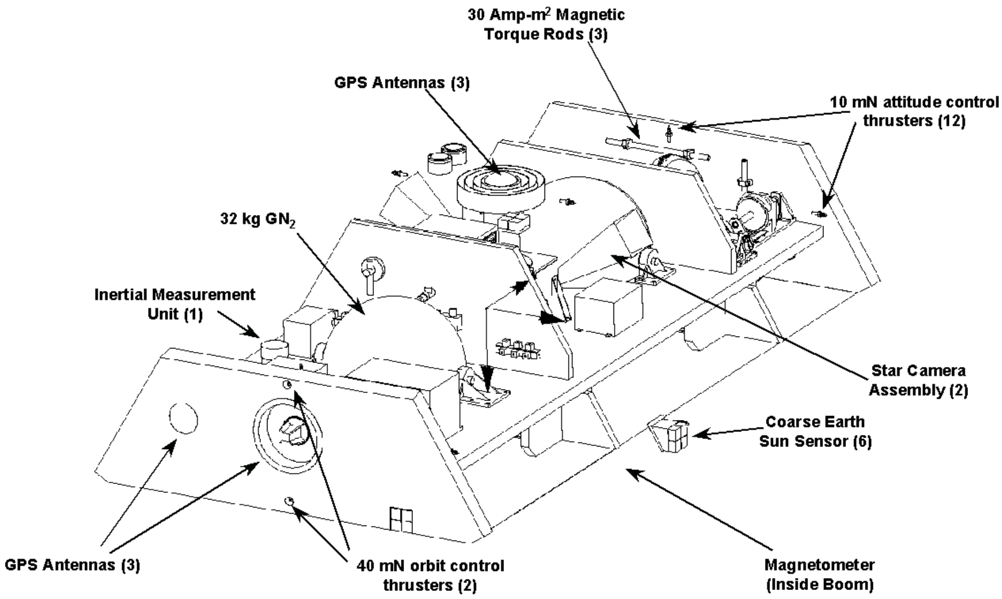


Figure 67: ADCS configuration of GRACE satellite

Configuration issues

- Optical sensors like Sun and Earth/horizon sensors and star cameras need to have an unobstructed view. For this reason they are usually mounted on the outer rim of the spacecraft
- Optical sensors provide a direction in space. To allow for full attitude determination 2 sensors are at least needed.
- Coarse sensors are mostly required to allow for initial acquisition (tumbling phase). Once the spacecraft is de-tumbled and has attained nominal attitude more accurate sensors take over
- Three-axis stabilized S/C: We need actuators that allow rotation about all three axis. S/C shape can be any, but aim is to have low MMOI.
- Spin stabilized: Usually cylindrical spacecraft body with axis of symmetry being the spin axis. Only few appendages
- Gravity gradient stabilized vehicle is usually a long vehicle (vertically aligned)

Dimensioning and sizing

In this section some data and/or estimation relationships are given that allow for estimating sensor and actuator mass of ADCS systems.

Sensors

Sensor sizing data are given in the next table.

Table 38: Generic characteristics of attitude sensors

Sensor Type	**Mass (kg)**	**Power (W)**	**Accuracy**
Sun	0.2 - 1.0	0-0.2	0.1 deg
Star	1-5	2-10	0.01 deg
Earth (horizon)	2 - 3.5	2-10	0.05 deg
Magnetometer	0.2 - 1.5	0.2 - 10	1 deg
Gyroscope	0.8 - 3.5	5-20	0.001 deg/hr
Accelerometer	0.1 - 1.0	0-1	4 g
IMU	3-25	10 -200	35/sec, 6 g
GPS	1	9	5m

Actuators

Actuator mass (in kg) can be used using the following simple scaling rules

Actuator type	**Estimating relationship**	
Reaction wheels	$M_{rw} = 1.7881 \cdot H_{rw}^{0.422}$; $R^2 = 0.9277$ 44 data points; H in range $10^{-4} – 10^{3}$ Nms	[119]
Torque rods	$\mathrm{M_{tr}} = 0.0167 \cdot D + 0.4876$; $\mathrm{R}^2 = 0.9595$ 10 data points; D in range 1 – 800 $\mathrm{Am^2}$	[120]

With H in Nms and D in $\mathrm{Am^2}$.

For mass of thrusters, see mass estimation of RCS under propulsion

Problems

A number of problems for exercising upon are available via Blackboard (Maple TA), whereas a few are also contained in a separate workbook available from the TU-Delft online print shop. Of this workbook also an electronic copy is available on blackboard.

4.6 Command and Data Handling (C&DH) system

Why command and data handling?
The C&DH system essentially makes up the brains/intellect and the nerve system of the spacecraft. Its main functions are to:

- Handle sensory information from both the payload (payload data) as well as from the internal systems of the satellite (housekeeping data; HK data). The latter provides info on health, status, internal environment of the spacecraft and may include such data as:
  - Temperatures (of equipment boxes, solar arrays, thrusters, etc.)
  - Pressures (in fuel tanks, plenum chambers, gas tanks, etc.)
  - Voltages and currents (of equipment power supplies)
  - Operating status of equipment
  - Other.
- Perform decision making (authorizes or generates commands)
- Command action. Typical commands include:
  - Switching instruments/devices on/off, like power switching and ordnance[18] control
  - Set some parameter to some value (like setting the volume on your MP3 player)
- Track time
- Act as memory

> *Example: Typical C&DH functions*
> *The Near Earth Asteroid Rendezvous (NEAR) spacecraft's command and data handling system is designed to manage complex operations and to collect data when the spacecraft is out of contact with ground control. During ground contacts, the C&DH system accepts uplink commands and memory loads that describe a time ordered set of events to follow, and it transmits previously recorded data back to the ground station.*

Some fundamentals
Signals used for telemetering can be either of an analogue or a digital form. An analog signal is a continuous signal which varies in amplitude, phase, or some other property in proportion to that of a variable. A digital signal is a discontinuous signal that changes from one state to another in discrete steps. In current command and data handling systems all the data are at some point digitized, just like commands. This process is referred to as **digitization**. Digital information exists as one of two digits, either 0 or 1. These are known as bits (a contraction of *binary digits*) and the sequences of 0s and 1s that constitute information are called bytes. The size of a byte is typically hardware dependent, but the modern *de facto standard* is 8 bits.

To digitize an analogue signal we may use:

$$DR_{analogue} = f_c \cdot N_{Nyq} \cdot n_{bits} \quad ; \quad f_s = f_c \cdot N_{Nyq} \qquad [121]$$

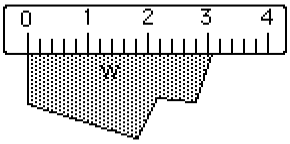
To digitize images, the following relation can be used:

$$DR_{image} = N_{images} \cdot S_{pixel} \cdot n_{bits} \qquad [122]$$

Here DR is data rate (for instance in bps, kbps or Mbps), f_c is highest frequency in analogue signal of interest, N_{Nyq} is number (usually taken equal to 2.2) taken from Nyquist-Shannon (or shortly Nyquist) criterion for sampling, f_s is sampling frequency, N_{images} is the number of images digitized per second,

[18] The term ordnance in this context refers to devices containing some explosive materials, like explosive bolts, pyrotechnic valves and igniters.

S_{pixel} is number of samples per picture or number of picture elements (pixels), and n_{bits} is number of bits per sample or per pixel, ranging from 2 up to about 16. The latter is important for the error made in the digitization of the information, i.e. the quantization error. To explain the importance of the number of bits with respect to this quantization error, we start by introducing a simple ruler used to measure the width w of some object, see figure.



Suppose the ruler uses cm as units with the smallest scale division being 2 mm. It follows w is a bit more than three cm but not exactly 3.2 cm. As a general rule of thumb the uncertainty of the measurement is taken one half the smallest scale division. This then gives an uncertainty of 1 mm. This uncertainty is indicated as the maximum (absolute) error of the measurement.

Now suppose we use 2 bits to represent our width measurement. 2 bits means that we have $2^{nbits} = 2^2 =$ 4 different states (0,0), (1,0), (0,1), (1,1) to represent the full scale of our ruler (4 cm). The 4 bits allows us to divide the scale of 4cm into 4 equal pieces of 1 cm each. Now taking (0,0) as 0.5 cm and (1,0) as 1.5, (0,1) as 2.5 and (1,1) as 3.5, we find that the largest error in our measurement is 0.5 cm. On the full scale, this means we have a (relative) error of 0.5/4 = 12.5%. With 3 bits (8 different states), this reduces to 6.25% and with 8 bits (256 states) we end up with a (quantization) error of less than 0.2%.

More in general, it follows for the quantization error (e_q):

$$e_q(\text{in }\%) = \frac{1}{2^{n_{bits}+1}} \cdot 100 \qquad [123]$$

Example: Data rate estimation
Suppose we have a camera on board of our spacecraft that is taking images from Earth. Given is that each image consists of 20,000 pixels (100 x 200). In addition it is given that the signal received for each pixel is digitized using a quantization error of less than 0.1%. It follows that each image consists of 20,000 pixels x 9 bits = 180000 bits. Suppose that our spacecraft is orbiting Earth at an altitude of 500 km, it follows an orbital velocity of 7.6 km/s(see syllabus, appendix H), which gives a ground velocity of about 7.05 km/s. Now assuming that all pictures should fit exactly (no space in between two images) and considering that along track we view 10 km, it means we should take 7.05/10 = 0.705 images per second. Total DR now becomes 0.705 1/s x 180 kbit = 126.9 kbps.

Key characteristics
Key characteristics of the C&DH system include:

- Payload data rate (varies widely depending on the payload). Typical values can be in the range from a few bits per second (bps) to several megabits per second (Mbps). Table 39 provides some typical signal data rates. Multiple such signals can be transmitted up and down to/from the satellite. Payload data rate also may vary during the mission. For instance for the SNAP vehicle the science acquisition data rate is 90 Mbps peak, whereas average acquisition data rate is only 45 Mbps.

Table 39: Typical signal data rates (without data compression)

Content	Digital	Analog
Text	50-80 bps	N.A.
Fax	16 kbps	N.A.
Voice	64 kbps	3.4 kHz
FM radio quality	1.024 Mbps	15 kHz
Music CD quality	1.4 Mbps	20 kHz
VHS-video	20 Mbps	N.A.
Video TV	130-166 Mbps	4-6 MHz
HDTV	400-1500 Mbps	18-27 MHz
Broadband data via internet	166 Mbps	N.A.

Example: Bit rate estimation
FM radio quality covers a frequency range of 15 kHz, see table. Using the Nyquist-Shannon criterion, and assuming that the signal amplitude is measured with a 32 bits representation, we find that in digital form, we need a data rate of 2.2 x 15 kHz x 32 bits = 1056 Mbps. Real value may differ slightly as the number of bits per sample may be chosen differently.

- Housekeeping (HK) data rate (depends on number of points monitored/measured, and measurement accuracy, range and frequency)
    - Simple TM systems are characterized by maximum 200 TM points, whereas highly complex systems are characterized by more than 1000 TM points. The next table provides an overview of the number of points telemetered for some specific satellites

Table 40: Number of TM points for several spacecraft

Spacecraft	# of TM points
Delfi C3	114 (estimate)
Delfi n3Xt	135 (current best estimate)
Intelsat 5	520
Eutelsat II	840
SPOT	~500
MSX	~400
ERS	6600
Envisat	13700

    - Typical measurement frequency/per parameter is 1 Hz; reasonably accurate measurement requires 8 bits or more. The larger the number of bits, the lower the quantization error.
- Number of controlled devices and number of commands; a simple satellite handles less than 50 commands and switches about 200 devices on/off. A complex satellite has more than 50 commands & more than 500 channels. For instance, the Thermal Ion Dynamics Experiment for measuring the characteristics of Earth's plasma environment has 155 different commands to control its operation. Commands or command messages are nothing more than a set of instructions for performing a specific task, such as changing the orbit of the spacecraft or deploying the payload. Command messages can come from:
    - a ground station via an uplink
    - the on-board computer (OBC), and
    - a hardline test interface

For illustration, Standard-Commands-for-Programmable-Instruments (SCPI) can be found in the SCPI standard, see http://www.ivifoundation.org/docs/scpi-99.pdf.
Specific commands for a DC power supply unit can be found in:
http://www.ivifoundation.org/downloads/Class%20Specifications/IVI-4.4_DCPwr_2010-06-09.pdf

- Handling speed: A good measure for handling speed is the clock speed of the processor typically expressed in Mega Instructions Per Second (MIPS). Typical radiation hardened (space) processors used include 80386, 80C86, NSCC-1, RAD6000, Mongoose V, and LEON[19].. These processors have handling speeds in the range from less than 1 MIPS up to 100 MIPS.

- Memory size (in MByte or GByte; 1 Byte = 8 bits). The memory size can be estimated based on volume of data produced over the fraction of the time between two memory dumps to ground that data is produced:

$$V_{produced} = DR_{produced} \cdot t_{dump\ fraction} \cdot \qquad [124]$$

  For a polar satellite, the time between two memory dumps could be of the order of about 12 hours in case only a single ground station is used. In case of multiple ground stations this time may be greatly reduced..

An example telemetry data table is given in Table 41. In total 56 TM points are shown. For each TM point are given a number, an identifier, a numerical value and the units of measurement.

Table 41: Typical telemetry data table

Uptime is 226/16:52:39. Time is 1/23/1995 06:56:20
Telemetry data is:

0	Rx E/F Audio(W):	2.140	V(p-p)	1	Rx E/F Audio(N):	2.165	V(p-p)
2	Mixer Bias V:	1.346	Volts	3	Osc. Bias V:	0.510	Volts
4	Rx A Audio (W):	2.140	V(p-p)	5	Rx A Audio (N):	2.140	V(p-p)
6	Rx A DISC:	0.411	kHz	7	Rx A S meter:	86.000	Counts
8	Rx E/F DISC:	-0.882	kHz	9	Rx E/F S meter:	116.000	Counts
10	+5 Volt Bus:	4.880	Volts	11	+5V Rx Current:	0.023	Amps
12	+2.5V VREF:	2.495	Volts	13	8.5V BUS:	8.367	Volts
14	IR Detector:	1.000	Counts	15	LO Monitor I:	0.001	Amps
16	+10V Bus:	10.657	Volts	17	GASFET Bias I:	0.004	Amps
18	Ground REF:	0.000	Volts	19	+Z Array V:	0.205	Volts
20	Rx Temp:	1.814	Deg.C	21	+X (RX) temp:	13.916	Deg.C
22	Bat 1 V:	1.302	Volts	23	Bat 2 V:	1.314	Volts
24	Bat 3 V:	1.304	Volts	25	Bat 4 V:	1.297	Volts
26	Bat 5 V:	1.319	Volts	27	Bat 6 V:	1.315	Volts
28	Bat 7 V:	1.313	Volts	29	Bat 8 V:	1.303	Volts
30	Array V:	10.085	Volts	31	+5V Bus:	4.802	Volts
32	+8.5V Bus:	7.998	Volts	33	+10V Bus:	11.147	Volts
34	BCR Set Point:	20.213	Counts	35	BCR Load Cur:	0.094	Amps
36	+8.5V Bus Cur:	0.027	Amps	37	+5V Bus Cur:	0.251	Amps
38	-X Array Cur:	-0.011	Amps	39	+X Array Cur:	-0.011	Amps
40	-Y Array Cur:	-0.012	Amps	41	+Y Array Cur:	-0.011	Amps
42	-Z Array Cur:	-0.017	Amps	43	+Z Array Cur:	-0.011	Amps
44	Ext Power Cur:	-0.020	Amps	45	BCR Input Cur:	0.213	Amps
46	BCR Output Cur:	-0.017	Amps	47	Bat 1 Temp:	8.470	Deg.C
48	Bat 2 Temp:	-18.760	Deg.C	49	Baseplt Temp:	7.260	Deg.C
50	FM TX#1 RF OUT:	0.026	Watts	51	FM TX#2 RF OUT:	-0.003	Watts
52	PSK TX HPA Temp:	-13.919	Deg.C	53	+Y Array Temp:	4.234	Deg.C
54	RC PSK HPA Temp:	-0.002	Deg.C	55	RC PSK BP Temp:	1.209	Deg.C
56	+Z Array Temp:	-5.448	Deg.C				

[19] An overview of processors used in space can be obtained from http://www.cpushack.com/space-craft-cpu.html

Example problem: Memory size
Consider a spacecraft with a payload data rate of 8 kbps. For housekeeping the spacecraft is equipped with 400 sensors each producing 8 bits of information every second. Calculate for this spacecraft the total data rate produced per second and determine the storage space needed in case this data has to be stored on board of the spacecraft for the duration of 2 hours.

Solution:
400 sensors each producing 8 bits of information every second gives a HK data rate of 3200 bps. Now we add the payload data rate of 8 kbps (8000 bps), which gives us a total data rate of 11.2 kilobit (kbit) produced every second or 11.2 kbps.
The storage space needed in case the data produced is stored for the duration of 2 hours is equal to: 11.2 kbps x 2 hrs x 3600 sec/hr = 80.64 Megabit (Mbit) or 80.64/8 = 10.08 MByte (MB).

Note that in reality the data rate and also the storage space tend to be higher/larger as we also need to timestamp the signals and we may need to add an identifier telling us what the data is about.

Another important aspect is that the data in the onboard memory must as some point be transmitted to ground. Ground contact times can be as short as 6 minutes. This then requires high read out data rates for the onboard memory. It follows:

$$V_{transmit} = t_{transmit} \cdot DR_{readout} \quad [125]$$

Example: To transmit 10.08 MByte in 6 minutes, requires a read out data rate of 224 kbps.

The read out data rate will impose requirements on the communications subsystem as this system should be able to transmit the data. In addition, the read out data rate also imposes requirements on the processor in terms of number of MIPS. As a rule of thumb, 1 MIPS is about 1-4 Mbps [Gray].

How to determine ground contact time has been dealt with in a simple way in an earlier course (Introduction to Aerospace Engineering). The actual time for transmission down to ground may be even less as during ground contact time also commands have to be sent up and so on.

System elements
System elements include:
1. Processor + operating system + internal clock (on board computer or OBC); Sometimes we have different processors for command and telemetry.
2. A motherboard that provides the electrical connections by which the processor communicates with the memory as well as external units (like a printer, video screen, keyboard in a PC system)
3. Network card, modem, etc.
4. Memory or data storage unit (solid state memory, hard drive, floppy disk, tape recorder)
5. Software
6. The network (harness/wires + connectors)
7. Data acquisition or On-Board Data Handling (OBDH) unit
8. Control unit for e.g. power switching, firing of pyrotechnical charges, etc.

The first three elements usually make up the on-board computer. For satellites requiring limited storage capacity also the memory may be integrated into the computer, but for large data storage facilities, the data storage unit may be a separate item. The items 7 and 8 are sometimes combined into a single Data Acquisition and Control Unit (DACU).

Computers, command processors, control units, etc.
Figure 68 shows some typical hardware elements of the C&DH system.

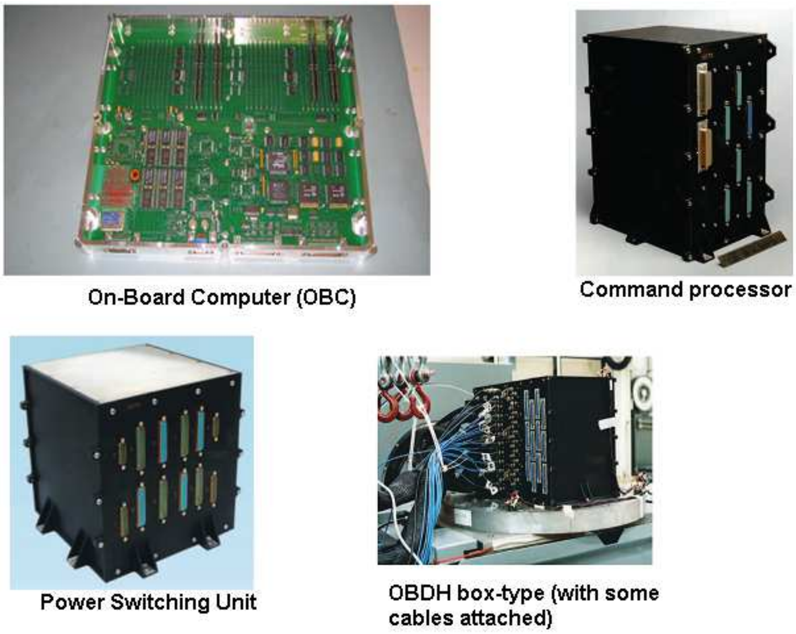


Figure 68: Examples of C&DH hardware

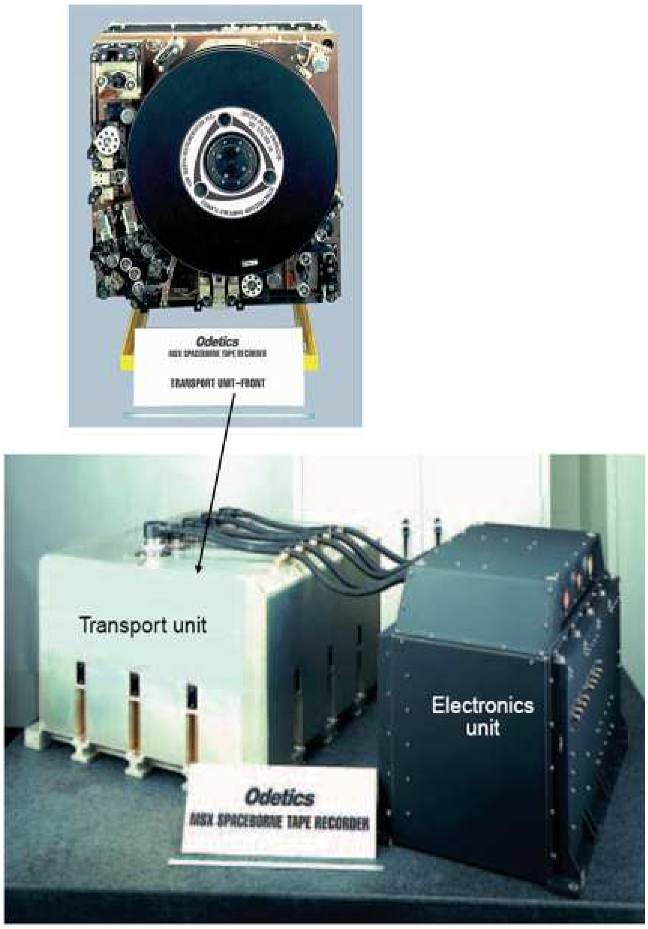
From this figure, we also learn that components are essentially box shaped (What is in the box is for now not interesting) and that extensive cabling is needed to connect all the boxes thereby allowing for information transfer. For the spacecraft designer it comes down to placing the boxes in the spacecraft, thereby taking into account:

- Mass
- Size of each box
- Centre of Mass (CoM)
- Mass Moment of Inertia (MMOI)
- Short power leads

Etc.

