AE1222-II: Aerospace Design & Systems Engineering Elements I

Part: Spacecraft (bus/platform) 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/determine 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 course web site/Mobius 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.
2015/2016	• Updated number of spacecraft to be built in introduction; • Removed further references to Spacecraft Systems Engineering as the course text book; • Various textual additions; • Errata have been worked in; • Extended the number of example calculations; • Added 28 Figures and 16 Tables;
2017/2018	• Worked in errata list; • Added summary list of important constants; • Added more problems for solving; • Added answers to selected problems; • Added an example delta-v budget; • Reworked chapter on structures and mechanisms and EPS; • Moved some text on mission delta-v from chapter on propulsion to chapter on requirements generation; • Table with solar system parameters added; • References to Blackboard have been changed to course web site; • Added several more data tables.
2019/2020	• Worked in errata list; • Moved section on setting margins slightly more upfront and added data on typical design margins for different parameters at different phases in the design; • Moved some problems from tutorials to the respective problem sections; • Added/updated some parametric estimation relationships in appendix C; • Added/updated some data provided for estimation.
2020/2021	• Worked in errata list • Added some text on kick stages and their mass estimation

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Abbreviations

ABET	Accreditation Board for Engineering and Technology
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
COMMs	Communications
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

MGA	Medium Gain Antenna
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
PCB	Printed Circuit Board
PCDU	Power Conversion and Distribution Unit
PCU	Power Conversion Unit
PDR	Preliminary Design Review
PDU	Power Distribution Unit
PLM	Payload Mass
PMS	Propellant Management System
QSL	Quasi Steady Load
RCS	Reaction control System
RF	Radio Frequency
rpm	rounds per minute
RSE	Relative Standard Error
RTG	Radio-Isotope Generator
RX	Receiver
SAR	Synthetic Aperture Radar
S/C	Spacecraft
SRB	Solid Rocket Booster
(S)SD	(Sample) Standard Deviation
SSR	Solid State (data) Recorder
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
VDM	Vehicle Dry Mass
VEB	Vehicle Equipment Bay
VEM	Vehicle Empty 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	Altitude
H	Angular momentum
I	Area or mass moment of inertia, impulse, current
I_{sp}	Specific impulse
J	Safety factor
J_s	Solar intensity
k	Spring constant, Boltzmann constant
L	Length, loss factor
m	Mass flow rate
M	Mass, spacecraft residual dipole
n	number
N	number of loops in coil, noise, number
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, swath width
W	Weight, work

Greek

α	Absorptivity, 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, specific mass
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

List of Important Constants

Astronomical Unit (AU)	149,597,871 km
Stefan Boltzmann constant (σ)	5.67E-8 W/(m^2K^4)
Earth gravitational acceleration (g_o)	9.80665 m/s^2
Earth gravitational constant (μ)	398,600.5 km^3/s^2
Boltzmann constant (k)	1.38E-23 J/K
Equatorial Earth Radius (R_E)	6378.14 km

1 Introduction

“Space is hot”. According to [Kyle], each year close to about 100 space launches are conducted worldwide. In 2013 a total of 81 space launches have been conducted of which 48 to low Earth orbit (LEO), 32 to a higher Earth orbit and 1 into deep space. In 2014 a total of 92 space launches have been conducted of which 50 in to LEO, 41 in higher Earth orbit and 1 deep space probe. With each space launch one or multiple spacecraft are launched. For instance, in 2020 a single Falcon 9 launch orbited 60 Starlink spacecraft.

According to a recent forecast by [Euroconsult], it is expected that a total of 1155 satellites will be built & launched over the next decade (2014-2023). This means that on average 115 satellites will be launched each year over the next decade. Euroconsult also expects this will bring $248 billion in revenues from the manufacturing and launch of these 1155 satellites.

- Governments all over the world will be responsible for more than 75% of the $248 billion in revenues expected from the manufacturing and launch.
- In the commercial space sector, responsible for the remaining 25% of the total revenues, Euroconsult anticipates a total of 350 satellites to be launched over the decade, most of which will be for the replacement of capacity existing in-orbit. These satellites will be equally divided between the geostationary orbit (GEO) and lower altitude orbits (MEO and LEO); 83% of market value remains concentrated in the geostationary orbit, the destination of 300+ satellites operated by 30 commercial companies for communications and broadcasting services.

Space is hot because of the various applications it is fit for. These applications range from civilian to military ones. Military applications are amongst others early warning, encrypted communication, reconnaissance (intelligence gathering), navigation, meteorology, etc. Civil applications range from both commercial, like telecommunications, Earth observation, asset tracking and navigation to governmental like space exploration, science and meteorology. Roughly 2 out of every 3 spacecraft launched today is for civil purposes, whereas the remaining one third is for military spacecraft.

Before we continue, it is good to define what we actually consider to be Space. In popular language, Space is the void between celestial bodies. More formally, on Earth, one used the Kármán line, at an altitude of 100 km above sea level as the start of outer Space in space treaties and for aerospace records keeping. Space is an extremely harsh environment for human exploration, because of the hazards of radiation, the extreme cold/hot conditions and vacuum (causing outgassing of materials and evaporation of lubrication materials), but also because of the large distances between the various celestial bodies.