A word of caution taken from the work of [Manning]: Over the last few decades, application of current terrestrial computer technology in embedded spacecraft control systems has been expensive and wrought with many technical challenges. These challenges have centered on overcoming the extreme environmental constraints (protons, neutrons, gamma radiation, cosmic rays, temperature, vibration, etc.) that often preclude direct use of commercial off-the-shelf computer technology.

Data storage
For data storage, two main options exist, being tape recorder (TR), see Figure 69, and solid state recorder (SSR). Solid state recorder is currently the main choice as it is less sensitive to failure (no mechanical parts), allows for a lighter design for the same data storage and allows for random access.



Characteristics of the MSX spaceborne tape recorders.

Record speeds	18.8 cm/s and 94 cm/s
Playback speed	94 cm/s
Tape dimensions	2.54 cm × 2.2 km
Tape format	42-track IRIG standard
Power consumption	45 to 225 W, mode dependent
Mass, transport unit	57.6 kg
Mass, electronics unit	21.8 kg
Total mass	79.4 kg
Size, transport unit	50.8 × 50.8 × 33.0 cm
Size, electronics unit	30.5 × 50.8 × 38.1 cm
Number of parts	16,732

Recorder modes include:
- Recording
- Playback

Figure 69: Space-flight proven tape recorder (l) and its characteristics (r)

Harness

A cable harness, also known as a wire harness, cable assembly, wiring assembly or wiring loom, is a string of cables and/or wires which transmit informational signals or operating currents (energy). The cables are bound together by clamps, cable ties, cable lacing, sleeves, electrical tape, conduit, a weave of extruded string, or a combination thereof, see figure.

On board computer software

On board computer software is a term that refers to digitally stored computer programs that allow a.o. for interfacing with hardware so that they are able to perform specific tasks. Software allows for a more flexible approach to the scheduling of tasks other than say using timers as in the old days. In general, the larger the on board software is, the more tasks can be initiated and controlled by the on board computer and the higher the level of autonomy of the spacecraft becomes, thereby reducing ground station involvement. As such, it reduces Up-/Downlink usage and reduces operational costs on ground.

Source lines of code (SLOC) is a software metric used to measure the size of a software program by counting the number of lines in the text of the program's source code (an A4 page typically contains 30 SLOC). A complicating factor is that the number of SLOC may vary with the programming language used. Still, the SLOC is considered a reasonable metric for software size. The MSX spacecraft has in total 280 k (280,000) lines of code of which 70 k lines are for the attitude subsystem, 1.5 k lines for the power subsystem, 15 k lines for C&DH, 25 k lines for tracking/navigation with the remainder for the payloads written partly in C and partly in assembler.

Figure 70 gives values for some specific S/C, starting with some old S/C and finishing with some recent S/C. The figure clearly shows that early S/C have very few lines of code indicating low level of autonomy. For more recent S/C the software size has increased substantially, thereby allowing for much more autonomy of the S/C and hence reduced level of ground control. However, with increasing software size, also the number of software errors increases (typically 3-4 per 30 lines of code). To keep the number of errors under control, software needs to be scrutinized thoroughly taking a lot of time with the associated high cost.

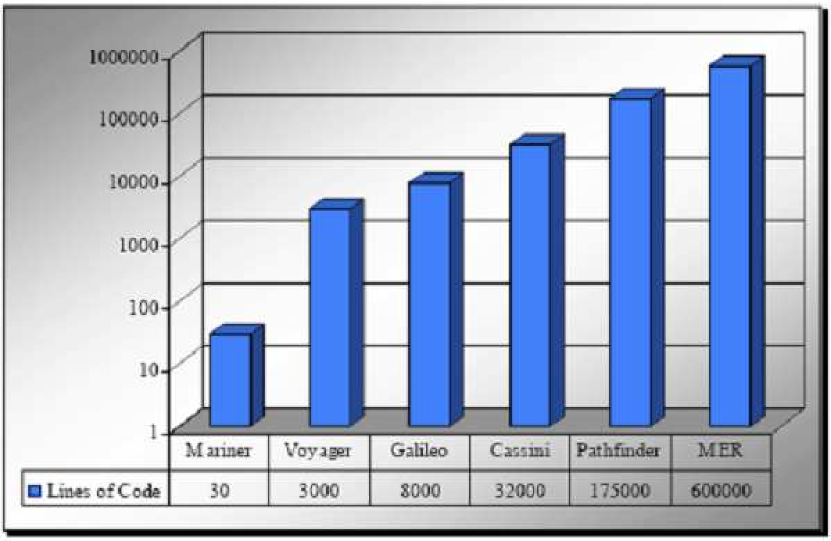


Figure 70: Size of software in S/C missions

To estimate the memory size to store the on board computer software we typically require 2-4 kB of memory size (depending on a.o. programming language used) for each 1000 lines of code (LOC).

Configuration

How the various C&DH elements relate to each other can be seen in Figure 71. The figure shows on top the various elements that provide measurement information to the central processor unit. This data is transferred to the computer, which processes the data, schedules actions based on the data received, stores the data and/or transfers the data to memory. Data is transferred to the computer via a data bus[20] (essentially a bunch of data information lines). Internal communication in the computer is via the computer bus. Via the In/Out (I/O) board the computer commands the various units including the payload units on/off or allows for changing the settings of these units. Via the housekeeping board (H/K board) the computer receives information about the health status of the satellite. The TM/TC (telemetry/telecommand) board the information is transferred to the TT&C system for transmission down to ground (TM) or command information (TC) is received from ground for on board processing. Important configuration issues include:

- S/C CoM: Preferably in geometric centre
- MMOI of S/C
- Short line length for data bus (to reduce mass)
- CPU needs to be cooled?
- Items must fit in the available space.

[20] In computer architecture, a **bus** is a subsystem that transfers data between computer components inside a computer or between computers and/or other devices external to the computer. An *internal bus* connects all the internal components of a computer to the motherboard. These types of buses are also referred to as a local bus, because they are intended to connect to local devices. An *external bus* connects external peripherals to the motherboard.
Buses can be parallel buses, which carry data words in parallel on multiple wires, or serial buses, which carry data in bit-serial form.

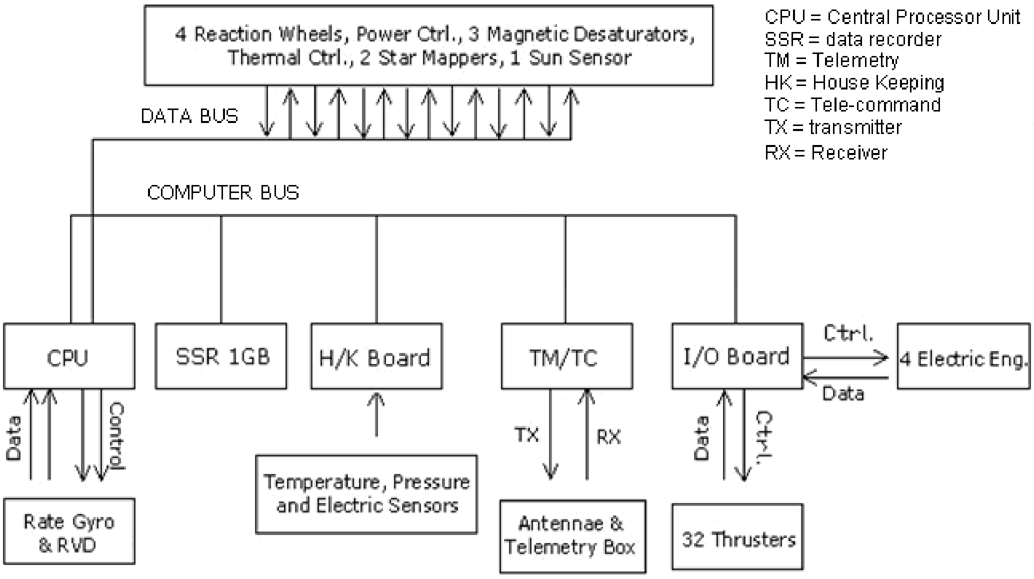


Figure 71: Typical C&DH set up (architecture)

Dimensioning and sizing

Here some simple mass estimation rules are given for typical elements making up a command and data handling system. Relations/data apply to single units only. In case of multiple units, the result must be multiplied by the number of units.

Data handling/acquisition system (excluding data storage)
For sizing of the data handling/acquisition system we can use:

> Size (mass) determined by number of TM channels: ~0.5-5 kg/100 channels [126]
> Mass density: ~0.5-1 kg/liter
> Specific power: ~1-2.5 W/kg

On board computers/TM encoders/TC decoders
For sizing of on board computers, TM encoder and TC decoders we may use:

> Size (mass) determined by MIPS[21] ~0.7 kg/MIPS [127]
> Mass density: 1.4 kg/l (SSD = 1.4 kg/l)
> Specific power:: 3.3 W/kg (SSD = 2.7 W/kg)

Here the onboard computer, TM encoder and TC decoder are considered as a single item although in some spacecraft designs they form separate units.

The above relations are very preliminary. Depending on their criticality in the total design of the spacecraft, it is advised to improve the estimations as soon as possible.

[21] The use of MIPS as the parameter determining the mass of the OBC/TM/TC unit is a bit questionable since the performance of the processors used in space is rapidly improving, from just a few MIPS in the 1990s to several hundreds of MIPS today. Also the power usage is rapidly decreasing. In case 1980's CMOS technology is used it is possible to reach 1 MIPS/W. However, for 1990s 32 bit RISC processors a value of 10 MIPS/W is attainable [Manning] and in the future several hundreds of MIPS/W seem feasible. This means an enormous reduction on the power required compared to 1980s CMOS technology.

Data storage devices

For mass estimation of data storage devices, two relations are available:

- Tape recorder mass (M_{rec}):

$$M_{rec}[kg] = 8.9 \cdot C_{rec}\ [GByte] + 11.31 \quad (0.1\text{ GByte} \leq C_{rec} \leq 6\text{ GByte}) \qquad [128]$$

- Solid state recorder:

$$M_{rec}[kg] = \frac{C_{rec}[GByte]}{.041 \cdot C_{rec}[GByte] + 0.3128} \quad (0.1\text{ GByte} \leq C_{rec} \leq 180\text{ GByte}) \qquad [129]$$

The latter relation has an R^2 value of 0.8873.

Harness

For spacecraft, the harness makes up a few % of the spacecraft mass. The harness mass depends greatly on number of signals transferred and/or the length of the cable harness. For spacecraft harness may be estimated by estimating the number of signals to be transferred times the length of the harness times the specific mass of the harness. The latter is estimated as 0.011 kg/m/signal (SSD = 0.003 kg/m/signal).

Problems

A number of problems for exercising upon are available via Blackboard (Maple TA), whereas a few are also contained in a separate workbook available from the TU-Delft online print shop. Of this workbook also an electronic copy is available on blackboard.

4.7 Telemetry, tracking and command

Why we need it!
The Telemetry, Tracking and Command (TT&C) system is needed to communicate status and commands, and to allow for tracking the spacecraft (to determine the satellite's position). 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 of the order of years, the satellite receives daily the commands required to reconfigure functions according to requirements at the time. Earth observation satellites, such as SPOT, Landsat, 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 TDRSS, 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 (HK) 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 transmit the payload and HK data. The latter enables the mission control center 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.

Definitions:

- Telemetry: The science and technology of automatic measurement and transmission of data by wire, radio, or other means from remote sources, as from space vehicles, to receiving stations for recording and analysis.
- Tele-command: The use of telecommunication for the transmission of signals to initiate, modify or terminate functions of equipment at a distance.
- Tracking: The act of measuring the direction and magnitude of spacecraft motion.

How it works
TT&C uses Radio Frequency (RF) transmission, i.e. electromagnetic (radio) waves, to transmit voice, data, image, radio or video via a carrier from a transmitter to a receiver. The receiver unit receives the modulated radio waves and converts them back into a signal. Hence the basic elements of any telecommunications system are a transmitter which transmits the signal and a receiver receiving the signal. In addition antennas may be used to provide direction to the signal. See later for more details.

Some fundamentals
The carrier signal used in radio-telecommunications is an electromagnetic wave of some frequency. Radio-frequency transmissions typically are in the wavelength range from a few KHz to approximately 40 GHz. As this is quite some range, this range is usually subdivided in a number of frequency bands. Hence a frequency band can be defined as a range of frequencies in the electromagnetic spectrum. Microwave frequency bands, as defined by the Radio Society of Great Britain (RSGB), are shown in Table 42.

Without going into detail, it is mentioned that the design of any communications equipment (for instance the antenna) greatly depends on the carrier frequency selected.

Table 42: Microwave bands

Band	Frequency range
VHF	30 to 300 MHz
UHF	300 MHz to 3 GHz
L band	1 to 2 GHz
S band	2 to 4 GHz
C band	4 to 8 GHz
X band	8 to 12 GHz
K_u band	12 to 18 GHz
K band	18 to 26.5 GHz
K_a band	26.5 to 40 GHz
Q band	30 to 50 GHz

Frequency is inversely proportional to wavelength, according to the equation given in Equation [130]. Here ν is the speed of the wave (c in a vacuum with c being equal to 300.000 km/s, or less in other media), f is the frequency and λ is the wavelength.

$$f = {}^{v}\!/_{\lambda} \qquad [130]$$

Radio-telecommunications typically spans a wavelength range from a few cm up to a few meter.

Exercise: Determine for the various frequency ranges in the above table the wavelength range.

Signals are transmitted using an analogue or a digital signal. An analog signal is a continuous signal which varies in amplitude, phase, or some other property in proportion to that of a variable. A digital signal is a discontinuous signal that changes from one state to another in discrete steps. Whether the signal is analog or digital, it needs to be modulated on to the carrier which than transports the signal to the receiving end. The amount of information carried is indicated by the bandwidth (in Hz of a multiple thereof), earlier defined in Chapter 2 of this syllabus; Typical bandwidths needed to transmit certain information are given in Table 39. Some additional elaborations though are needed to take into account multiple communication channels (for instance in case we have 100 voice channels, say 100 people communicating with each other at the same time) and/or to convert bits into Hz.

Example: Bandwidth analog transmission
To allow for the transmission of 100 voice channels simultaneously in an analog way, we need a bandwidth of 100 x 3.4 kHz = 340 kHz. In case we transmit this information in the L-band, we might have a carrier frequency of say 1.6 GHz (depends on ITU regulations) and a bandwidth of 340 kHz or about 0.34 MHz.

In case of digital transmissions, we first need to digitize and compress the information and then modulate the signal onto the carrier signal. The required bandwidth in that case is given by:

$$B = DR \times CF / \text{spectrum utilization} \qquad [131]$$

- Spectrum utilization: Measure for number of bits transmitted per unit of frequency (roughly between 0.2-2 bits per Hz); The higher this value the better this is for the bandwidth
- CF is compression factor ranging from about a factor 2 to 10 and more, depending on the amount of loss of information accepted
- DR = (uncompressed) data rate

Example (1): Bandwidth digital signal
To transmit n uncompressed digital signal of 1 Mbps using a spectrum utilization of 1, we find we need a bandwidth of 1 MHz. When taking into account a compression factor of 5, we find a signal data rate of 200 kbps. For identical spectrum utilization we obtain a bandwidth of 0.2 MHz.

Example (2): Bandwidth analog signal after digitization and compression
To transmit an analog signal of 340 kHz, we find a signal data rate of 744 kbps. Using a compression factor of 5, we find a signal data rate of 149.6 kbps. In case spectrum utilization is 1, this gives a bandwidth of 150 kHz, which is about a factor 2 lower than in case of analog transmission. For a spectrum utilization of 0.2, we find a bandwidth of 744 kHz, which is in excess of the bandwidth needed in case of an analog transmission.

Currently most transmissions are in digital form. It is essentially the compression factor that ensures that digital transmissions require less bandwidth then analog transmission.

Bandwidth is a scarce commodity and obtaining some bandwidth can be quite expensive. For instance in 2012 the Dutch government obtained 3.8 billion Euro for leasing bandwidth to various companies for fast mobile voice and internet at 3.8 billion Euro. Still there are also some frequency bands (for radio amateurs, etc.) that do not cost a lot except for the cost for a permit to use.

Data on how much bandwidth is available in a certain band can be obtained from the International Telecommunications Union (ITU), see later in more detail.

Example: Maximum data rate from given bandwidth
For instance, in case we have available a bandwidth of 10 MHZ at a carrier wavelength of 1600 MHz, this would allow for a data rate of maximum 10 Mbps given a spectrum utilization of 1. To transmit more data at this frequency is simply not possible/allowed.

Generally speaking we can say that the available bandwidth increases with increasing frequency.

The travel time (t) of a signal through space depends on the speed of the wave (c) and the distance (d) travelled:

$$t = d/c \qquad [132]$$

As in space distances are quite large, this may lead to long time periods in between transmission and receiving the signal. This can seriously hamper the proper working of the S/C.

Exercise: Consider the distance between Mars and Earth and determine the time it takes for a signal to travel the distance to Earth and back. Consider what will be the consequence in case of emergency measures/maneuvers.

Another important point is that to allow communication over some distance the transmitter and receiver need to “see” each other. For instance a spacecraft in low Earth orbit can only be seen from ground for a very short time, typically about 10-15 minutes. In an earlier course a simple method has been introduced allowing you to calculate the maximum contact time (in case of an overhead pass). During the contact time, the spacecraft data (payload data + TM data) are transferred to ground and commands are sent up to the spacecraft. This may lead to high data transmission rates and hence large bandwidth.

Example: Effect of ground contact time
Consider that we have a continuous image data rate generated of 8 Mbps over an orbit. Given an orbital period of 120 minutes, this leads to a total amount of data of 7.2 GByte (GB) per orbit. Given a ground contact time of 10 minutes, this leads to a transmission data rate of 8 x 12 = 96 Mbps or with a spectrum utilization of 0.5 a bandwidth needed of 192 MHz. In reality, the case might be even worse as during ground contact time also some time may be needed for the system to receive commands.

Some typical transmission rates are given in the next table.

Table 43: TT&C characteristic data [Sarsfield]

Mission	Spacecraft	Downlink Data Rate (kpbs)	Comm. Band	Transmitter Power (W)
Clementine		128	S-band	5.0
Discovery	NEAR	9	X-band	5.0
	Mars Pathfinder	11	X-band	13.0
Explorer	SMEX-SWAS	1,800	S-band	5.0
	SMEX-TRACE	2,250	S-band	5.0
	MIDEX-MAP	666	S-band	5.0
New Millennium	Deep Space 1	10	X-band	12.5
	Earth Observer 1	105,000	X-band	5.0
SSTI	Lewis	n/a	S-band	n/a
	Clark	2,500	X-band	18.0
Surveyor	Mars Global Surveyor	85	X/Ka-band	25.0
	Mars Surveyor '98—Lander	2	X-band	15.0
	Mars Surveyor '98—Orbiter	111	X-band	15.0
Baseline	RADCAL	19	C-band	10.0

The table also shows transmitter (output) power P. This is the actual amount of power of radio frequency (RF) energy that a transmitter produces at its output. Typically we tend to keep transmitter power as low as possible for the information to be received as to limit power usage of the communications system. Still we need to make sure that the signal has sufficient power to be received successfully and to allow distinguishing the signal from the background noise.

Example: A spacecraft communicating with ground (or another spacecraft) can be well compared with two people communicating.

For instance, we have one person talking and one listening. The person that talks uses his vocal cords and his mouth to produce sound waves of certain strength. These sound waves travel through the atmosphere and are picked up by the other person's ears, where the ear drums convert the sound waves into a signal that can be understood by our brains. If those two people are in a room full with others they might find that communication is more difficult due to the noise generated by the other people. To overcome this noise, the two people talking can reduce their distance or the person talking can talk more forcefully (put more power into the signal) or use his hands (like a megaphone) to direct his voice to the listener and thereby create some gain. The person listening might use his hands to guide the sound waves to his ears or use some hearing device.

Hereafter we will discuss the basic relations governing the operation of a telecommunications system. We will start at the transmitter end, where the communications signal is generated.

We start with the transmitter power. This is the signal power at the output of the transmitter. From the transmitter the signal is transferred to the antenna, where the signal is directed toward the horizon as a beam, thereby creating gain and increasing the radiated power in the beam direction. There is also some loss (negative gain) from the feed line, which reduces some of the power output to the antenna by both resistance and by radiating a small part of the signal. For the effective (isotropic) radiated power we have:

$$EIRP = P \cdot L_l \cdot G_t \qquad [133]$$

At the receiver end (given antenna size) the amount of power received reduces with the distance. This is given by the power flux density (W_f), i.e. the power per unit area at distance (r). Consider transmitter is point source (no antenna present, no losses):

$$W_f = P / 4\pi r^2 \qquad [134]$$

Antenna size: The larger the antenna, the more focused the beam and hence the higher the power density in the beam (G_t is gain factor of transmitting antenna)

$$W_f = P \cdot G_t / 4\pi r^2 = EIRP / 4\pi r^2 \qquad [135]$$

Gain factor is 1 when antenna transmits power equally in all directions. Gain factor increases when antenna transmits power within a (narrow) angle. Gain factors of 10000 and higher are feasible.

The power received (C) can now be determined using:

$$C = W_f \cdot A_{ant} \cdot \eta_{ant} \qquad [136]$$

A_{ant} = antenna frontal area
η_{ant} = antenna efficiency, typically 0.5-0.7

A certain amount of power in a signal is not the only criterion of importance. One should also consider the energy per bit (E_b):

$$E_b = C/R \qquad [137]$$

Here R is bit rate.

At the receiver end, the signal should be above the noise level or otherwise the signal cannot be read. To this end, we need to be able to estimate the noise level and compare this with the energy per bit level. In most designs, one aims for a signal level a factor 10 in excess of the noise level:

$$E_b / N_o \geq 10 \qquad [138]$$

Here N_o is noise spectral density.

To estimate the noise level and hence the noise spectral density, one first should realize that noise in a space telecommunications signals results from the presence of the atmosphere (atmospheric noise), solar radiation (solar noise), cosmic radiation (cosmic noise) as well as noise generated by the electronics themselves (thermal or Johnson noise). The amount of noise or the noise power (N) in a first approach can be determined using:

$$N = K\, T_s\, B = No\, B \qquad [139]$$

Here:

- k is Boltzmann's constant 1.380 x 10^{-23} J/K
- B is bandwidth (in Hz)
- T_s is system noise temperature (in K), depends on frequency

The calculation of the system noise temperature can be quite complicated and therefore is left for later courses and/or for self-study. The relation essentially indicates that noise level increases with temperature. This is one of the reasons why sometimes elements in the transmission system are cooled to reduce the noise level.

So far, we have discussed RF output power, but did not consider input power. Typically there is a difference between the two that is referred to as the transmission efficiency (η). Typical values of this efficiency are in the range 10-20%.

$$P_{in} = P_{out} / \eta \qquad [140]$$

Regulations

Today, radio-spectrum is quite in demand and its use is heavily regulated first and foremost by national authorities and by the ITU (International Telecommunications Union). The latter is a United Nations agency responsible for coordinating the shared global use of the radio-spectrum. It is headquartered in Geneva. All global use of the radio-spectrum should go through the ITU via the national authorities (for the Netherlands, this is the "Agentschap Telecom" of the ministry of economic affairs) and is subject to formal registration and approval by the ITU authorities.

To protect the scarce resources of allocated frequency spectra, space agencies have had to cooperate with each other and coordinate their efforts in order to present a unified position at the ITU. This is done, for instance, through committees such as the Space Frequencies Coordination Group (SFCG), in which ESA is an active member. This need for cooperation plus the high cost of developing and maintaining a ground network has led the major space agencies to achieve a high degree of compatibility for TT&C matters.

For transmission of command/telemetry data of spacecraft, the following general guidelines apply:

- Until 1970s, most satellites' TT&C performed through VHF links (130 MHz bands)
- Since early 1980s, most satellites use S-band (2 GHz) for their TT&C
- X-band (8 GHz) is used for some deep-space probes
- Future: Deep space missions will all use X-band. Some near-Earth missions could also use X-band.
- See [FSS, table 12.2] and/or the next table which provides the technical profile of a typical ESA S- or X-band ground station.

Table 44: Typical ESTRACK station technical profile (courtesy ESA)

Characteristic	**Typical range**
Antenna dish diameter	15m, 35m
Transmit frequency	
S-band	2025-2120 MHz
X-band	7145-7235 MHz
Receive frequency	
S-band	2200-2300 MHz
X-band	8400-8500 MHz
Telemetry (downlink)	
Normal data rate	up to 1 Mbps
Maximum data rate	up to 105 Mbps
Telecommand (up-link)	
Normal data rate	2 Kbps
Tracking	
Range accuracy	1 m
Range rate accuracy	0.1 mm/s

System elements
Key elements of a radio-communications system include:

- **Transmitter:** an electronic device which, usually with the aid of an antenna, propagates an electromagnetic signal such as radio, television, or other telecommunications.
- **Receiver:** An electronic circuit that receives its input from an antenna, uses electronic filters to separate a wanted radio signal from all other signals picked up by this antenna, amplifies it to a level suitable for further processing, and converts the signal into a form usable for the consumer, such as sound, pictures, digital data, measurement values, navigational positions, etc
- **Antenna sub system:** The antennas on board of a spacecraft provide the dual functions of receiving the uplink and transmitting the downlink signals.

Some alternative terms in use include:

- **Transceiver:** A device that has both a transmitter and a receiver which are combined and share common circuitry
- **Transponders:** A series of interconnected units which forms a single communication channel between receive and transmit antennae in a satellite. Some of the units utilized by a transponder in a given channel may be common to a number of transponders.

Various elements introduced in the preceding are shown in ***Figure 72***.

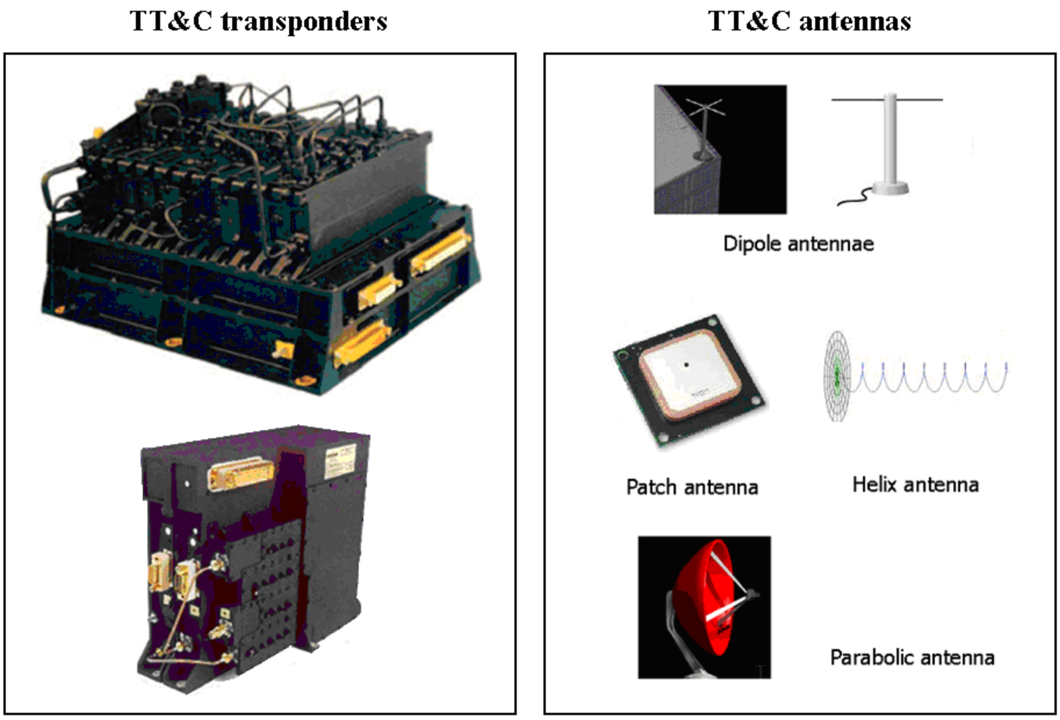


Figure 72: TT&C transponders and antennas

From the figure we also learn that different types of antennas can be distinguished. This is because the type of antenna depends on the transmitter frequency used. In addition, the different shapes allow for differences in antenna gain. From top to bottom the antennas are more directionally sensitive with increasing gain (i.e. better ability to detect a weak signal). The advantages and disadvantages of the two antenna types are given in Table 45.

Table 45: Advantages and disadvantages of two antenna types

High Gain Antenna (HGA)	Low Gain Antenna (LGA)
• Directed and actively pointing antenna for high capacity link • Must accurately be pointed to ground station	• Omni-directional , no need to point • Emergency commanding • Ranging

Most spacecraft have two or more different communications systems on board. One for low transmission rates and one for high transmission rates (usually at another frequency band). In case of low transmission rates one usually uses omnidirectional antennas, whereas for high data rates we tend to use high gain antennas. For instance, the Voyager spacecraft communicates both in S-band and in X-band. For S-band transmissions, Voyager is equipped with both low and high gain antennas. For the X-band Voyager only has a HGA. [Ludwig].

Main configuration issues

Figure 73 shows a schematic of a typical TT&C system. The system comprises of two low gain antennas with hemispherical coverage, two transponders each with transmitter and receiver, and two command decoders. The interface with the rest of the satellite is via the on-board Data Handling (OBDH) subsystem.

- The uplink carrier with the telecommand (TC) signal from the ground station is received by one of the low gain antennas and applied to both receiver inputs via the diplexer (this is some kind of distributor).
- The receiver(s) output the uplinked signal to the active decoder. The decoder recovers the TC data and sends it to the OBDH.
- The active transmitter generates a downlink carrier phase and frequency coherent with the uplink carrier, which allows measurement of Doppler by the ground station and/or other satellites (like TDRSS), aiding satellite localization.
- The uplink signal also contains the ranging signal which is demodulated by the receiver and transmitted back to the ground with the telemetry (TM).

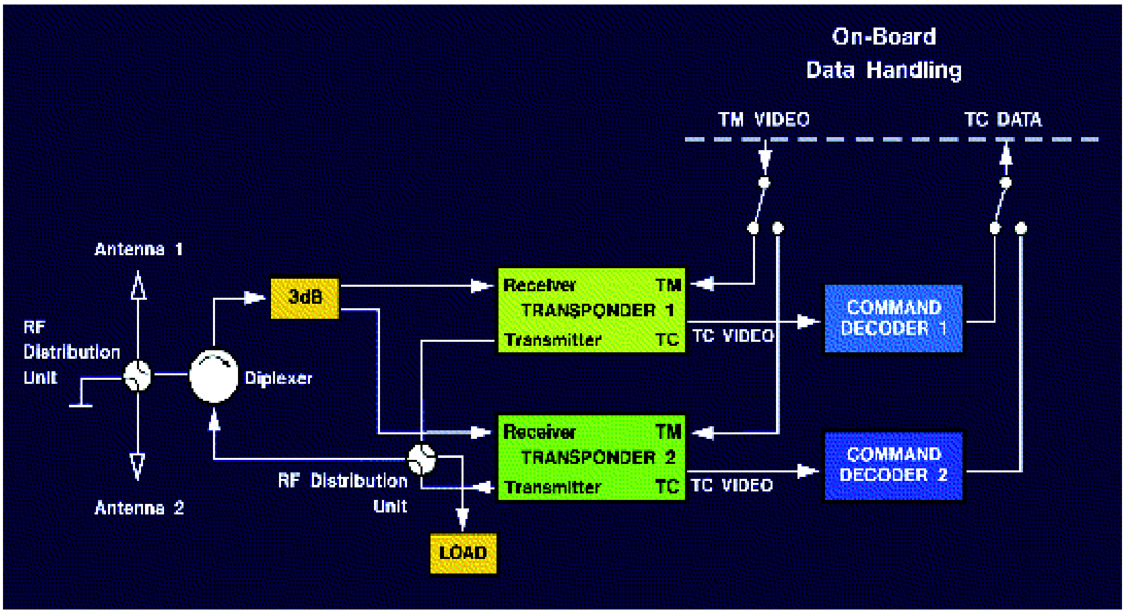


Figure 73: Typical TT&C architecture

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- The uplink signal also contains the ranging signal which is demodulated by the receiver and transmitted back to the ground with the telemetry (TM).

The TT&C system must be operational during all mission phases even if attitude control is lost, thus the antenna system coverage must be as near as possible to omni-directional. The hemi-spherical coverage antenna has low gain (LGA). Since the TT&C system is important for mission success, most components in the system are redundant. In case of high data rates, one may also consider adding a high gain antenna on board. Configuration issues include:

- Direct line of sight between transmitting and receiving station
- The antenna needs to be positioned to adequately cover the receiving/listening area.
- HGA may be mounted on mechanism to allow for steering
- Transponders usually consume a lot of power and therefore generate a lot of heat. They are therefore usually mounted on cold side of satellite
- Short line length between the various elements to reduce losses in the system
- Typically all components are redundant except for the antenna system; the latter is a passive system that has a low failure probability
- Receive and transmit antennas may be one and the same

Dimensioning and sizing
A rough mass estimate of the communications system can be obtained from the information as contained in the section on budgeting (chapter 4). A more accurate estimate can be obtained by collecting historical size and mass data of more or less comparable spacecraft. In case accuracy is still not ideal, we could start breaking up the system in its main components and start dimensioning each of the components.

A first step is to distinguish between transponder/transceiver and antennae. For TT&C transponder sizing the following rules can be used as a first approach:

- Specific power (based on RF output power P): 2.9 W/kg (SSD = 2.8 W/kg)
- Mass density: 0.5-1 kg/liter
- Transmitter dc-to-RF efficiency: 10-50% (average = 18%)

Of course RF power is determined based on the requirement that a reasonable signal to noise ratio should be attained.

For the sizing of antennae, no specific sizing rules are available yet. In that case it is proposed to first determine the type of antennae and their size (gain). As a next step, we than collect data on existing antennae and select a suitable one, or consider having someone designing specific antennae. In some cases the data collected might necessitate us already to reconsider our original choice for the type of antennae.

As an example, we give here the data for a small S-band helix and patch antenna as developed by [Surrey].

	Patch	Helix
RF power handling	Up to 10 W	Up to 10 W
Mass	< 80 g	500 g
Size	82 x 82 x 20 mm	100 x 100 x 500 mm

Decoder and encoder mass have been dealt with in the previous chapter. As a warning, it is mentioned that all elements should be accounted for once, meaning encoders and decoders that have already been taken into account in the C&DH system should not be taken into account when dealing with the TT&C system.

Problems
A number of problems for exercising upon are available via Blackboard (Maple TA), whereas a few are also contained in a separate workbook available from the TU-Delft online print shop. Of this workbook also an electronic copy is available on blackboard.

4.8 Navigation (not part of examination)

Important definitions

- **Navigation** is the process of reading, and controlling the movement of a craft or vehicle from one place to another. For spacecraft it includes the whole of tracking, ranging, orbit determination and timing of actions
- **Tracking** is the use of consecutive observed locations of the same target into tracks (important parameters are range, elevation, and azimuth)
- **Ranging** is a term merely applying for distance metering with moving objects. Combining several metering results in a time sequence leads to tracking

Why navigation (or orbit determination)

- To know where we are (position)
- To estimate time of arrival (ETA, travel time)

Navigation allows for the S/C to determine when it should perform certain operations such as starting an experiment or adjusting its attitude (the direction in which the satellite is pointing), and or start communications with a ground station or other spacecraft in orbit.

Most spacecraft currently are controlled from the ground; some are controlled by astronauts inside of them. A few, like Deep Space 1 (DS1), have special equipment that can navigate "on the fly." The trend is towards more autonomy meaning that more spacecraft currently under design will be able to navigate by themselves in future.

How it works

Regardless of who or what is doing the controlling, though, there are common elements to spacecraft navigation. All navigation systems use the positions of known objects in space as well as the information coming back from the spacecraft to tell where a spacecraft is. The principle of determining one's position is explained using Figure 74. The position of an object on Earth or in space is determined compared to a pre-defined reference frame. To determine the position of some subject (subscript s) requires 2/3 distances or view angles taken from a known position. 2/3 distances define a circle (in 2D) or sphere (3D) around the object. Vehicle location is where circles (spheres) intersect. Once the spacecraft's position is known, the flight path is plotted and thrusters are fired. Key characteristics are position determination accuracy and timeliness (how long it takes to determine some position; if it takes too long we might experience a collision).

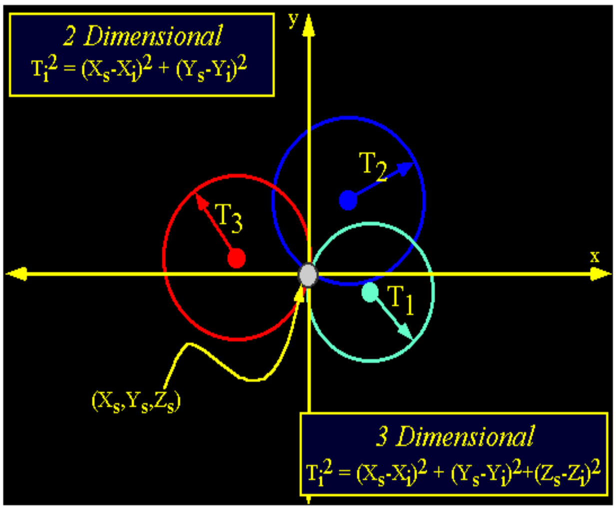

Figure 74: Principle of positioning

Approach/options
Two basic options exist:

- Ranging by a ground station or space-based system like the US TDRSS. For instance, a radar dish, or antenna, transmits pulses of radio waves or microwaves which bounce off any object in their path. The object returns a tiny part of the wave's energy to a dish or antenna which is usually located at the same location as the transmitter. The time it takes for the reflected waves to return to the dish enables a computer to calculate how far away the object is. Doppler processing of the signal can be added to provide accurate velocity information. The ground station performs the processing of the signals and then transmits the location to the S/C. This way the complexity of the S/C can be kept low (high reliability, low cost), but it may take some time for the S/C to know its exact location since the time needed for a communications signal to relay a message increases with distance. Ranging may also be accomplished by a special transponder integrated into the on board command and data handling system, see [SSE] section 13.5.1
- Using an on-board navigation system for instance a GPS/Galileo receiver with the appropriate antennas to receive signals from a global positioning system or star cameras that determine viewing angles with respect to known celestial bodies (mostly deep space missions). Elaborate on board software is needed to determine orbit and location in orbit. Because of the high computer load it may require a dedicated computer on board of the S/C. Such an on-board navigation system allows for increased S/C autonomy and hence short decision making time, but also increased complexity (increased cost) and a higher computational load of the on-board computer. Also the ability to handle “unforeseen” circumstances may reduce.

Depending on the accuracy required, the following options may be selected:

- Tracking with an accuracy up from 50 m:
    - Low accuracy (a few km in LEO up to 50 km in GEO): Ground stations using Doppler tracking; currently the only available option for deep space missions.
    - Moderate accuracy in LEO/MEO: Tracking by satellites (e.g. TDRSS; up from 50 m)
- High accuracy in LEO (15 – 100 m): GPS with or without SA
- Very high accuracy in LEO (1m level): Doris; since 1998, the DORIS system provides orbits in real time, to within a few meters.
- Extremely high position accuracy (cm-level): Doris system (delayed time only), and laser ranging
- To make accurate measurements of change of velocity during trajectory corrections accelerometers can be used. Such accelerometers are sometimes integrated together with gyros that measure rotational motion in a so-called Inertial Measurement Unit (IMU). This is essentially an integrated unit with sensors, mounting hardware, electronics and software.

Fundamentals
Range can be determined using the know velocity of the wave and measuring the time it takes between transmission and receiving the signal, see [132]. Travel velocity can be determined using:

$$f_w = f_b\sqrt{\frac{c+v}{c-v}} \qquad [141]$$

c = velocity of EM waves in vacuum (~3 x 10^8 m/s)
f = frequency (w refers to measured and b to actual freq.)
v = velocity

It is mentioned that only the component of the velocity vector aimed along the line connecting the S/C with the receiving station can be determined. Extensive processing is needed to allow for improved tracking and navigation.

Frequency (or wavelength) usually is selected based on limiting attenuation in bad weather and interference with e.g. communication signals.

Components

Most spacecraft nowadays use a dedicated transponder on-board integrated in the TT&C system. This transponder receives ranging tones from the ground station and retransmits them through the telemetry channels. Turnaround time provides range, whereas the shift in transmitted and received frequency provides the velocity towards the receiving station (after some computation).

The Doris system requires a Doris receiver on board plus 2500 lines of code, requiring 60 Kbits of memory. ***Figure 75*** shows three generations of DORIS receivers, illustrating the trend to smaller and less heavy, but equally capable equipment, and a typical DORIS antenna (length of 42 cm) on the right. A GPS receiver and some GPS antennas are shown in Figure 76, whereas Table 46 and Table 47 present characteristic data on size, power usage and mass of single and dual frequency GPS receivers. Single frequency GPS receivers are less expensive than dual frequency receivers; however they take longer (typically about 15 minutes, but maybe up to about 30 minutes) to arrive at an acceptable solution as compared to less than a minute for the dual frequency receivers.

Figure 75: Typical components of DORIS SC navigation system

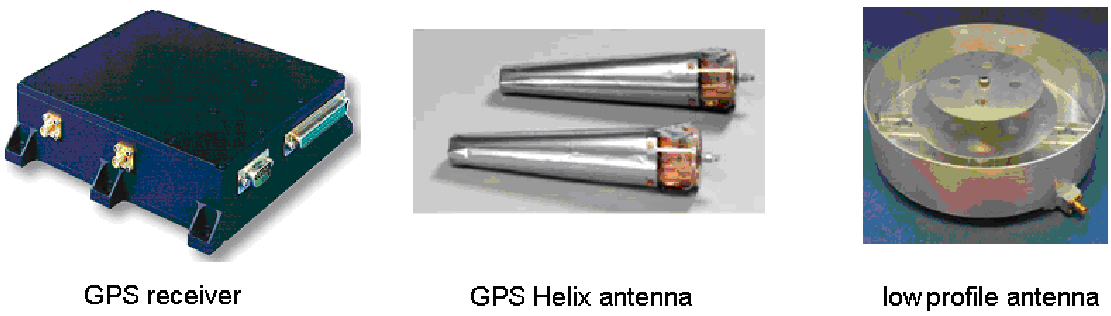

Figure 76: Typical GPS receiver with accompanying antennas

Table 46: Single-frequency GPS receivers for space applications [IAA-B6-0501]

Manufact.	Receiver	Chan	Ant	Power Weight	TID [krad]	Missions, References
Alcatel (F)	TopStar 3000	12-16 C/A	1-4	1.5 W 1.5 kg	>30	Demeter, Kompsat-2;
EADS Astrium (D)	MosaicGNSS	6-8 C/A	1	10 W 1 kg	>30	SARLupe, TerraSAR-X Aeolus
General Dynamics (US)	Viceroy	12 C/A	1-2	4.7 W 1.2 kg	15	MSTI-3, Seastar, MIR, Orbview, Kompsat-1
SSTL (UK)	SGR-05	12, C/A	1	0.8W, 20g	>10	
	SGR-20	4 x 6 C/A	4	6.3 W 1 kg	>10	PROBA-1, UOSat-12 BILSAT-1
DLR (D)	Phoenix-S	12 C/A	1	0.9 W 20 g	15	Proba-2, X-Sat, FLP, ARGO, PRISMA
Accord (IND)	NAV2000HDCP	8 C/A	1	2.5W 50 g		X-Sat

Table 47: Dual-frequency GPS receivers for space applications [IAA-B6-0501]

Manufact.	Receiver	Chan	Ant	Power Weight	TID [krad]	Missions
SAAB (S)	GRAS/GPSOS	12 C/A,P1/2	3	30 W 30kg		METOP
Laben (I)	Lagrange	16 x 3 C/A,P1/2	1	30 W 5.2 kg	20	ENEIDE, Radarsat-2, GOCE
General Dynamics (US)	Monarch	6-24 C/A,P1/2	1-4	25 W 4 kg	100	
JPL (US) / BRE (US)	BlackJack / IGOR	16 x 3 C/A,P1/2	4	10 W 3.2/4.6kg	20	CHAMP, GRACE, Jason-1 / COSMIC, TerraSAR-X
Alcatel (F)	TopStar 3000G2	6 x 2 C/A,L2C	1			Under development; PROBA-2
Austrian Aerospace (A)	Inn. GNSS Navigation Recv.	Up to 36 C/A,P1/2	2		>20	Under development; SWARM
BRE (US)	Pyxis Nautica	16-64 C/A,P1/2 L2C, L5	1-4	20 W 2.5 kg		Under development
NovAtel (CA)	OEM4-G2L	12 x 2 C/A,P2	1	1.5 W 50 g	6	CanX-2; CASSIOPE
Septentrio (B)	PolaRx2	16 x 3 C/A,P1/2	1 (3)	5 W 120 g	9	TET

4.9 Other subsystems (not part of examination)

Environmental control and life support system (ECLSS)
In human spaceflight, the environmental control and life support system is a group of devices that allow a human being to survive in outer space as for example the Lunar module of Apollo 11 given in Figure 77. The life support system may supply: air, water and food. It must also maintain the correct body temperature, an acceptable pressure on the body and deal with the body's waste products. Shielding against harmful external influences such as radiation and micro-meteorites may also be necessary. Components of the life support system are life-critical, and are designed and constructed using safety engineering techniques.

Figure 77: Astronaut in space with ECLSS integrated in backpack and suit

Destruct system
The following text centers about the destruct system of the Space Shuttle Solid Rocket Booster (SRB). Ground commands arm the safe and arm (S&A) device approximately five minutes prior to SRB ignition. If destruct action is required, the nominal range safety destruct procedure will consist of energizing the “arm” command several times, application of a one second pause, then energizing the “fire” command several times or until the destruct action is accomplished. The fire command to the Pyrotechnic Initiator Controller (PIC) discharges its capacitor, igniting the NSD. The detonation from the NSD is propagated through the S&A device transfer charge and the CDF train to the linear shaped charge (LSC). The detonation output of the LSC cuts the case along 70 percent of the length of the Solid Rocket Motor causing destruction of the SRB. An example of a command destruct system is given in Figure 78. A typical minimum overall system reliability goal for the Command Destruct System is 0.999 at a 95 percent confidence level.

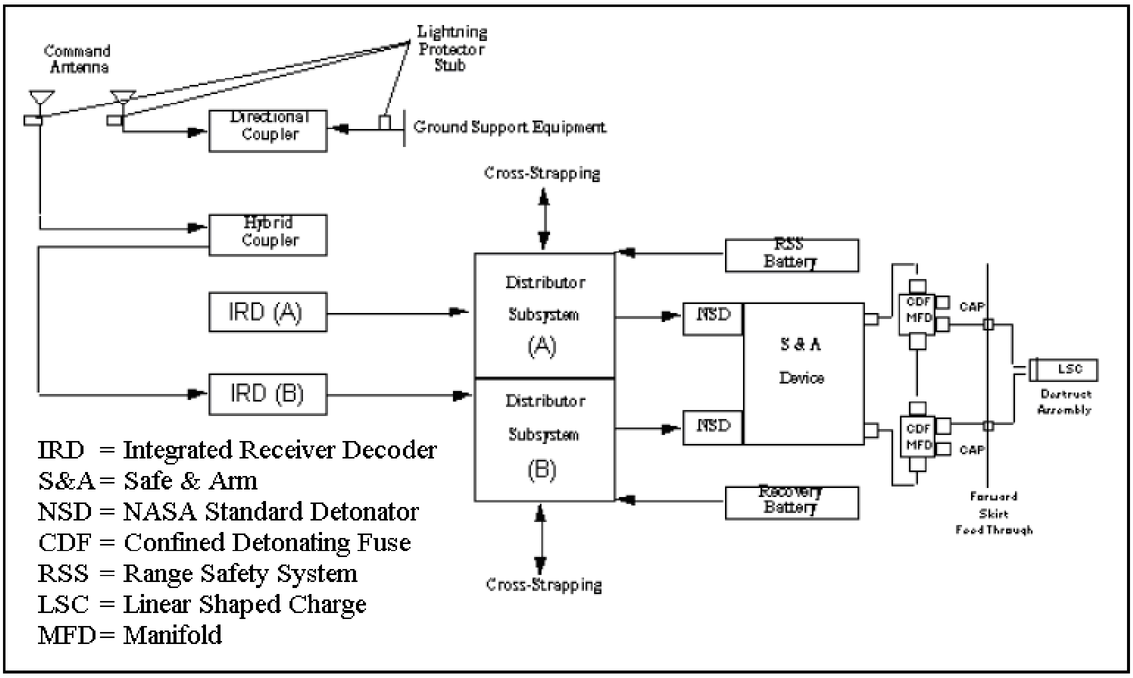

Figure 78:SRB Command Destruct System Functional Diagram

Lander system (parachute, landing gear, balloons)

Landing system includes all equipment needed to ensure a proper landing of the spacecraft. Such systems are amongst others used on lander spacecraft, re-entry vehicles as well as on future aeroplane like space launchers. Examples of (typical components of) lander systems are given in Figure 79 and Figure 80.

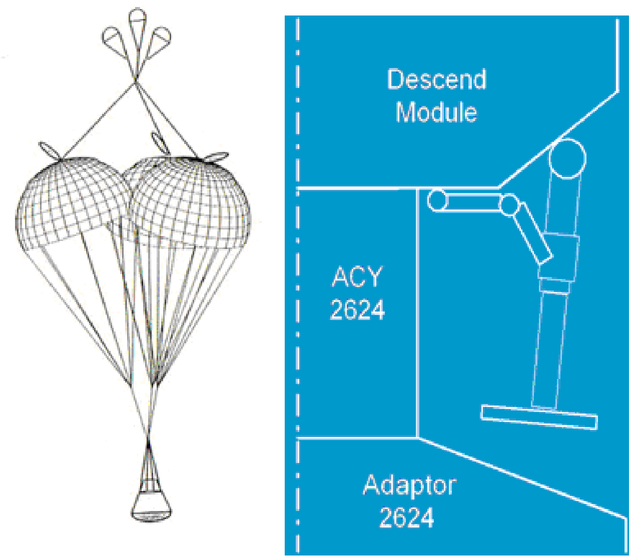

Figure 79: Typical components of a landing system

Figure 80: Vehicle Landing on Mars

Recovery equipment
The recovery system is to ensure the capability to recover a spacecraft, be it an orbiting space capsule or a rocket booster, like the Ariane 5 Solid Rocket boosters. The recovery system may include an altitude determination and command system, a parachute system, a floatation system, strong points for hoisting and a beacon system that provides for information on the whereabouts of the spacecraft/booster. An example of a recovery system and a recovery vehicle are given in Figure 81 and Figure 82, respectively.

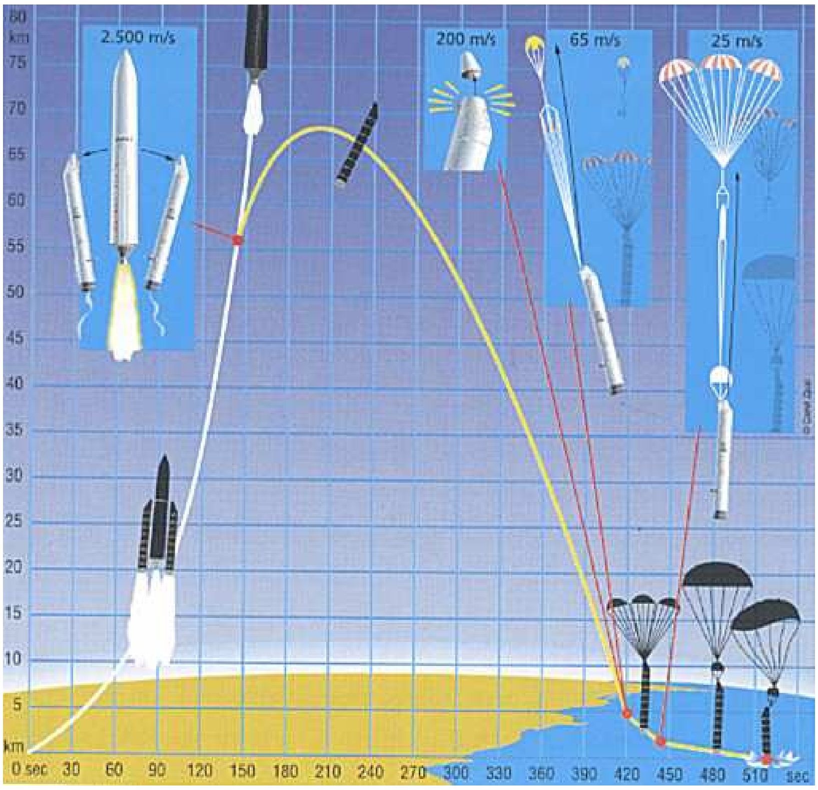

Figure 81: Dutch Space developed booster recovery system

Figure 82: Recovery or re-entry vehicle

Launch escape system (LES)

A Launch Escape System is a top-mounted rocket connected to the crew module of a crewed spacecraft as shown in Figure 83 and used to quickly separate the crew module from the rest of the rocket in case of emergency. Since the escape rockets are above the crew module, an LES typically uses separate nozzles which are angled away from the crew module to prevent the LES exhaust from striking the module, cutting through the hull, and immolating the crew. The LES is designed for use in situations where there is an imminent threat to the crew, such as an impending explosion

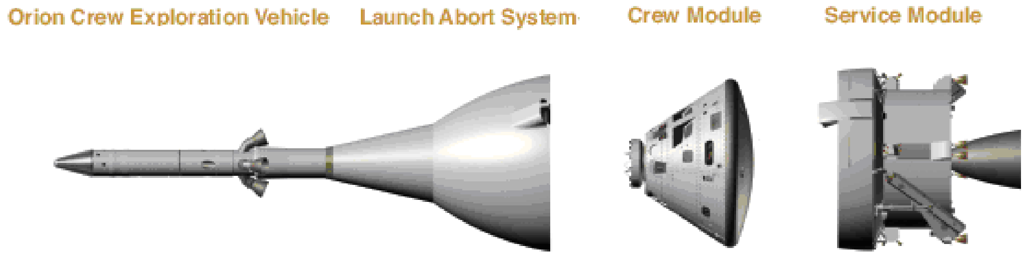

Figure 83: Orion crew exploration vehicle launch abort system

Avionics

The science and technology of electronics and the development of electronic devices as applied to aeronautics and astronautics (from dictionary). Most rockets have some avionics ring/interstage that contains all the instruments and electronics. Next to the avionics the ring/interstage may also contain the power system, etc. Typical applications include:

- Command and Telemetry Processing
- Computers
- Power Distribution and Control
- Attitude and Propulsion
- Spacecraft Thermal Management
- Payload Interface Modules
- Low Voltage Power Supplies

5 Summary

S/C design: An iterative process with each time more detailed and more accurate analysis

- Discussed the need for requirements (to find out what we need to design for) and in the process we learned about sources of requirements, types of requirements (functional, etc.), requirements on requirements (defining them in a SMART way) and requirements flow down.
- Method introduced for estimation of S/C characteristics:
    - General arrangement/configuration/lay-out.
    - Mass, size and power properties (by mission phase).
    - Summary of subsystem characteristics.
    - System parameter; lifetime, reliability, cost, development time.
- Details should be worked out during further analysis (see also later in this lecture series)
    - Iterate, negotiate, and update requirements, constraints and design budgets with feedback from subsystem designers
- Repeating the calculations for different design options allows for performing trade studies.
- Exercises: See Blackboard (Maple assignments) + workbook Spacecraft Design and Sizing”
- Method in principle can also be used for design of other types of spacecraft, but requires own estimation formula. Also system breakdown may result in breaking down the spacecraft into different systems.

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Appendix A: Space maneuvers and mission characteristic velocity

Introduction

To allow a spacecraft (S/C) to maneuver in space and/or to control its attitude, it should have some means to change its velocity. The velocity change required for the various maneuvers usually comes from orbit and attitude control analysis activities and are reported in a ΔV budget (pronounce 'delta V budget). Such a budget than allows for the spacecraft or propulsion system engineer to select the propulsion type and to come up with a propellant mass budget. However, in the very early stages of a project, results of trajectory and attitude control analysis are difficult to come by. The following data subdivided over three broad categories may help you to generate a first rough ΔV budget for the mission at hand without the need for time-consuming calculations.

Given the preliminary character of the data, it is advised to add a proper margin to this data. Note that more accurate data (for instance to be able to reduce the design margin), can be obtained through orbit analysis in the later stages of the design. Fundamentals of orbit analysis and design can be obtained from a number of text books.

Launch/landing

Table 1: Typical ΔV value(s) for sub-orbital flight

Maneuver	ΔV, km/s
Earth surface into LEO	9.5-9.8 km/s (depending on size of launcher)
LEO to Earth surface	Orbital maneuvering burn to lower perigee into the atmosphere, atmospheric drag takes care of the rest.
Moon surface into Low Lunar Orbit (LLO)	2.6 km/s
LLO to Moon surface	2.9 km/s
Mars Surface to low Mars orbit	5.7 km/s
Low Mars orbit to Mars surface	4.7 km/s (atmospheric drag does play a role)

Impulsive shot space maneuvers

Table 2: Typical Δv value(s) for impulsive shot space maneuvers

Maneuver	ΔV
Orbit transfer:	
LEO to GEO	3.95 km/s (no plane change required)
LEO to GEO	4.2 km/s (including plane change of 28 deg)
GTO to GEO (1)	1.5 km/s (no plane change required)
GTO to GEO (2)	1.8 km/s (incl. plane change of 28 deg.)
LEO to Earth escape	3.2 km/s
LEO to trans-lunar orbit	3.1 km/s
LEO to lunar orbit	3.9 km/s
GTO to lunar orbit	1.7 km/s
LEO to Mars orbit	5.7 km/s
LEO to solar escape	8.7 km/s
Orbit control:	
Station-keeping (GEO)	50-55 m/s per year
Station-keeping in Moon orbit	100-400 m/s per year
Station-keeping in L1/L2	30-100 m/s
Orbit control: Drag compensation (Earth orbit)	
alt.: 400-500 km	< 100 m/s per year max. (<25 m/s average)
alt.: 500-600 km	< 25 m/s per year max. (< 5 m/s average)
alt.: >600 km	< 7.5 m/s per year max.
Attitude control: 3-axis control in Earth orbit	2-6 m/s per year
Auxiliary tasks (Earth orbit):	
Spin-up or de-spin	5-10 m/s per maneuver
Stage or booster separation	5-10 m/s per maneuver
Momentum wheel unloading	2-6 m/s per year

Constant low thrust space maneuvers

Because of gravity loss, low thrust-to-weight (T/W) propulsion systems suffer a loss in performance equivalent to increasing the effective mission ΔV. For example, the impulsive ΔV for a high T/W transfer from LEO to GEO is 4.2 km/s; for a low T/W transfer, the effective ΔV is about 5.9 km/s. However, even with gravity losses, low T/W propulsion systems can still out-perform high T/W impulsive systems, because the very high specific impulse of some low T/W systems (greater than 1000 s) more than compensates for the increase in effective ΔV.

Table 3: Typical ΔV value(s) for constant low thrust (acceleration < 0.001 m/s^2) orbit transfer (propellant mass is negligible)

Maneuver	ΔV, km/s	Transfer time
LEO (200 km altitude) to GEO (no plane change)	4.71	a is 0.001 m/s^2: ~55 days
LEO (200 km altitude) to GEO (including 28 deg. plane change)	5.97	a is 0.001 m/s^2: ~70 days
LEO to MEO (19150 km altitude; no plane change)	3.83	a is 0.001 m/s^2: ~44 days
LEO to Earth escape for different values of initial acceleration-to-local gravitational acceleration: 10^{-2} 10^{-3} 10^{-4} 10^{-5}	 5.82 6.66 7.08 7.43	
LEO to Low Lunar orbit GTO to Low Lunar orbit	~8 3.6-4	months-year 250-450 days
LEO to Mars orbit	~15	~2.2 years

1. Transfer or trip time for constant thrust spiral is is calculated by dividing total propellant mass by mass flow. Total propellant mass is calculated using the rocket equation. In case of negligible propellant mass (constant acceleration), transfer time can be calculated by dividing the velocity change by the acceleration.
2. Δv for LEO to GEO transfer orbit calculated using T.N. Edelbaum's equation: $\Delta v = \mathrm{SQRT}(v_1^2 - 2 v_1 v_2 \cos(\pi/2\ \Delta i) + v_2^2)$ where v_1 is circular velocity initial orbit, v_2 is circular velocity final orbit, and Δi is plane change in degrees.
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5. Value for LEO to Low Lunar Orbit taken from Optimized Low-Thrust Transfer for Space Tugs by Pukniel
6. Values for LEO to Lunar/Mars orbit taken from NASA-JPL.

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Appendix B: Spacecraft data

In this document, data are introduced that can be used for spacecraft vehicle level parameter estimation. Typical parameters considered are mass, volume, power, cost and life.

Spacecraft overall Mass, Cost and Life Data

In the table on the next two pages mass, cost and life data of specific Earth satellites are collected. The first column gives the application for which the spacecraft is used. Terminology has been kept identical to the one introduced in ae1102. The second column gives more details on what the spacecraft actually does in the field of application, for instance DTH is Direct to Home communications, FSS is Fixed Satellite Services, data relay and so on. Columns 3 and 4 give the satellite name and the operator, i.e. the company that operates the satellite. Column 5 gives the name of the bus/platform used. This essentially is the spacecraft platform used to support the payload. Manufacturers nowadays have a standard range of platforms that can be adapted to suite a range of applications. Column 6 gives the target or operating orbit of the spacecraft. We distinguish Low Earth Orbit, Geostationary Earth Orbit and Intermediate Earth Orbit (MEO). Next few columns give information on life, launch mass and dry spacecraft mass. Here launch mass is considered the total mass at launch and may include the S/C wet mass (i.e. S/C mass including consumables and mass margin), the launch vehicle adapter and when the spacecraft is equipped with a kick stage it also includes the mass of a kick stage. S/C dry mass is S/C mass excluding consumables and mass margin. The columns 10 and 12 give the S/C cost as well as the year in which the cost was reported. Column 11 and 13 then give the cost standardized to year 2000 cost (as to allow for a fair comparison) and the year 2000 cost per kg.

Spacecraft

Application	Mission	Satellite	Operator	Bus	Orbit	Life	Launch mass [kg]	Dry mass [kg]	Cost [M$]	Cost FY 00 [M$]	Year	Specific cost [k$/kg]
Communications	FSS	Thaicom	Shinawatra	HS 376 L	GEO	13.5	1080	436	50	59.7	1991	137.0
Communications		Apstar 1A	APT satellite Co	HS 376	GEO	10	1383	557	80	87.1	1995	156.5
Communications	DBS	BS-3N	Telecom Japan	GE 3000	GEO	7	1210	575	88	109.7	1990	190.8
Communications	FSS	Arabsat 1	ASCO	Spacebus 1000	GEO	7	1310	600	44.8	87.8	1981	146.4
Communications	Mobile	Inmarsat 2	Inmarsat	Eurostar 1000	GEO	10	1385	624	65	95.4	1985	153.0
Communications		Galaxy 9	Hughes Comms. Inc.	HS 376	GEO	10	1200	654	80	87.1	1995	133.3
Communications		Brasilsat B-3A	Embratel	HS 376W	GEO	12	1765	826	70	83.6	1991	101.2
Communications	MSS	MTSat-1	Min. of transp. Japan	FS-1300	GEO	12	2900	1223	118.5	129.1	1995	105.5
Communications	DTH	PAS 6	PanamSat	FS 1300	GEO	15	3020	1260	150	166.5	1994	132.1
Communications		MSAT 1	TMI comm.	HS-601	GEO	12	2770	1270	100.0	124.7	1990	98.2
Communications	FSS	Solidaridad	Telecom Mexico	HS-601	GEO	14	2776	1291	91.7	109.6	1991	84.9
Communications		Intelsat V			GEO	7	2000	835	56.4	92.5	1996	110.8
Communications	FSS	Intelsat 709	Intelsat	FS 1300	GEO	10.9	3560	1450	90	101.9	1993	70.3
Communications		Telstar 5	ATT Skynet	FS 1300	GEO	12	3758	1467	120	130.7	1995	89.1
Communications		Intelsat 806	Intelsat	Astro Space 7000	GEO	10	3245	1538	82.3	91.3	1994	59.4
Communications	FSS, VSAT	JCSat 4	Japan Sat. Syst. Inc.	HS 601	GEO	10	3100	1820	125	136.2	1995	74.8
Communications	Data relay	TDRS	Goddard	HS-601 HP	GEO	10	2200	1735	160.5	174.8	1995	100.8
Communications	DTH	Amos 1	SpaceCom Sat.	IAI	GEO	10	961	450				
Communications	DTH	PAS 5	PanamSat	HS 601 HP	GEO	15	3720		110	119.8	1995	
Communications	DTH	Hot Bird-4	Eutelsat	Eurostar 2000	GEO	14.5	2900				1994	
Communications	Military	DSCS III			GEO	10	1200	867.3			1996	196.9
Communications	DTH	Measat-2	Binariang (Malaysia)	HS 376	GEO	10	1400		100	111.0	1994	
Communications	BBS	Eutelsat-3	Eutelsat	Spacebus 3000	GEO	15					1995	
Communications	DTH	Galaxy 8i	Hughes Comms. Inc.	HS 376	GEO	15			80		1995	
Communications		Astra 1G	SES	HS 601 HP	GEO	15			110	119.8	1995	
Communications	Data	Orbcomm	Orbcomm	OSC Microlab	LEO	4-6	40.3	39	3.3	3.7	1993	95.8
Communications	MSS	Globalstar	Globalstar	Loral	LEO	7.5	450	400	13.9	15.1	1995	37.9
Communications	MSS	Iridium	Iridium	Lockheed	LEO	5-8	689	574	5.83	6.6	1993	11.5
Communications	Internet	Skybridge	Skybridge		LEO		1250		37.5			
Earth observation	Imaging	Orbview 3		OSC Microlab	LEO	3	185					
Earth observation	Imaging	Clark	NASA		LEO	3	278	266	49	54.4	1994	204.5
Earth observation	Imaging	Lewis	NASA	TRW AB600	LEO	3	385.6	276	59	65.5	1994	237.3
Earth observation	Imaging	Eros 1A			LEO	6	250		100		1998	
Earth observation	Imaging	Ikonos			LEO	5	725					
Earth observation	Wind speed	QuickSCAT	NASA	Ball	LEO	3-5	971		52	54.3	1997	
Earth observation	Imaging	Quickbird -2	EarthWatch	Ball	LEO		906					
Earth observation	Radar	Sentinel I	ESA	EADS	LEO	7	2300		320.6	275.0	2007	
Earth observation	Imaging	AEOLUS	ESA	EADS	LEO	3	1500	1100	210.6	190.9	2005	173.6
Earth observation	Imaging	SPOT 4	SPOT	Helios	LEO	5	2500	2200				

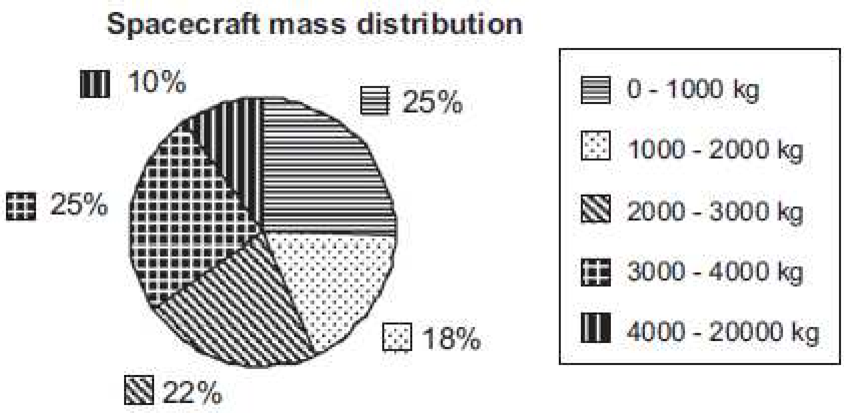
Spacecraft

Application	Mission	Satellite	Operator	Bus	Orbit	Life	Launch mass	Dry mass	Cost	Cost	Year	Specific cost
Earth observation	Imaging	Landsat 6	EOSat		LEO	5	2750	1740	398	462.8	1992	266.0
Navigation	GPS	Block I	NAVSTAR-GPS JPO		MEO	5	770	479.1	20	21.3	1996	44.6
Navigation	GPS	Block IIA	NAVSTAR-GPS JPO	Rockwell	MEO	7.5	1881	870	40	41.8	1996	48.0
Navigation	GPS	Block IIR	NAVSTAR-GPS JPO	Astro Space 4000	MEO	10	2040	980.2	28.75	37.3	1989	38.0
Navigation	GPS	Block III	NAVSTAR-GPS JPO		MEO				41.5	45.2	1995	
Weather	Imaging	Meteosat	Eumetsat		GEO	7	696	288	82.9	90.3	1995	313.6
Weather	Imaging	MSG	Eumetsat		GEO	7	2043		121.5	129.7	2001	
Military	Reconaissance	DSP block 5	US air force	TRW	GEO	5	2277	1704.4	400	518.8	1989	304.4
Military	Reconaissance	Helios 1A/B							900		1995	
Military	Reconaissance	Helios 2					4200		1000		1995	
Science	X-ray telescope	XMM-Newton							700			
Science	Bepicolombo mission								300		2000	
Science	Gaia mission								300			
Science	Solar orbiter mission								150			

Some observations related to the foregoing data:

Related to Mass

S/C mass ranges up to 4,200 kg at launch and 2200 kg dry (leaving out some exceptions, like space station and Envisat[1]). Research conducted by [Tafazoli] shows that there is no clear mass range which is most suited to spacecraft, see figure "Spacecraft mass distribution", except for the limited number of spacecraft in the highest mass range. So it seems that currently most spacecraft are designed for the mass range up to 4000 kg. More than 4000 kg still is somewhat special.



Related to Cost

Spacecraft cost ranges from 10.000 US$/kg up to 500.000 US$/kg (FY 2000). In more detail, it shows:

- GEO communications satellites: 50,000 to 200,000 US$/kg (FY 2000) with an average specific cost of 120,000 k$/kg
- LEO communications satellites: 10,000-95,000 US$/kg with an average value of 48,000 US$/kg
- Navigation satellites: 45,000 US$/kg
- Weather and military satellites: 300,000 US$/kg

Related to Life

- LEO S/C have a life ranging up to 5-7 year
- MEO S/C have a life in the range up to 5-10 year
- GEO S/C have a life ranging up to 10-15 year

[1] Adapted from Wikipedia: **Envisat** ("Environmental Satellite") is an Earth-observing satellite. It was launched on 1 March 2002 aboard an Ariane 5 from the Guyana Space Centre in Kourou, French Guyana into a Sun synchronous polar orbit at an altitude of 790 km (± 10 km). It orbits the Earth in about 101 minutes with a repeat cycle of 35 days.

This €2.3 billion European Space Agency (ESA) program launched the largest earth observation satellite put into space (as of late 2006), being 26 m × 10 m × 5 m and having a mass of 8.5 t.

Spacecraft data in relation to payload data

Spacecraft dry mass and payload mass data

Table 1: Dry mass and payload mass percentage of some large satellites [SMAD, SSE]

Satellite	Dry Mass (kg)	Payload mass %
FLTSATCOM 1-5	849.6	26.54
FLTSATCOM 6	870.9	26.38
FLTSATCOM 7-8	1041.9	32.80
DSCS II	475.9	23.02
DSCS III	867.3	32.34
NATO III	320.4	22.12
INTELSAT IV	532.8	31.24
INTELSAT V	835.0	28.85
INTELSAT VI	1779.0	37.60
TDRSS	1565.7	24.56
GPS Blk 1	479.1	20.49
GPS Blk 2, 1	699.1	20.15
GPS Blk 2, 2	858.0	23.02
P80-1	1704.4	41.06
DSP 15	2114.9	36.91
DMSP 5D-2	814.6	29.85
DMSP 5D-3	1012.3	30.45
Average		28.7
Standard Deviation		6.2
Avg. % of Payload Mass		100

The table shows typical dry mass values for large satellites in the mass range 500-2000 kg. Payload mass on average is 28.7% of the dry mass, with a standard deviation of 6.2%. This means that 65% of all satellites have a payload mass in the range 28.7% ± 6.2% (22.5% - 34.9%) of the dry mass. From the above data a simple estimation relationship to estimate S/C dry mass as a function of payload mass can be deduced of the form:

$$M_{dry} = 1/X \cdot M_{payload}$$

Here X is the payload mass to dry mass fraction (= payload mass percentage divided by 100). For our example given here based on the average payload mass percentage, X would be 0.287, and we would find that dry mass on average is 3.48 (= 1/0.287) times the payload mass.

The first 10 S/C in the above table are communications satellites. Next follow 3 navigations satellites and then 4 observation satellites. The three navigation satellites all seem to have a lower than average payload mass fraction (percentage), whereas the 4 observation spacecraft show a slightly higher than average payload mass fraction. This might indicate that (average) payload mass fraction differs depending on the type of spacecraft.

The next table shows identical mass data, but now specifically for large GEO communications satellites. Average payload mass percentage and standard deviation agree fairly well with the results indicated in Table 1.

Table 2: Mass distribution of some large GEO telecommunications satellites [MediaGlobe]

Spacecraft #	Name	Dry Mass (kg)	Payload mass (% of dry mass)
1	ANIK E	1270	27.6
2	Arabsat (not 2)	573	21.4
3	Astra 1B	1179	30.0
4	DFS Kopernikus	656	24.1
5	Fordsat	1094	28.9
6	HS 601	1459	49.8
7	Intelsat VII	1450	30.8
8	Intelsat VIIA	1823	28.8
9	OLYMPUS	1158	28.5
10	SATCOM K3	1018	19.0
11	TELSTAR 4	1621	24.1
	Average		28.4
	Standard Deviation		8.0

Table 3 provides data specifically for small satellites with a dry mass in the range 1-300 kg. From the calculated average payload fraction (payload mass/dry mass), we see that payload mass for smaller satellites is on average smaller than for the larger spacecraft (Tables 1 and 2). We also see that payload fraction shows a larger spread with values ranging from 12.2% up to 33.7%.

Table 3: Small satellite mass data [Zandbergen]

S/C name	Application	Dry mass (kg)	Payload mass (% of dry mass)
Orsted	Science	56.3	22.9
Freya	Science	216.9	33.7
SAMPEX	Science	160	32.5
ANS	Science	129.3	33.2
Viking	Science	289.6	16.8
Bird	Science	77.74	30.0
NATO III	Commns	316.6	22.1
Gurwin II	Techn. test	47.0	14.0
Temisat	Comm.	41.9	25.8
ORBCOMM	Comm.	47.5	19.4
PoSAT-1	Comm../test	50.2	12.2
Hausat-1	Techn. test	1	16.0
Delfi C3	Techn. test	2.9	16.7
Average			22.7
Average (total is 100%)			**22.3**
Average (excl. propulsion)			23.5

Based on the Table 3, we find that S/C dry mass on average is 4.5 times payload mass, but the table also shows that S/C dry mass may range anywhere in between 3 - 8 times payload mass.

Spacecraft Power data

Table 1: Total spacecraft power (in Watt EOL) and payload power (expressed as a percentage of total power) for several large geostationary telecom. Satellites [MediaGlobe]

Satellite	Total Load (W)	Payload power (% of total power)
ANIK E	3482	86.2%
Arabsat (not 2)	1362	72.7%
Astra 1B	2790	76.6%
DFS Kopernikus	1412	63.5%
Fordsat	3110	79.1%
HS 601	3350	79.4%
Intelsat VII	3569	72.3%
Intelsat VIIA	4567	79.1%
OLYMPUS	2832	75.9%
SATCOM K3	3150	81.6%
TELSTAR 4	5673	84.9%
Average %		77.4%
STD		6.37%

NA) Not Available, most likely incorporated in other subsystem

From the data we find typical S/C power levels up to 6 kW. However, most of the satellites in our list are quite old. History shows an ever increasing trend in power usage with early spacecraft using about 1 W and current spacecraft using 1 – 15 kW of power. Evolving trends suggest a further two orders of magnitude increase may still be needed. Considering the payload power fraction of total power, we find that payload on average consumes 77.4% of total power with a standard deviation of 6.4%. This high percentage is typical for communications satellites. For other types of spacecraft (Earth Observation S/C and deep space S/C) generally lower percentage values apply.

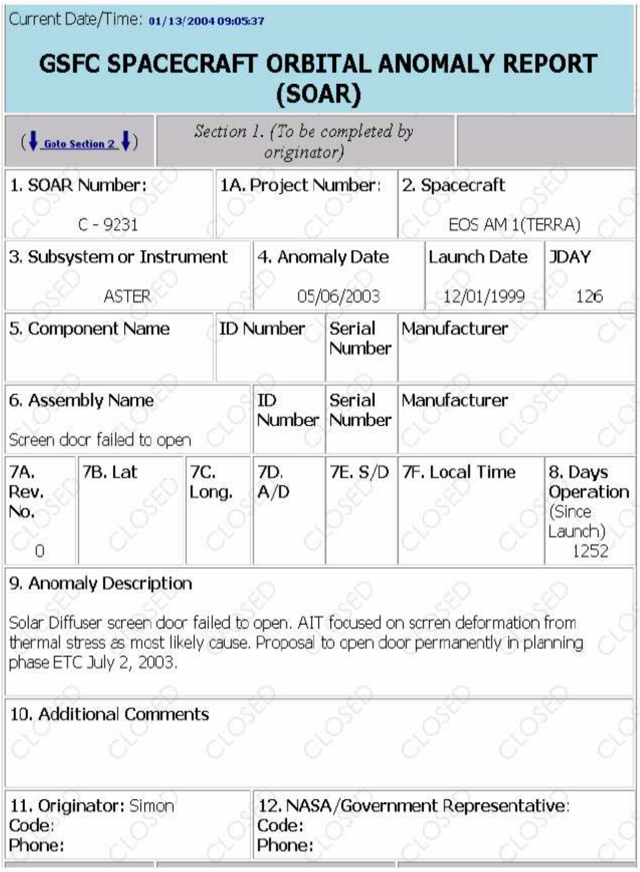
Spacecraft failures and failure data

Spacecraft failures are typically reported using so-called “Anomaly Reports” see for instance *Figure 1* taken from [Remez].

In the year 2003 GSFC (Goddard Space Flight Centre) had 61 orbiting spacecraft [Remez]. For these spacecraft a total of 439 anomalies were counted of which most (86%) had negligible or no effect what so ever on the spacecraft or mission. 13% had a minor effect and only 2% had a substantial or major to catastrophic effect. It is failures in the latter two – three categories that usually lead to an insurance claim, whereas only failures from the final category are counted as real failures. From the data given, we can determine an average of 0.14 major or catastrophic failures per spacecraft per year (at least for the spacecraft GSFC are operating).

Jane’s Space Directory provides tabulated data on reported S/C failures thereby focusing on the more serious failures. Over the period 1995 up to and including 2000 the tabulated data shows ~565 serious failures or ~95 serious failures per year. Given that at any instant in time we have about 800 active S/C, this gives on average 0.12 serious failures per spacecraft per year.

[Sultan] has investigated the distribution of reported failures (most likely only the more serious ones are reported) in spacecraft over the period 1995 up to and including 2000. Sultan found that on average 40% of the failures are attributed to the payload and the remainder to the S/C bus, i.e. the platform.


Current Date/Time: 01/13/2004 09:05:37

GSFC SPACECRAFT ORBITAL ANOMALY REPORT (SOAR)

(Goto Section 2)	*Section 1. (To be completed by originator)*						
1. SOAR Number: C - 9231		1A. Project Number:		2. Spacecraft EOS AM 1(TERRA)			
3. Subsystem or Instrument ASTER		4. Anomaly Date 05/06/2003		Launch Date 12/01/1999		JDAY 126	
5. Component Name		ID Number		Serial Number	Manufacturer		
6. Assembly Name Screen door failed to open		ID Number		Serial Number	Manufacturer		
7A. Rev. No. 0	7B. Lat	7C. Long.	7D. A/D	7E. S/D	7F. Local Time	8. Days Operation (Since Launch) 1252	
9. Anomaly Description Solar Diffuser screen door failed to open. AIT focused on scrren deformation from thermal stress as most likely cause. Proposal to open door permanently in planning phase ETC July 2, 2003.							
10. Additional Comments							
11. Originator: Simon Code: Phone:			12. NASA/Government Representative: Code: Phone:				

Figure 1: Orbital Anomaly report [Remez]

Using data on failures, we can calculate failure rate data. Once failure rate data are known, reliability can be determined using:

$$R = (e)^{-(\lambda \cdot t)} \quad [1]$$

Here R indicates reliability, λ indicates the failure rate of the vehicle and t indicates the time period (life) considered.

Another way is to use reported reliability data of spacecraft to compute failure rate data and then use these data to compute reliability for different mission durations. Some spacecraft reliability data as well as more detailed data for platform, and payloads are given in the next table. Notice that S/C reliability can be determined by multiplying payload reliability with platform reliability.

Table 1: Spacecraft reliability data

System	Designation	Company/country	Reliability (minimum) [-]	Life [yr]
S/C	Geostat. Met. Sat.	Japan	0,5	5
	SPOT 4	SPOT	0,8	4
Platform	Eurostar	MMS	0,86	10
	ETS	NASDA	0,8	10
	BS-3a	Japan, Telcomm.	0,8	7
	FS1300	Space Systems Loral	0,875	15
Comms. payload	HS-376	Hughes (now Boeing)	0,97	10
	HS-393	Hughes (now Boeing)	0,8	10
	BS-3a	Japan, Telcomm.	0,9	7
	TV-sat 2	Telenor	0,68	7,3
	Eutelsat 2	Eutelsat	0,74	7

Spacecraft development/production time data

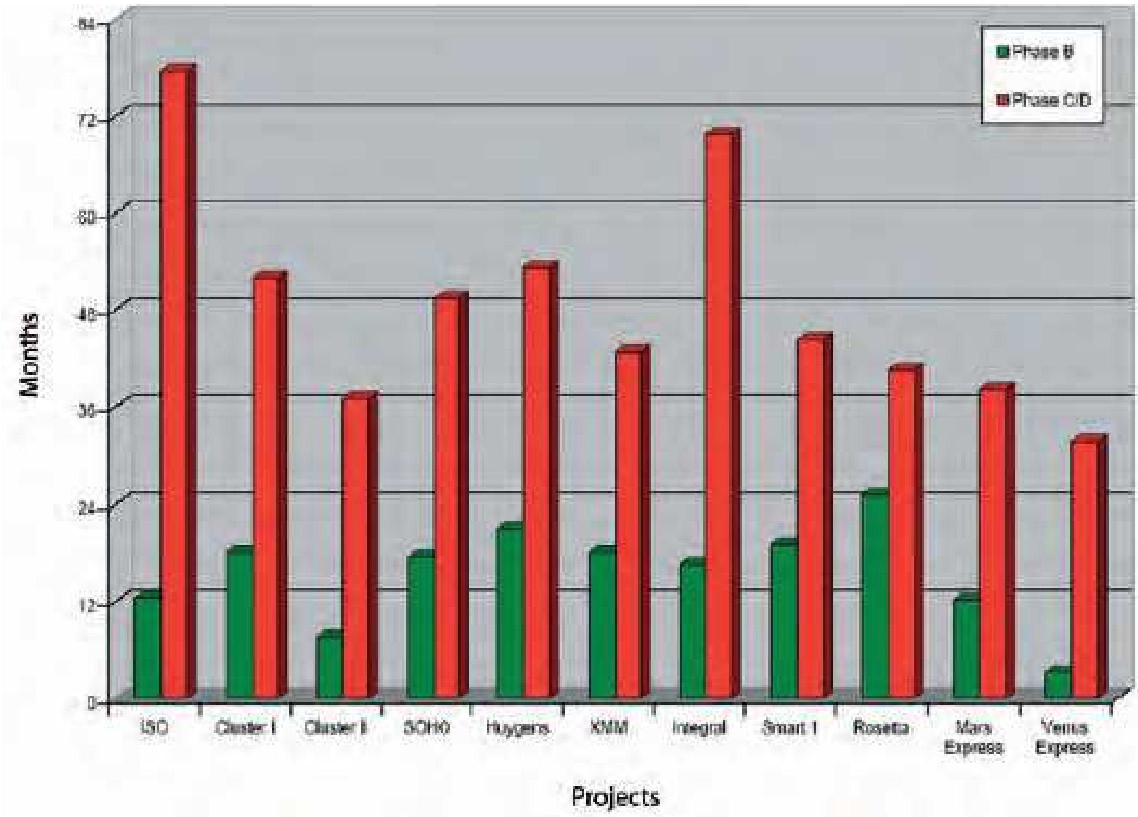


Figure 1: Spacecraft development time (ESA)

From the data given in the above figure, it follows that:

- Phase B for ESA projects typically take 13.8 months (~1 year) with an SSD of 5.4 months (~0.5 year).
- Phase C/D for ESA projects typically takes 47.7 months (or ~4 years) with a SSD of 14.2 months (~ 1 year)

Without providing evidence, we mention that a typical phase A/0 study typically takes 4 months.

Note the duration of the phase B and C/D for Venus Express in the figure. We find that especially the phase B is much shorter than average. This is because Venus Express essentially was kept identical to Mars Express, meaning same instruments, same basic lay-out and bus equipments. The reduction in phase C/D duration is much less than for the phase B. This is attributed to the need to adapt all equipments as to cope with the different environments and that integration and test activities still make up a substantial amount of time. So for Venus Express most tests performed for Mars Express were repeated just to make sure that it would really work under all conditions.

The next gives typical production times as currently realized by commercial GEO communication satellite producers. We find an average value close to 28 months with one manufacturer peaking at 40 months. Note that the data given in the figure are average manufacturer data and do not take into account satellite-to satellite variations. This means that individual satellites may require less (or more) time to manufacture than indicated in the figure. This depends to a large extend on whether the spacecraft is one in a series of identical ones or is a "new" design.

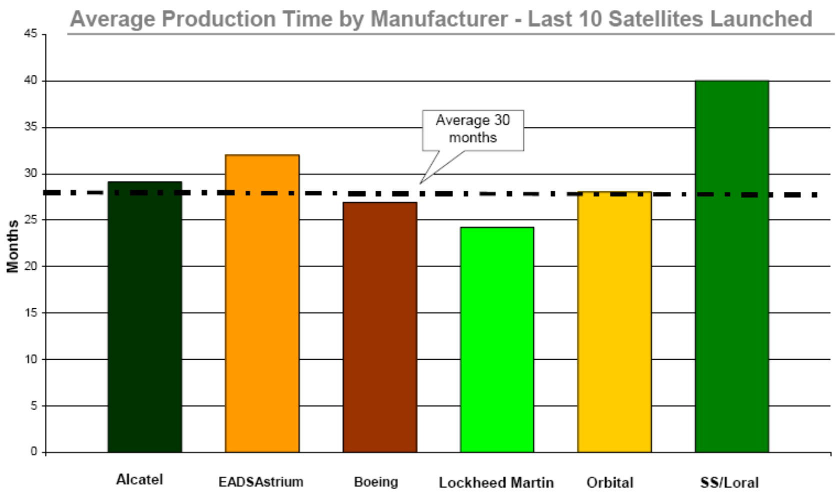


Figure 2: Commercial GEO communications satellite production time [Futron 2004]

The production times cover design, development and manufacturing. That the average production time is much shorter than the time required for the design, development and manufacturing of ESA spacecraft is attributed to the philosophy of commercial satellite manufacturers to only incorporate well tried and tested technologies in the design, whereas ESA spacecraft also include a lot of "firsts".

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5. Zandbergen B.T.C., personal files, 2009

Appendix C: Spacecraft level estimating relationships for mass, power, etc.

Introduction

In this section a number of estimation relationships are given that allow for generating a rough estimate for various spacecraft charactereristics (say the dependent variable “y”) in relation to some independent variable (say “x”), as determined by the experimenter. These relationships have all been determined through curve fitting, where simple mathematical functions are fitted best to a series of data points.

As a curve fit usually is only valid for a certain range of data, this range is included in the tables, provided that this data is available.

To allow for information on the “goodness of fit” of the relation, data is provided on number of data points used as well as on the standard deviation, standard error of estimate (SEE) and/or the R-squared[2] (R^2) value, again provided that this data is available.

[2] Statistical measure of how well a regression line approximates real data points; an R-squared of 1.0 (100%) indicates a perfect fit.

Part 1: Spacecraft vehicle estimation relationships

Vehicle mass

Vehicle mass can be estimated directly using the relations from *Table 1* (loaded mass) or by adding the results from *Table 2* (dry mass) and
*Table **3*** (on station propellant mass). In case significant maneuvers are needed, propellant mass is to be calculated separately using the rocket equation.

Table 1: On-station mass/loaded mass estimation

Earth orbiting spacecraft	
$M_{on\ station} = 3.66\ M_{PL}$ kg [1]	
Source	Zandbergen (34 data points)
Payload mass range	20-2150 kg
R^2	0.899
SEE	30.4%
Earth Observation spacecraft	
$M_{on\ station} = 3.78\ M_{PL}$ kg [2]	
Source	Zandbergen (16 data points)
Payload mass range	20-2150 kg
R^2	0.905
SEE	34.7%
Manned entry vehicles	
$M_{on\ station} = 1.861\ M_{PL} + 1952.1$ kg [3]	
Source	Zandbergen (6 data points)
Payload mass range	20-6000 kg
R^2	0.745
SEE	44.2%
Unmanned entry vehicles	
$M_{Loaded} = 1.404\ M_{PL} + 224.3$ kg [4]	
Source	Zandbergen (10 data points)
Payload mass range	15-300 kg
R^2	0.322
SEE	56.3%
Deep space probes	
$M_{Loaded} = 4.285\ M_{PL} + 333.6$ kg [5]	
Source	Zandbergen (23 data points)
Payload mass range	10-325 kg
R^2	0.379
SEE	21.0%

Table 2: Dry mass estimation

Earth orbiting spacecraft	
$M_{Dry} = a \cdot M_{PL}$ kg	[6]
Source	Brown (46 data points)
a	Average: 4.8 Range: 3-7
Payload mass range	20-550 kg
$M_{Dry} = a \cdot M_{PL}$ kg	[7]
Source	Larson & Wertz
a	Average: 3.33 Range: 2.0-5.9
Payload mass range	-
$M_{Dry} = 2.058 \cdot M_{PL} + 342.8$ kg	[8]
Source	Zandbergen (19 data points)
Payload mass range	50-950 kg
R^2	0.9436
SEE	14.6%
Communications satellites	
$M_{Dry} = 3.6 \cdot M_{PL}$ kg	[9]
Source	Brown (7 data points)
Payload mass range	100-620 kg
$M_{Dry} = 1.8225 \cdot M_{PL} + 545.1$ kg	[10]
Source	Zandbergen (12 data points)
Payload mass range	100-750 kg
SEE	19.4%
Planetary vehicles/deep space probes	
$M_{Dry} = 2.5112 \cdot M_{PL} + 215.9$ kg	[11]
Source	Zandbergen (23 data points)
Payload mass range	10-325 kg
R^2	0.638
SEE	22%
$M_{Dry} = 7.5 \cdot M_{PL}$ kg	[12]
Source	Brown (11 data points)
Payload mass range	10-160 kg

Table 3: RCS propellant mass estimation

satellites	
$M_{RCS} = 0.105 \cdot \left(M_{SC}\right)_{\text{on station mass}}$ kg	[13]
Source	Zandbergen
SSD	55% of estimated value

Vehicle power

Table 4: Total power estimation (photovoltaic systems only)

All missions	
$P_t = 1.13 \cdot P_{PL} + 122$ W	[14]
Source	Brown (40 data points)
Payload power range	5 W – 1000 W
Large satellites (> 500 W total)	
$P_t = 1.85 \cdot P_{PL}$ W	[15]
Source	SMAD
Comment	Actual values are reported to be within ±35 % from the estimated value
Small satellites (< 500 W total)	
$P_t = 2.5 \cdot P_{PL}$ W	[16]
Source	SMAD
Mini spacecraft (< 100 W total)	
Operating power is 2 to 3 times the payload power	
Source	SMAD
Communications satellites	
$P_t = 1.1148 \cdot P_{PL} + 348.1$ W	[17]
Source	Zandbergen (11 data points)
Payload power range	1-5 kW
R^2	0.9856
Comment	Actual values are within 10 % from the estimated value
$P_t = 1.17 \cdot P_{PL} + 56$ W	[18]
Source	Brown (10 data points)
Payload power range	100-1500 W
Meteorological satellites	
$P_t = 1.96 \cdot P_{PL}$	[19]
Source	Brown (8 data points)
Payload power range	100-450 W
Planetary vehicles/deep space probes	
$P_t = 332.93 \cdot \ln\left(P_{PL}\right) - 1046.6$ W	[20]
Source	Brown (3 data points)
Payload power range	75-250 W

Vehicle size

Table 5: Size estimation

All Spacecraft	
$V_{S/C} = 0.01 \cdot M_{Loaded}\ \text{m}^3$	[21]
Source	SMAD
Mass range	135-3625 kg
Slope range	0.005-0.05
Density range	20-172 kg/m^3
Average density	79 kg/m^3
Small satellites	
$V_{S/C} = \dfrac{M_{Loaded}}{\rho_{S/C}}\ \text{m}^3$	[22]
$\rho_{S/C} = -106.9 \cdot \ln\left(M_{Loaded}\right) + 922.5\ \ kg/m^3$	[23]
Source	Zandbergen
R^2	0.4591
Mass range	1-500 kg
Micro satellites	
$V_{S/C} = 0.0019 \cdot M_{Loaded}\ \text{m}^3$	[24]
Source	TU-Delft, Aas
Slope range	0.0006-0.005
Density range	194-1584 kg/m^3
Mass range	1-50 kg
Deep space probes	
$V_{S/C} = 0.0044 \cdot M_{Loaded}\ \text{m}^3$	[25]
Source	Zandbergen
Slope range	0.0022-0.0167
Density range	60-458 kg/m^3
Average density	245 kg/m^3
Mass range	92-1062 kg

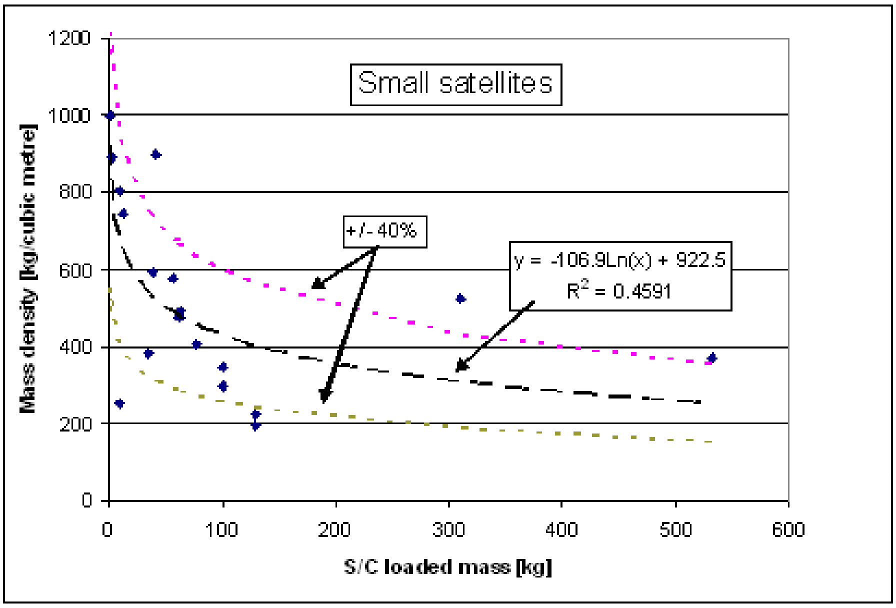


Figure 1: Spacecraft mass density versus spacecraft loaded mass for small satellites [Zandbergen]

Vehicle cost

Table 6: Cost estimation

Spacecraft (general)	
$C_{S/C} = 0.3531 \cdot (M_{SC})_{Dry}^{0.839}$ FY2000 M$ [26]	
Source	Zandbergen
Mass range	40-2350 kg

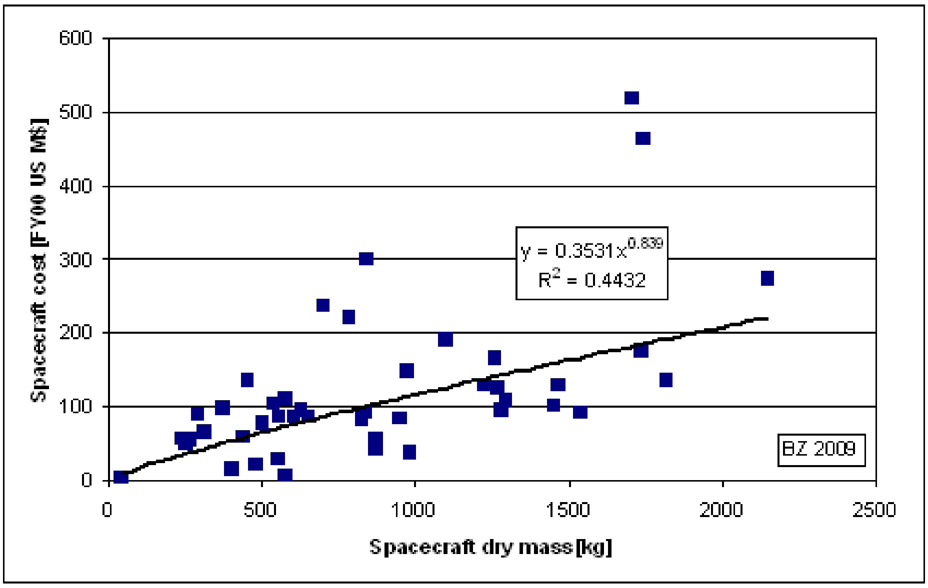


Figure 2: Spacecraft cost versus spacecraft dry mass [Zandbergen]

GEO Communications satellites (Commsats)	
$C_{S/C} = -0.0673 \cdot (M_{SC})_{Dry} + 190.1$ FY2000 k$/kg [27]	
Source	Zandbergen
Mass range	400-1800 kg
R^2	0.5912
SEE	23%
Navigation satellites (Navsats)	
$C_{S/C} = 40 - 45$ FY2000 k$/kg [28]	
Source	Zandbergen
Mass range	400-1000 kg
Earth Observation satellites (EOSats; optical imaging)	
$C_{S/C} = 0.0264 \cdot (M_{SC})_{Dry} + 192.95$ FY2000 k$/kg [29]	
Source	Zandbergen
Mass range	250-2000 kg
R^2	0.276
SEE	13.8%

Vehicle reliability

Table 7: Reliability estimation

General	
$R = (e)^{-(\lambda \cdot t)}$ [30]	
λ is failure rate and t is operational life (not storage life).	
Spacecraft (total)	
λ	0.056-0.139

Vehicle reliability may also be determined based on independent estimates of payload and spacecraft bus reliability using:

Table 8: Reliability estimation details

General	
$R_{SC} = R_{payload} \times R_{bus}$ [31]	
Here $R_{payload}$ and R_{bus} can be estimated using the relation [30] and the failure rate data as given below.	
Communication Payloads	
λ	0.003-0.032
Other Payloads	
$\lambda = 0.667 \times \lambda_{bus}$	
Spacecraft bus	
λ	0.009-0.053

Vehicle development time

ESA/NASA spacecraft:

- Phase A/0: 4 months with an estimated SSD of 2 months.
- Phase B: 13.8 months (~1 year) with an SSD of 5.4 months (~0.5 year).
- Phase C/D: 47.7 months (or ~4 years) with a SSD of 14.2 months (~ 1 year)

Vehicle life

Earth orbiting spacecraft:

- LEO S/C: up to 5-7 year
- MEO S/C: up to 5-10 year
- GEO S/C: up to 10-15 year

Example spacecraft vehicle estimation

In this section an example spacecraft sizing is performed. For this example we have an Earth observation spacecraft with the following data:

- Payload
  - Earth observation camera
  - Mass: 300 kg
  - Power: 280 W average, 790 W peak
  - Dimensions: 1.5 m x 1 m x 0.5 m
- Mission:
  - Life 10 yr
  - Maneuvering: Δv = 800 m/s (incl. 100 m/s for margin + reaction wheel unloading)
  - No separate kick stage needed. Vehicle is injected into final orbit by launcher
- Launcher
  - Maximum diameter under fairing: 4 m

(3.32)To limit ourselves, we will focus on estimating vehicle mass and vehicle power only. It is advised that you first try for your own and then check your answer with the answers given in Table 9. When doing so, you'll probably experience that not all the answers you calculated are the same as the answers given in this table. The reason for this will hopefully be clear to you after you have read the accompanying text.

Table 9: Estimated vehicle mass, vehicle power and vehicle size for the given satellite

Vehicle mass			
Source	TU-Delft (16 data points) – Earth orbiting S/C	Brown (46 data points) – Earth orbiting S/C	Larson & Wertz – Earth orbiting S/C
Loaded mass	1134 kg	-	-
Dry mass	-	1440 kg	999 kg
Propellant mass	-	440 kg	305 kg
Loaded mass	1134 kg	1880 kg	1304 kg
Vehicle power			
Method	Brown – Meteorological satellites	Brown – Other satellites	SMAD – Large satellites
Total power (based on average power)	549 W	438 W	518 W
Total power (based on peak power)	1548 W	1015 W	1462 W

Vehicle mass

Vehicle mass is estimated using three different methods. First the loaded mass is estimated using the Formula [2] given in Table 1, which gives in a loaded mass of 1134 kg. Next the loaded mass is estimated by first estimating the dry mass and the propellant mass after which the two are added together. Two different relationships are used to estimate the vehicle dry mass. The first relationship is given by Formula [6] and the second by Formula [7], see Table 2. Formula [6] is taken from [Brown (46 data points)], and Formula [7] is taken from [Larson & Wertz]. Using the given average values for 'a' one gets a dry mass of 1440 and 999 kg respectively. Now we add to this estimate the propellant mass. Propellant mass in this case is estimated based on the use of a propulsion system with a specific impulse of 300 seconds. Results are shown in Table 9.

When considering the results for the loaded mass, we find that the loaded mass estimate ranges from about 1130 kg to 1880 kg, which is quite a range. This more or less clarifies why for such estimates, the estimation accuracy is considered quite low and that for our design our estimate might easily be off by 50%. However, as long as we determine the mass of our vehicle concepts using the same estimation formula, we can expect that at least the order of the concepts in terms of mass remains the same.

Vehicle power

For the estimation of vehicle power Table 4 is used. However, which value for payload power should be used (average or peak)? Essentially both can be used, but you should consider that the relation has been derived for photovoltaic systems. In most cases such systems have a battery system that provides for power during eclipses and when peak power is needed. This means that photovoltaic systems most of the time are designed to deliver average power. Based on this reasoning, it is considered better to use average power than peak power for estimating the total power needed. However, if you consider designing a power system without any means of storing excess power and or providing for peak power, than it might be better to design for peak power.

We know that the satellite is an Earth observation satellite. To estimate vehicle power we now have three estimation relationships [19], **Error! Reference source not found.** and [15] that could be used. The results are given in Table 9. Comparing the values found, we find that the three values are reasonably comparable, thereby lending credibility to our estimates.

Appendix D: Spacecraft subsystem level estimating relationships for mass, power, etc.

Introduction

This document provides estimation relationships for spacecraft subsystems, see section 2, but first we provide typical data used for determining such relations.

1. *S/C subsystem data*

The spacecraft subsystems all contribute to the spacecraft (platform) mass. Table 1 and table 2 provide mass distribution data expressed as percentages of S/C dry mass for a number of large S/C.

Table 1: Mass distribution of some large GEO telecommunications satellites [SMAD, SSE]

	Percentage of Spacecraft Dry Mass by Subsystem							Dry Mass
	Payload	Structures	Thermal	Power	TT&C	ADCS	Propulsion	(kg)
FLTSATCOM 1-5	26.54	19.26	1.75	38.53	2.98	7.01	3.94	849.6
FLTSATCOM 6	26.38	18.66	1.99	39.39	2.99	6.77	3.83	870.9
FLTSATCOM 7-8	32.80	20.80	2.14	32.75	2.50	5.68	3.34	1041.9
DSCS II	23.02	23.50	2.77	29.32	6.97	11.46	2.96	475.9
DSCS III	32.34	18.18	5.56	27.41	7.23	4.35	4.09	867.3
NATO III	22.12	19.29	6.51	34.74	7.51	6.33	2.43	320.4
INTELSAT IV	31.24	22.31	5.14	26.49	4.30	7.41	3.14	532.8
INTELSAT V	28.85	21.21	3.21	22.44	3.45	9.00	11.84	835.0
INTELSAT VI	37.60	17.94	3.08	25.40	4.74	4.14	7.10	1779.0
TDRSS	24.56	28.03	2.78	26.36	4.07	6.17	6.92	1565.7
GPS Blk 1	20.49	19.85	8.70	35.77	5.84	6.16	3.61	479.1
GPS Blk 2, 1	20.15	25.13	9.86	30.97	5.20	5.41	3.29	699.1
GPS Blk 2, 2	23.02	25.37	11.03	29.44	3.10	5.25	2.68	858.0
P80-1	41.06	19.00	2.35	19.92	5.21	6.33	6.13	1704.4
DSP 15	36.91	22.53	0.48	26.94	3.84	5.51	2.23	2114.9
DMSP 5D-2	29.85	15.63	2.79	21.48	2.46	3.07	7.42	814.6
DMSP 5D-3	30.45	18.41	2.87	28.97	2.02	2.92	8.66	1012.3
Average	28.7	20.9	4.3	29.2	4.4	6.1	4.9	
Standard Deviation	6.2	3.2	3.1	5.6	1.7	2.1	2.7	

TT&C: Telemetry, Telecommand & Communications
ADCS: Attitude Determination and Controls Subsystem

Table 2: Mass distribution of some large GEO telecommunications satellites [MediaGlobe]

Spacecraft		Percentage of Spacecraft Dry Mass by Subsystem							Dry Mass
Name		Payload	Structure	Thermal	Power	TT&C	ADCS	Propulsion	(kg)
1	ANIK E	27.6	22.8	4.7	28.7	2.9	3.9	9.4	1270
2	Arabsat (not 2)	21.4	15.8	5.3	30.9	5.1	11.6	10.0	573
3	Astra 1B	30.0	16.2	4.5	30.7	2.3	6.2	10.2	1179
4	DFS Kopernikus	24.1	18.4	4.1	30.8	4.4	7.2	11.0	656
5	Fordsat	28.9	19.5	5.0	33.2	0.9	7.4	5.1	1094
6	HS 601	49.8	12.2	3.1	19.3	4.7	4.4	6.5	1459
7	Intelsat VII	30.8	17.3	6.7	25.8	1.0	10.1	7.6	1450
8	Intelsat VIIA	28.8	15.4	6.9	27.4	0.9	9.1	7.5	1823
9	OLYMPUS	28.5	21.6	5.2	27.4	3.0	5.2	9.2	1158
10	SATCOM K3	19.0	17.6	4.4	35.6	3.5	6.7	13.2	1018
11	TELSTAR 4	24.1	10.9	5.6	35.0	4.8	4.4	6.2	1621
	Average	28.4	17.1	5.0	29.5	3.1	6.9	8.7	
	Standard Deviation	8.0	3.6	1.1	4.6	1.6	2.5	2.4	

TT&C: Telemetry, Telecommand & Communications
ADCS: Attitude Determination and Controls Subsystem

The data clearly shows that next to the payload, the structures and power subsystem are the most important contributors to spacecraft mass.

Table 3 provides mass distribution data of specific small satellites. Data are again given as percentages of S/C dry mass.

Table 3: Small satellite mass data [Zandbergen]

S/C name	Application	Dry mass (kg)	Percentage of Spacecraft Dry Mass by Subsystem								
			Payload	Structures	Thermal	Power	TT&C	ADCS	Propulsion	Harness	C&DH
Orsted	Science	56.3	22.9	28.8	0.9	16.9	10.3	7.3	0.0	5.5	7.5
Freya	Science	216.9	33.7	22.5	2.4	18.7	2.6	5.9	8.2	0.0	6.0
SAMPEX	Science	160	32.5	23.1	2.5	20.0	3.1	6.3	0.0	5.0	7.5
ANS	Science	129.3	33.2	29.4	0.9	8.7	5.4	13.5	0.0	0.0	6.0
Viking	Science	289.6	16.8	16.1	2.7	6.9	3.1	3.5	5.8	0.0	0.0
Bird	Science	77.74	30.0	25.9	8.2	14.7	2.4	9.3	0.0	0.0	0.9
NATO III	Comm.	316.6	22.1	19.3	6.5	34.7	7.5	6.3	2.4	0.0	0.0
Gurwin II	Techn. test	47.0	14.0	36.0	0.0	12.3	4.9	7.3	0.0	3.4	22.1
Temisat	Comm.	41.9	25.8	19.8	0.0	36.8	0.0	1.7	0.0	9.1	6.9
ORBCOMM	Comm.	47.5	19.4	15.3	1.9	22.5	0.0	6.8	4.0	0.0	6.8
PoSAT-1	Comm. / test	50.2	12.2	13.7	0.0	32.9	11.2	20.9	0.0	3.0	6.2
Hausat-1	Techn. test	1	16.0	27.0	3.0	21.0	14.0	6.0	0.0	-	4.5
Delfi C3	Techn. test	2.9	16.7	26.5	2.3	15.4	12.1	5.2	0.0	6.7	5.1
Average			22.7	23.3	3.1	20.1	7.0	7.7	5.1	5.4	7.2
Average (total is 100%)			**22.3**	**23.0**	**3.1**	**19.8**	**6.9**	**7.6**	**5.0**	**5.3**	**7.1**
Average (excl. propulsion)			23.5	24.2	3.2	20.8	7.2	7.9	0.0	5.6	7.5

TT&C: Telemetry, Telecommand & Communications
ADCS: Attitude Determination and Controls Subsystem
C&DH: Command & Data Handling

Like for large S/C, we find that next to the payload, the power and the structures subsystem contribute most heavily to spacecraft mass. Notice that not for some S/C some subsystems do not contribute to mass. This does not mean that these spacecraft are without these subsystems, but rather that the mass has been included in some other subsystem. For instance mass of harness may be included in power subsystem mass and or the mass of the C&DH subsystem. For Spacecraft with passive thermal housekeeping, the mass of the thermal subsystem sometimes is included in the structures subsystem. Propulsion system mass sometimes is included in the mass of the ADCS. With respect to the latter though, Temisat, Hausat and Delfi C3 all lack a propulsion system on board.

Spacecraft subsystem power usage data

Table 1 provides data on power usage of the various S/C subsystems for specific spacecraft.

Table 1: Average power distribution (in Watt EOL) for several large geostationary telecom. satellites [MediaGlobe]

Satellite	Payload	TT&C	ADCS	Thermal	Propulsion	Power Generation & Distribution	Charging	Total Load (W)
ANIK E	86.2%	1.2%	0.8%	2.9%	NA	0.7%	8.2%	3482
Arabsat (not 2)	72.7%	2.8%	9.2%	6.7%	NA	1.3%	7.3%	1362
Astra 1B	76.6%	1.5%	1.0%	3.8%	NA	2.4%	14.7%	2790
DFS Kopernikus	63.5%	2.0%	2.8%	16.6%	NA	3.3%	11.9%	1412
Fordsat	79.1%	1.7%	4.2%	3.0%	NA	1.3%	10.8%	3110
HS 601	79.4%	2.4%	2.1%	8.4%	NA	0.9%	6.9%	3350
Intelsat VII	72.3%	1.1%	6.3%	7.4%	0.2%	2.3%	10.5%	3569
Intelsat VIIA	79.1%	0.6%	5.0%	4.9%	0.1%	1.2%	9.2%	4567
OLYMPUS	75.9%	1.6%	4.1%	10.1%	NA	1.2%	7.1%	2832
SATCOM K3	81.6%	1.4%	0.9%	3.0%	0.0%	1.6%	11.5%	3150
TELSTAR 4	84.9%	1.7%	1.3%	2.4%	NA	0.7%	8.9%	5673
Average %	77.4%	1.6%	3.4%	6.3%	0.2%	1.5%	9.7%	
STD	6.37%	0.61%	2.66%	4.29%	0.09%	0.81%	2.42%	

NA) Not Available, most likely incorporated in other subsystem

It should be clear from this data that for large GEO communications satellites the payload consumes most power (roughly 75%). The remainder is used to power the various spacecraft subsystems. Of these the most power hungry system is the power subsystem and more important the battery charging part of the power subsystem. Another important power consumer is the thermal subsystem.

Power data must be considered carefully as when collecting data from literature, it is sometimes unclear whether peak or average power values are listed. These values may differ considerably, depending on the duty cycle of the apparatus considered. For instance a propulsion system may work for only 2% of the total mission time. During that time power consumption might be considerable, but average power consumption is much less. In addition, it is not always clear if the values given hold for EOL or BOL.

Spacecraft subsystem reliability data

Spacecraft are subject to failure. Some of these failures can be attributed to the payload and some to the spacecraft bus or platform supporting the payload. For design purposes, it is interesting to determine the contribution of the various S/C subsystems to bus/platform failures and hence to spacecraft failures. For instance, according to [Sarsfield], the area of greatest concern with respect to failures on board of spacecraft is the performance of mechanical systems. The performance and reliability of electrical and electronic components have improved dramatically in recent years. The design and development of mechanical systems, however, have not advanced in parallel. Many of the most serious recent spacecraft anomalies can be traced to mechanical system failures. Some examples of recent Spacecraft mechanical failures are outlined in the next table taken from the work of Sarsfield.

Table 1: Example of Mechanical Failures in Recent Spacecraft [Sarsfield]

Mission	Event	Impact	Likely Failure Mode
Mars Observer	Propulsion system failure	Loss of Spacecraft	Leakage and ignition of hypergolic propellants—rupture of high-pressure lines
Galileo	Stuck high-gain antenna	Degraded performance	Excessive friction due to misalignment in antenna restraint pins
Alexis	Damaged solar array	Degraded performance	Attachment bracket broke free after deployment
Mars Global Surveyor	Failure to latch solar array	Modification of flight plan	Structural failure of solar array damper arm attach fitting

[Sultan] has investigated in detail the distribution of reported failures (most likely only the more serious ones are reported) in spacecraft over the period 1995 up to and including 2000. Sultan found that on average 40% of the failures are attributed to the payload and the remainder to the S/C bus, i.e. the platform. The distribution of the bus failures over the various bus subsystems is given in *Figure 1*.

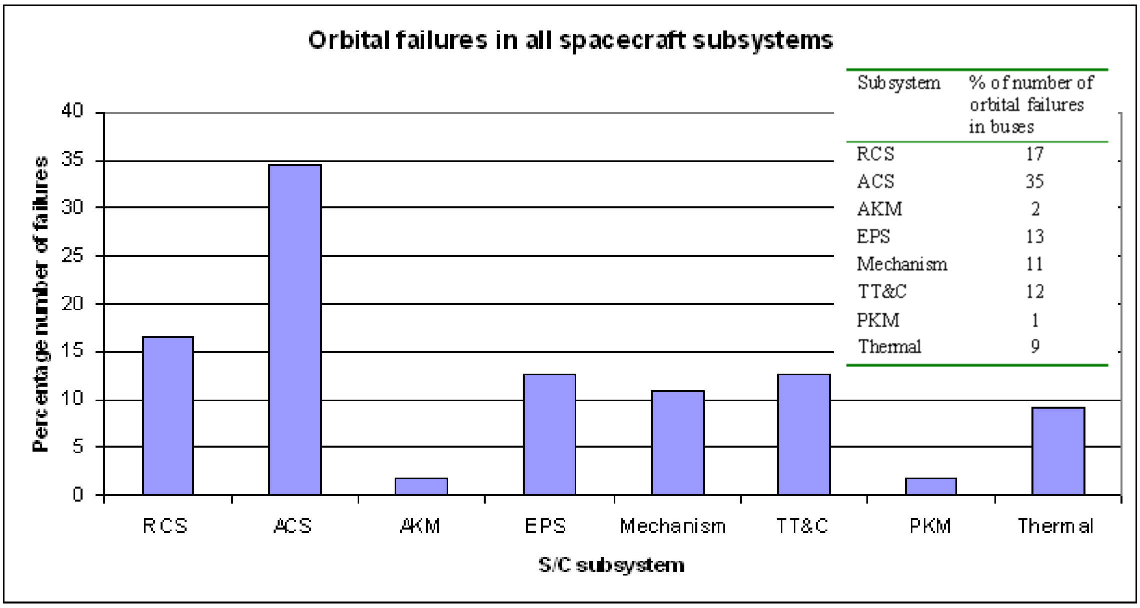


Figure 1: Percentage number of failures distributed over the various spacecraft subsystems

RCS: Reaction Control Subsystem
ACS: Attitude Control Subsystem
EPS: Electric Power Subsystem

TT&C: Tracking, Telemetry & Command
AKM: Apogee Kick Motor
PKM: Perigee Kick Motor

The data seems to be somewhat contradictory to the result found by Sarsfield, but some components in the ACS systems, like reaction wheels, can also be considered mechanisms.

[Tafazoli] performed an identical investigation as Sultan, based on a study of 156 failure cases and using a slightly different break down, see *Figure 2*.

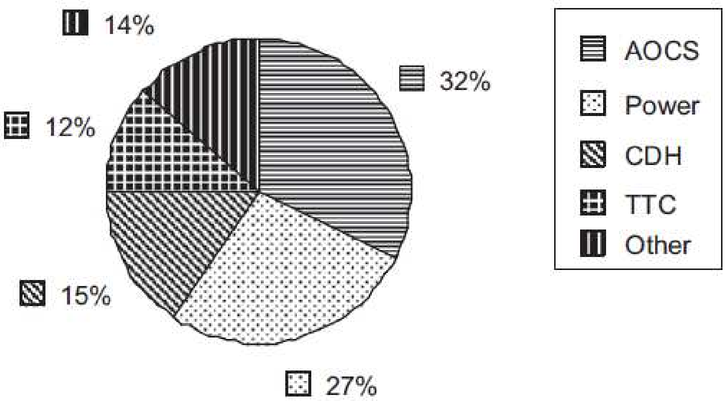


Figure 2: Spacecraft subsystem failures as a percentage of total number of failures [Tafazoli]

The figure shows the failure breakdown for the different spacecraft subsystems: The category "other" regroups MECH, payload and miscellaneous subsystem failures. We observe that 59% of all failures affect AOCS and power subsystems.

The large difference in the failure percentages for power between Sultan and Tafazoli cannot be explained in a satisfactory way, but might be attributed to differences in spacecraft considered and/or in the subsystems and/or in what is considered a failure.

Now using the percentage failure data presented above, we can determine failure rates of the various subsystems. Once failure rates are known and assuming constant failure rates (in time), we can estimate subsystem reliability using the relation given in *Table 9*.

Table 2: Reliability estimation

Spacecraft subsystem i
$R_i = (e)^{-(\lambda_i \cdot t)}$ [1]

Here subscript i indicates a specific subsystem, λ_i indicates the failure rate of the subsystem i and t indicates the time period considered.

For instance, given a total number of 565 S/C failures over a 6-year period, see earlier reported data based on data from Jane's, it could be argued that in 119 cases (0.35 x 0.60 x 560; here the factor 0.60 indicates that only 60% of the total number of failures is attributed to the bus with the remainder attributed to the payload) the ACS system is the cause of failure. This comes down to 19.8 failures per year. Estimating 800 operational S/C at any one time over the period investigated by Sultan, this leads to 0.0248 ACS failures per S/C per year. The same reasoning applied to the RCS gives about 9.6 failures per year or 0.012 RCS failure per S/C per year.

Suppose now that we are aiming for an RCS life of 10 years. It follows for the reliability of the RCS:

$$\text{RCS reliability: } R_{RCS} = e^{-(0.0012*10)} = 0.988$$

In an identical way also the reliability of the other subsystems can be estimated.

Several notes of caution must be taken into account.

1. AKM and PKM as included in the Sultan data are not always included in every S/C bus. This of course affects the ratio of failures found.
2. Results may differ based on the data used (Sultan, Tafazoli or other). Hence the designer should use proper margins to account for such differences.

Another approach sometimes taken is to use component failure rate data, see *Table 3*, to determine subsystem failure rates (MTTF is mean time to failure; MTTF = $1/\lambda$). This, however, is considered beyond the scope of this document.

Table 3: Spacecraft component reliability data

System	Component		Company	Reliability [-]	Life [yr]	Failure rate/yr	Failure rate/hr	MTTF [hr]
Power	Array	5kW		0,999	5			
	Batteries		SAFT					3,00E+06
ADS	Earth sensor	ESG					6,19E-07	
		Static GEO		0,998	5			
		PESA					2,50E-06	
	Star sensor	Star Mapper	TPD			0,0200		
		Rosat star tracker	SIRA Ltd.	0,912	1,5			
		SED star tracker	Sodern				3,80E-06	
	Horizon sensor	NOHS	Lockheed Martin				8,00E-07	
	Inertial navigation Unit	SIRU	Delco					6,00E+06
		SIGI	Honeywell					2,00E+04
	Gyro	Regys 10	Sagem					2,00E+05
		Regys 3S	Sagem	0,99	15			
	GPS receiver	Viceroy	Motorola	0,91	8			
Actuators	Torqrods			0,99997	1			
	Momentum wheel	Magnetic bearing (MWX)		0,89	10			
Communications	TWT		Thomson					*3,00E+06*
C&DH	Space processor	1750A	Litton Applied Techn.	0,95	5			
		C&DH subsystem	Spectrum Astro	0,95	1			
		RISC		0,95	5			
		DSBC	Honeywell	0,9935	12			
	Memory	Solid state recorder	Lockheed Martin	0,99	5			
		Magnetic disk	Spectrum Astro	0,9	5			1,00E+05

2. *Estimation relationships*

Various estimation relationships exist that allow for estimating important characteristics of the various S/C subsystems. In the next few sections various such estimation relationships are presented.

Spacecraft subsystem mass estimation relationships

Subsystem mass of medium to large spacecraft and small spacecraft can be estimated using:

$$M_i = \left(\frac{\%}{100}\right) \cdot M_{dry} \qquad [2]$$

Here subscript i indicates a specific subsystem, M_i gives the estimated mass of subsystem i and % indicates the percentage value as indicated in the *Table 4* for medium to large spacecraft and *Table 5* for small spacecraft.

Table 4: Medium to large spacecraft mass estimation subsystems [SMAD]

Subsystem	Subsystem mass	Range
Propulsion	4.75 % of M_{Dry}	2.5-7 % of M_{Dry}
ADCS	6 % of M_{Dry}	3-9 % of M_{Dry}
Communications	4.75 % of M_{Dry}	2.5-7 % of M_{Dry}
Thermal	8.5 % of M_{Dry}	2-15 % of M_{Dry}
Power	30 % of M_{Dry}	20-40 % of M_{Dry}
Structures	20 % of M_{Dry}	15-25 % of M_{Dry}

Table 5: Small spacecraft mass estimation subsystems (mass below 500 kg) [Zandbergen]

Subsystem	Subsystem mass
Propulsion	6.1 % of M_{Dry}
ADCS	9.6 % of M_{Dry}
Communications	9.2 % of M_{Dry}
Thermal	3.8 % of M_{Dry}
Power	24.5 % of M_{Dry}
Structures	29.3 % of M_{Dry}
Harness	7.2 % of M_{Dry}
Command & Data Handling	10.4 % of M_{Dry}

In the *Table 4* no data are included for the Command & Data Handling system. This does not mean that this system is not present, but that the data probably has been included in some other subsystem.

The data in *Table 5* has also been used to generate a pie chart. From this chart it is quite clear that the main contributors to small spacecraft mass are the power and the structures subsystem.

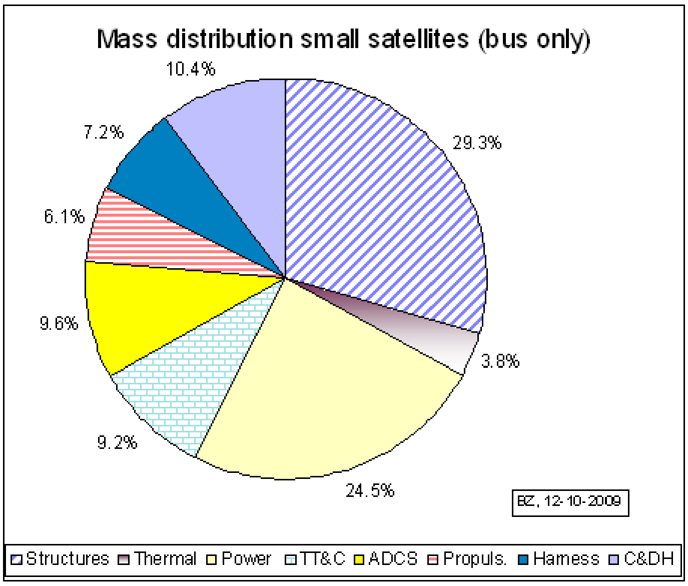
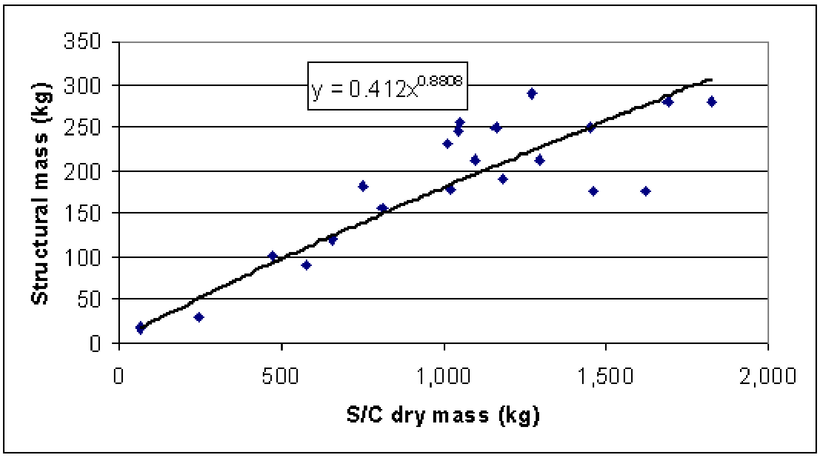


Figure 3: Small spacecraft mass distribution [Zandbergen]

The next figure shows a more detailed estimation relationship as determined for the structures subsystem. In this relationship it is taken into account that the structural mass may differ depending on the mass of the vehicle, whereas in the estimations using average payload mass fractions this is neglected. The relationship shown in the figure is based on 22 data points for spacecraft in the (dry) mass range 50-1750 kg. As a measure for the spread seen in the figure, the Standard Error of Estimate has been determined. It follows SEE is 21.7%.



Spacecraft power estimation relationships

For small and medium large spacecraft, subsystem power may be estimated using:

$$P_i = \left(\frac{\%}{100}\right) \cdot P_{total} \qquad [3]$$

Here subscript i indicates a specific subsystem, P_i indicates the estimated power usage of subsystem I, % indicates the percentage value of subsystem I as indicated in *Table 6*.and P_{total} is total spacecraft power (considered here as a known).

*Table 6: Power estimation of subsystems [SMAD] (*Includes conversion and line losses)*

Spacecraft size:	**Micro**	**Small**	**Medium-large**
Spacecraft power:	**< 100 W total**	**~200 W**	**> 500 W**
Subsystem*		**Percentage of operating power**	
Payload	20-50 W	40	40-80
Propulsion	0	0	0-5
Attitude control	0	15	5-40
Communications	15 W	5	0-50
C&DH	5 W	5	0-50
Thermal control	0	5	0-5
Electric power	10-30 W	30	5-25
Structure	0	0	0
Margin	5-25 % of power based on design maturity		

For micro-spacecraft with total power level below 100 W, *Table 6* provides specific values of power usage for the various subsystems.

From the work of [Brown] the following table, providing power allocation guidelines for different spacecraft can be obtained.

Table 7: Subsystem power allocation guide [Brown]

	Percentage of subsystem total			
Subsystem	Comsats	Metsats	Planetary	Other
Thermal control	30	48	28	33
Attitude control	28	19	20	11
Power	16	5	10	2
CDS	19	13	17	15
Communications	0	15	23	30
Propulsion	7	0	1	4
Mechanisms	0	0	1	5

Spacecraft system cost estimation relationships

Subsystem cost of small spacecraft can be estimated using:

$$C_i = \left(\frac{\%}{100}\right) \cdot C_{SC} \quad [4]$$

Here subscript i indicates a specific subsystem, C_i gives the estimated cost of subsystem I, % indicates the percentage value as indicated for subsystem i in *Table 8* and C_{sc} is total S/C cost (considered here as a known).

Table 8: Small spacecraft cost estimation subsystems (mass below 500 kg; 4 data points) [Zandbergen]

Subsystem	Mass subsystem
ADCS	13 % of C_{SC}
Communications	12 % of C_{SC}
Thermal	2 % of C_{SC}
Power	30 % of C_{SC}
Structures	14 % of C_{SC}
Harness	1 % of C_{SC}
Command & Data Handling	28 % of C_{SC}

The data in *Table 8* have been used to generate a pie chart.

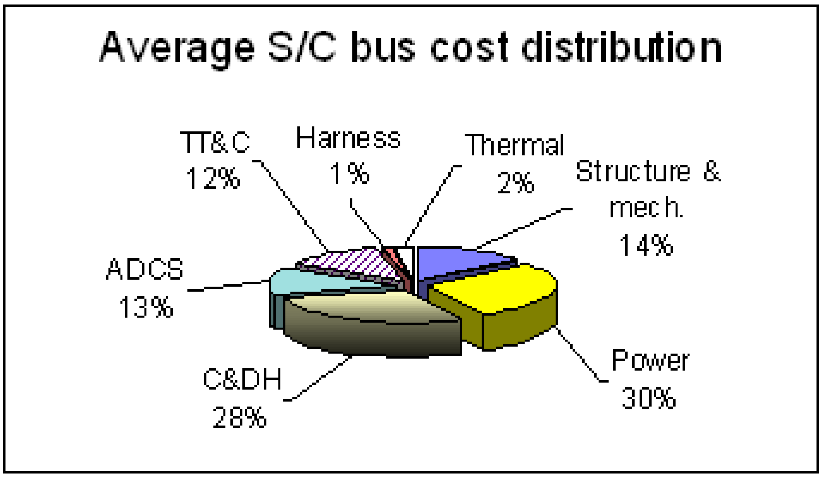


Figure 4: Small spacecraft cost distribution (4 S/C) [Zandbergen]

The pie chart clearly shows that the power and the C&DH subsystem contribute most to the cost. In contrast, the electrical harness (power cables and data lines) and the thermal system are low cost subsystems.

Spacecraft subsystem reliability estimation relationships

In this section subsystem reliability is estimated using the relation given in *Table 9*.

Table 9: Reliability estimation

Spacecraft subsystem i	
$R_i = (e)^{-(\lambda_i \cdot t)}$	[5]

Here subscript i indicates a specific subsystem, λ_i indicates the failure rate of the subsystem i and t indicates the time period considered.

The set of relations presented is based on the earlier presented data of [Sultan] and on average 0.1188 serious spacecraft failures per year and taking of which 60% are considered bus failures with the remainder being payload failures [Sultan].

Table 10: Spacecraft subsystem failure rate data based on distribution of failures according to Sultan

Subsystem	Failure rate (failures/spacecraft/yr)
RCS	0.012
ADCS	0.0248
AKM	0.0014
EPS	0.0092
Mechanism	0.0078
TT&C	0.0085
PKM	0.0007
Thermal	0.0064

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Appendix E: Some statistics

Averaging and standard deviation

Large amounts of data are often compressed into more easily assimilated summaries, which provide the user with a sense of the content, without overwhelming him or her with too many numbers. There are a number of ways in which data can be presented. One approach is to estimate "summary statistics" for the data. For a data series, X_1, X_2, X_3, ... X_n, where n is the number of observations in the series, the most widely used summary statistics are as follows –

- mean (μ), which is the average of all of the observations in the data series

$$\text{Mean} = \mu_x = \sum_{i=1}^{i=n} \frac{x_i}{n}$$

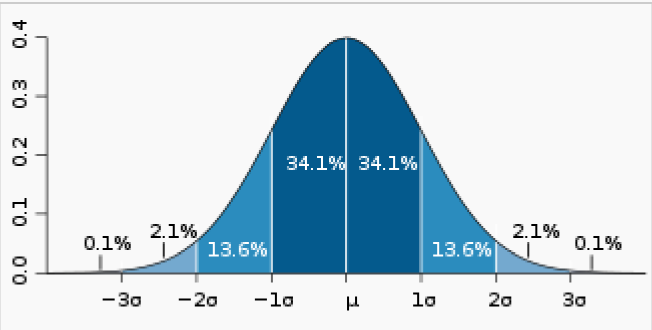
- median, which is the mid-point of the series; half the data in the series is higher than the median and half is lower
- variance, which is a measure of the spread in the distribution around the mean, and is calculated by first summing up the squared deviations from the mean, and then dividing by either the number of observations (if the data represents the entire population) or by this number, reduced by one (if the data represents a sample)

$$\text{Variance} = \sigma^2{}_x = \frac{1}{n-1} \cdot \sum_{i=1}^{i=n} (x_i - \mu)^2$$

- Standard deviation is the square root of the variance.

Normal distribution and confidence bounds

An important distribution is the **normal distribution** or **Gaussian distribution.** In probability theory and statistics this is a continuous probability distribution that describes data that clusters around a mean or average. The graph of the associated probability density function is bell-shaped, see figure below, with a peak at the mean, and is known as the **Gaussian function** or **bell curve**.



Dark blue is less than one standard deviation from the mean. For the normal distribution, this accounts for about 68% of the set (dark blue) while two standard deviations from the mean (medium and dark blue) account for about 95% and three standard deviations (light, medium, and dark blue) account for about 99.7%.

The normal distribution can be used to describe, at least approximately, any variable that tends to cluster around the mean. For example, the heights of adult males in Europe are roughly normally distributed, with a mean of about 1.78 m. Most men have a height close to the mean, though a small number of outliers have a height significantly above or below the mean. A histogram of male heights will appear similar to a bell curve, with the correspondence becoming closer if more data is used.

About 68% of values drawn from a normal distribution are within one standard deviation $\sigma > 0$ away from the mean μ; about 95% of the values are within two standard deviations and about 99.7% lie within three standard deviations. This is known as the "68-95-99.7 rule" or the "empirical rule."

What are Confidence Bounds?

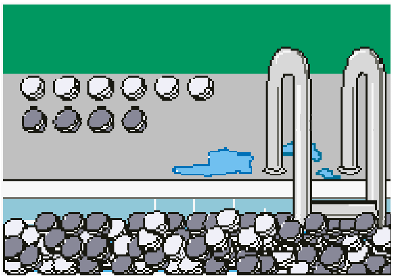
One of the most confusing concepts to a novice engineer is estimating the precision of an estimate or measurement. This is an important concept in the field of engineering, leading to the use of confidence intervals (or bounds). In this section, we will try to briefly present the concept in relatively simple terms but based on solid common sense.

The Black and White Marbles

To illustrate, consider the case where there are millions of perfectly mixed black and white marbles in a rather large swimming pool and our job is to estimate the percentage of black marbles. The only way to be absolutely certain about the exact percentage of marbles in the pool is to accurately count every last marble and calculate the percentage. However, this is too time-0 and resource-intensive to be a viable option, so we need to come up with a way of estimating the percentage of black marbles in the pool. In order to do this, we would take a relatively small sample of marbles from the pool and then count how many black marbles are in the sample.

Taking a Small Sample of Marbles

First, pick out a small sample of marbles and count the black ones. Say you picked out ten marbles and counted four black marbles. Based on this, your estimate would be that 40% of the marbles are black.



If you put the ten marbles back in the pool and repeat this example again, you might get six black marbles, changing your estimate to 60% black marbles. Which of the two is correct? Both estimates are correct! As you repeat this experiment over and over again, you might find out that this estimate is usually between X1% and X2%, and you can assign a percentage to the number of times your estimate falls between these limits. For example, you notice that 90% of the time this estimate is between X1% and X2%.

Taking a Larger Sample of Marbles

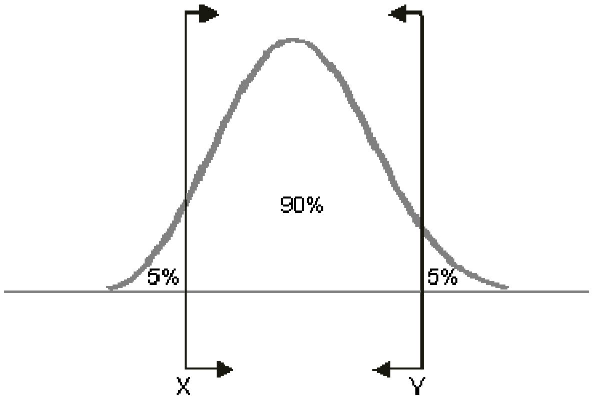
If you now repeat the experiment and pick out 1,000 marbles, you might get results for the number of black marbles such as 545, 570, 530, etc., for each trial. The range of the estimates in this case will be much narrower than before. For example, you observe that 90% of the time, the number of black marbles will now be from Y1% to Y2%, where X1% < Y1% and X2% > Y2%, thus giving you a more narrow estimate interval. The same principle is true for confidence intervals; the larger the sample size, the more narrow the confidence intervals.

Confidence interval

If we perform ten identical tests on our units, and analyze the results, we will obtain slightly different results each time. However, by employing confidence bounds, we obtain a range within which these results are likely to fall. This range of plausible values is called a confidence interval. Confidence bounds are generally described as being one-sided or two-sided.

Two-Sided Bounds

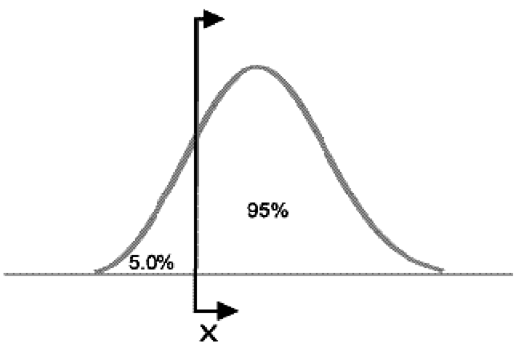
When we use two-sided confidence bounds (or intervals), we are looking at a closed interval where a certain percentage of the population is likely to lie. That is, we determine the values, or bounds, between which lies a specified percentage of the population. For example, when dealing with 90% two-sided confidence bounds of (X, Y), we are saying that 90% of the population lies between X and Y with 5% less than X and 5% greater than Y.

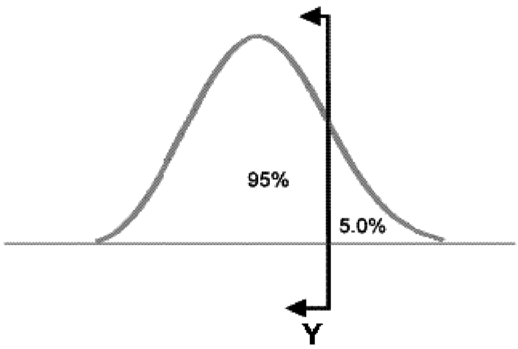


One-Sided Bounds

One-sided confidence bounds are essentially an open-ended version of two-sided bounds. A one-sided bound defines the point where a certain percentage of the population is either higher or lower than the defined point. This means that there are two types of one-sided bounds: upper and lower. An upper one-sided bound defines a point that a certain percentage of the population is less than. Conversely, a lower one-sided bound defines a point that a specified percentage of the population is greater than.

For example, if X is a 95% upper one-sided bound; this would imply that 95% of the population is less than X. If X is a 95% lower one-sided bound, this would indicate that 95% of the population is greater than X.





Care must be taken to differentiate between one- and two-sided confidence bounds, as these bounds can take on identical values at different percentage levels. For example, in the figures above, we see bounds on a hypothetical distribution. Assuming this is the same distribution in all of the figures, we see that X marks the spot below which 5% of the distribution's population lies. Similarly, Y represents the point above which 5% of the population lies. Therefore, X and Y represent the 90% two-sided bounds, since 90% of the population lies between the two points. However, X also represents the lower one-sided 95% confidence bound, since 95% of the population lies above that point; and Y represents the upper one-sided 95% confidence bound, since 95% of the population is below Y.

Most Likely Estimate (MLE) and Standard Error of estimate (SE)

A parametric relationship attempts to explain one variable, which is called the dependent variable, using the other variable, called the independent variable. Parametric relationships essentially can be of any form (linear, quadratic, power, log, etc.). For instance Excel allows for determining such estimating relationships. It seeks the line that fits the data best. Goodness of fit is expressed by R^2 value. The closer this value is to 1, the better the fit.

Another way of expressing the goodness of fit is to determine the Standard Error of estimate (SE):

$$SE = \sqrt{\frac{1}{n-m} \cdot \sum \left(\frac{y_i}{f(x_i)} - 1 \right)^2}$$

Here $\varepsilon_i = \frac{y_i}{f(x_i)}$ is referred to as multiplicative (or relative error) error, n is number of observed values, m is the number of parameters being estimated, f(x) is estimate and y is true or actual value of the parameter.

Dealing with uncertainty

In case of summing up various estimates like summing up the mass of the various subsystems to arrive at a total system mass, the standard error (or sample standard deviation) of the sum is given by the square root of the sum of the squares of the individual (absolute) errors, provided that the individual estimates are independent (uncorrelated data) from each other:

$$SE = \sqrt{\sum_{i=1} SE_i^2}$$

In case of dependent or perfectly correlated variables, the SE is given by:

$$SE = \sum_{i=1} SE_i$$

In case we are dealing with of SSD, identical rules apply.

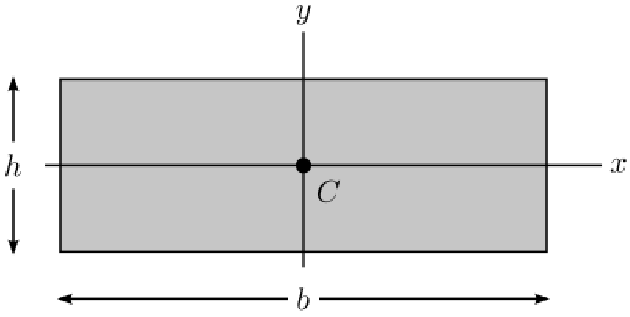
Example: For a S/C you have estimated a payload cost of 40 M€ with a standard error (or standard sample deviation) of ± 4 M€ and a bus cost of 30M€ ± 6 M€. Total cost of the S/C is 30 M€ + 40 M€ = 70 M€ with an uncertainty of:

1. *maximum 10 M€ in case a rise in cost of the payload leads to an equal rise in bus cost;*
2. *maximum $(4^2 + 6^2)^{0.5}$ = 7.2 M€ in case a rise in cost of the payload has no effect on the cost of the bus.*

Appendix F: Area and Mass Moments of Inertia

<u>Area</u> moment of inertia of some principal geometries:

Rectangular cross section



$$I_x = \frac{bh^3}{12}$$

- b = width (x-dimension),
- h = height (y-dimension)

$$I_y = \frac{hb^3}{12}$$

- b = width (x-dimension),
- h = height (y-dimension)

Circular cross section

$$I_0 = \frac{\pi}{64}D^4 = \frac{\pi}{4}r^4$$

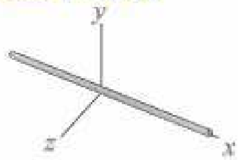
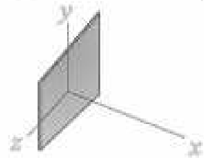
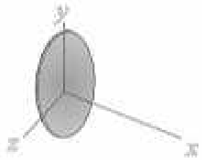
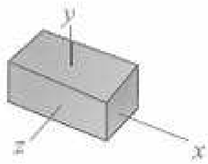
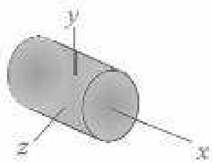
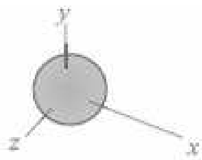
- D = diameter
- r = radius

Thin walled cylinder:

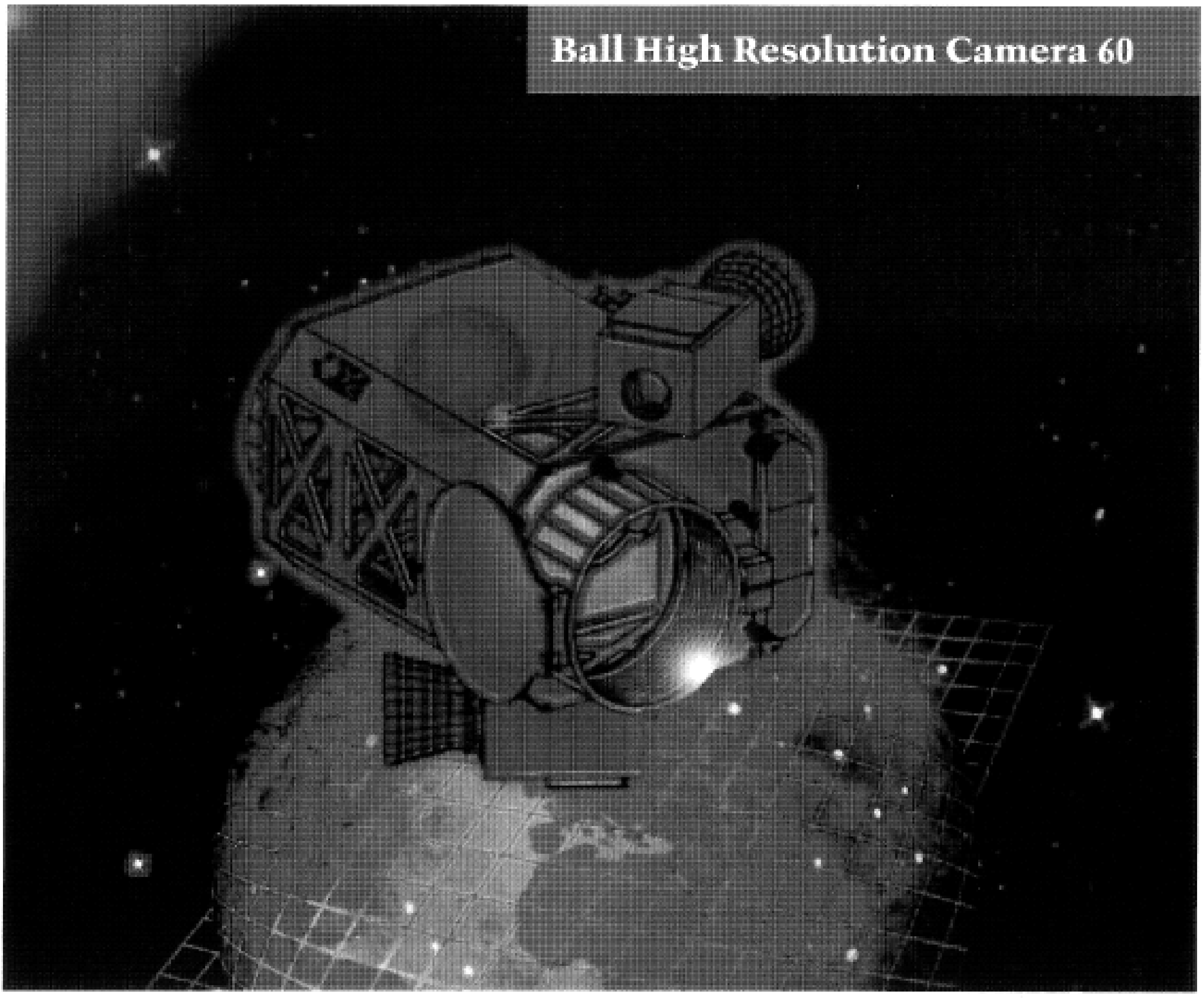
$$I_o = \pi \cdot r^3 \cdot t$$

- r is radius
- t is wall thickness

Principal Mass Moments of Inertia of Solid Geometrical Shapes

	I_x	I_y	I_z	
Slender Rod m = mass, l = length of rod	0	1/12 ml^2	1/12 ml^2	
Rectangular Plate m = mass, b = height of plate, c = width of plate	1/12 $m(b^2+c^2)$	1/12 mc^2	1/12 mb^2	
Thin Disk m = mass, r = radius of disk	½ mr^2	¼ mr^2	¼ mr^2	
Rectangular Prism m = mass, a = depth (x), b = height (y), c = width (z)	1/12 $m(b^2+c^2)$	1/12 $m(a^2+c^2)$	1/12 $m(a^2+b^2)$	
Circular Cylinder m = mass, l = length of cylinder, r = radius	½ mr^2	1/12 $m(3r^2+l^2)$	1/12 $m(3r^2+l^2)$	
Sphere m = mass, r = radius	2/5 mr^2	2/5 mr^2	2/5 mr^2	

Appendix G: Some Earth Observation instrument characteristics



Ball's High Resolution Camera (BHRC 60) is a state-of-the-art optical remote sensing payload that provides simultaneous one-meter class panchromatic and four-meter class multispectral imagery. This pushbroom implementation is specifically designed to cover broad areas without the need for repointing.

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BHRC 60 Performance Specifications

Item	BHRC 60 Performance
Spatial Resolution	Panchromatic: ~0.5 to 1.25 m ground sample distance (GSD) or 1.37 mrad Multispectral: ~2 to 5 m GSD or 5.47 mrad for 4 VNIR bands (Landsat-like) Optional IR capability
Ground Swath Width	2.12 deg cross track (14 to 34 km depending on altitude)
Data Acquisition Modes	Pushbroom imaging system
Operations	Simultaneous imaging in all bands; simultaneous data transmission capability available through optional equipment
Data Compression	Average 2 bits per pixel from 11 bit initial quantization
Calibration	<10 % absolute
Design Life	>5 years achieved with redundant architecture for orbits between 400 to 900 km from 0 degrees to sun synchronous
Onboard Storage Capacity	Optional equipment scalable up to 200 Gbits (equivalent to over 90 square images)
Communications Image Data	Optional 320 Mbps X-band transmitter and gimbaled antenna
Payload Mass	Total weight: 296 kg Total weight with options: 342 kg
Power Consumption	792 W when imaging (peak) <25 W non-imaging (orbital average)
Telescope Size	115 cm x 141 cm x 195 cm (rectangular)

Note: Numerical ranges reflect orbit altitude options (400 km to 900 km)

For your most demanding remote sensing needs, the BHRC 60 pushbroom camera provides the highest resolution and performance currently available on the market.

Image quality is paramount when extracting the information you need. That is why Ball Aerospace & Technologies Corp. has created a camera that combines high throughput for low-lighting conditions with an unobscured optical design for maximum clarity of objects of any size. The wide field of view instantly captures a broad area – an important feature when attempting to acquire information on objects whose location might be uncertain.

Control of the system is as easy as using your personal camera. You set the exposure time by selecting various levels of time-delay-integration (TDI) to maximize the signal over a wide range of sun angles. The data is automatically compressed to retain the maximum information content of each image. Now just point and turn it on. When you are finished, turn it off. The area covered by the BHRC 60 is limited only by the amount of onboard storage.

Data continuity is important. For links to the past, the BHRC 60 provides the same multispectral band passes as the first four bands of Landsat. For the future, the built-in redundancy and on-orbit performance tracking provide a long-life system for years to come.

As optional equipment to ease the task of integration into your spacecraft, the BHRC 60 can be ordered with solid-state recorders capable of simultaneous record and playback, an X-band transmission system tailored to meet international radiation restrictions, and star trackers mounted to the telescope structure for optimal precision of the geolocation of your images.

The BHRC 60 is part of the Ball Aerospace product line of telescopes and sensor options that span a range of spatial resolutions and spectral bands. Please contact us to discuss how our series of products can be combined to meet your specific requirements.

HIBRIS

The HIBRIS instrument, see figure, is an imaging spectrometer under development by Cosine B.V. (The Netherlands, Leiden). It actually consists of two different instruments integrated in one module; the near infrared hyperspectral imager (NSI) and a thermal infrared imaging spectrometer (TI). It is being designed for the BepiColombo mission to Mercury (ESA, 2009)**.** The specifications of the HIBRIS can be found in the next table. The mechanical layout of the HIBRIS can be found in the figure next to the text.

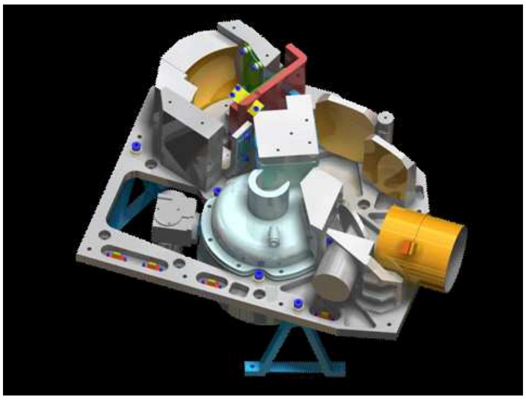

Table 1: Specification of the HIBRIS instrument (taken from Cosine Brochure)

HIBRIS	
Total Mass	7.1 kg
Size (LxWxH)	22x26x21 cm
Avg. Power consumption.	4.11 W
Near Infrared Hyperspectral Imager (NSI)	
Linear Variable Filter Ranges	700 to 1400 nm
	1400 to 2800 nm
	2800 to 5200 nm
Spectral Resolution	1% of Central λ
Spatial Resolution (at 600 km AGL)	50 m
FOV	14.5°
Thermal Imager (TI)	
Spectral Range	10 to 73 μ m
Spectral Resolution	7 μm
Spatial Resolution (at 600 km AGL)	715 m
FOV	6.9°

Appendix H: Earth Satellite Parameters

GENERAL PARAMETERS					
Alt (km)	Mean DENSITY (kg/m^3)	Circular Velocity (km/s)	Range to Horizon (km)	Period (min)	Max Eclipse (min)
0	1.2	7.905	0	84.49	42.24
100	4.79×10^{-7}	7.844	1,134	86.48	38.40
150	1.81×10^{-9}	7.814	1,391	87.49	37.76
200	2.53×10^{-10}	7.784	1,610	88.49	37.28
250	6.24×10^{-11}	7.755	1,803	89.50	36.90
300	1.95×10^{-11}	7.726	1,979	90.52	36.59
350	6.98×10^{-12}	7.697	2,142	91.54	36.33
400	2.72×10^{-12}	7.669	2,294	92.56	36.11
450	1.13×10^{-12}	7.640	2,438	93.59	35.92
500	4.89×10^{-13}	7.613	2,575	94.62	35.75
550	2.21×10^{-13}	7.585	2,705	95.65	35.61
600	1.04×10^{-13}	7.558	2,831	96.69	35.49
650	5.15×10^{-14}	7.531	2,952	97.73	35.38
700	2.72×10^{-14}	7.504	3,069	98.77	35.29
750	1.55×10^{-14}	7.478	3,183	99.82	35.20
800	9.63×10^{-15}	7.452	3,293	100.87	35.13
850	6.47×10^{-15}	7.426	3,401	101.93	35.07
900	4.66×10^{-15}	7.400	3,506	102.99	35.02
950	3.54×10^{-15}	7.375	3,608	104.05	34.97
1,000	2.79×10^{-15}	7.350	3,709	105.12	34.94
1,250	1.11×10^{-15}	7.229	4,184	110.51	34.83
1,500	5.21×10^{-16}	7.113	4,624	115.98	34.83
2,000	—	6.898	5,433	127.20	35.03
2,500	—	6.701	6,176	138.75	35.40
3,000	—	6.519	6,875	150.64	35.86
3,500	—	6.352	7,543	162.84	36.38
4,000	—	6.197	8,187	175.36	36.94
4,500	—	6.053	8,812	188.19	37.53
5,000	—	5.919	9,422	201.31	38.13
6,000	—	5.675	10,608	228.42	39.36
7,000	—	5.458	11,760	256.66	40.60
8,000	—	5.265	12,886	285.97	41.84
9,000	—	5.091	13,993	316.31	43.06
10,000	—	4.933	15,085	347.66	44.27
15,000	—	4.318	20,405	518.46	50.00
20,000	—	3.887	25,595	710.60	55.24
20,184	—	3.874	25,785	718.05	55.42
25,000	—	3.564	30,723	921.94	60.07
30,000	—	3.310	35,815	1,150.85	64.56
35,000	—	3.104	40,884	1,396.10	68.77
35,786	—	3.075	41,679	1,436.07	69.41

Table adapted from the book Space Mission Analysis and Design, by Larson and Wertz, Microcosm Press.