What exactly is a spacecraft (S/C)? A S/C is a vehicle or device designed for spaceflight, i.e. flight through (outer) Space. Spacecraft travel to many different places and have various uses. In [Jane’s], the following types of spacecraft are distinguished based on either their function(s) or their destination:

- (Earth) satellite – man-made equipment that orbits around the Earth or the Moon
- Orbiter spacecraft – A spacecraft designed to travel to enter orbit about a planet or some other object. It may 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.
- 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
- 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 generally consists 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.

These spacecraft are usually characterized by the payload carried, their destination in Space, electric power level used, how they are controlled, the type of propulsion used as well as their mass, size, life, reliability and cost. Table 1 provides an overview of specific spacecraft including their main characteristics. Below some important observations are given for the various characteristics:

- 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).
- Performance: Some spacecraft 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.
- 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).
- Maintainability: The Space Shuttle, the international space station and the Hubble Space Telescope are some examples of spacecraft that have been designed with some degree of maintainability. The majority of spacecraft though essentially are not maintained at all.

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

Table 1: Spacecraft and some characteristic data

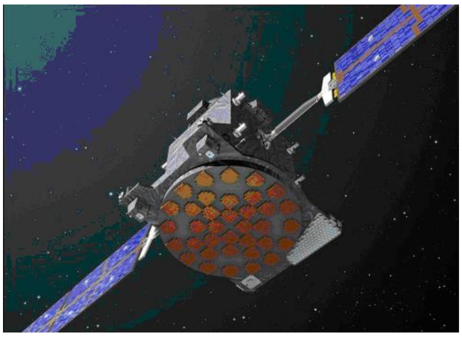
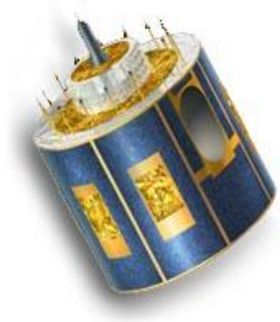
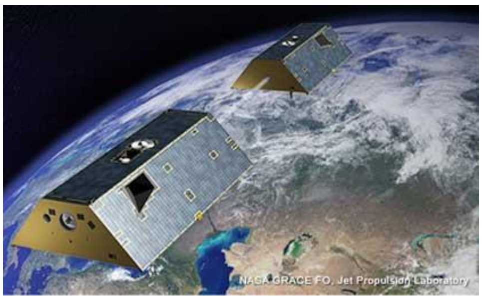
	STAR One C1 communications satellite (Brasil) • Payload: 28 C-band & 16 Ku-band transponders + 1 X-band transponder • Orbit: GEO • Mass: 4100 kg • Dry mass: 1750 kg • Dimensions: 4.0 m x 3.2 m x 2.4 m, 22.40 m span • Electric Power: 10.5 kW • Attitude: 3-axis controlled • Life: 15 yr (min.)
	GIOVE A navigations satellite (ESA) • 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° pitch/roll, 2.1° yaw • Orbit determination: NORAD, laser ranging, GPS • Propulsion: Δv = 90 m/s • Communications: S-band • Mission life: 27 months • Cost: 33.9 million US$ (FY 2006)
	Meteosat weather satellite (1st 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 (NASA/DLR) • 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: ±0.1° on critical components • Life: 5 yr • S.C cost: 127 million US$ (FY 2002)

Table 1*: Continued*

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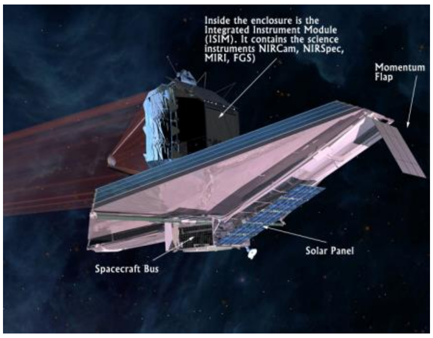
<tr>
<td>
</td>
<td>James Web Space Telescope (JWST, NASA)

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
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<td>ENVISAT (ESA)

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)
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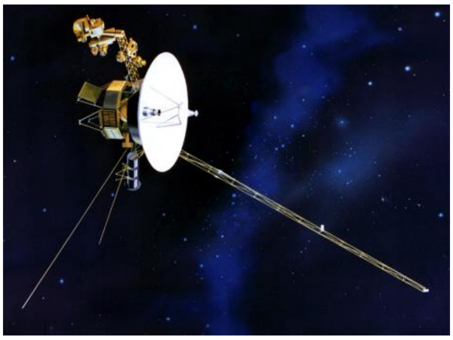
<td></td>
<td>Voyager Deep Space Probe (NASA)

Launch mass 825.5 kg
Dry mass: 722 kg
Payload mass: 125 kg
Spin stabilized at 60 rpm
Electric power: 421 W @ Begin of Life (BOL)
Communications: @ 8 GHz wavelength with antenna of 3.7 m diameter
Total mission cost (2 Voyagers, launch + operations: 865 US M$ (FY1977)
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<td>Mars Express (ESA)

Launch mass: 1,223 kg (including 120 kg adapter)
Payload mass: 173 kg
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°
Communication: 1.6 m high gain antenna
Electric power: 650 W at max. distance from the Sun
Operating temperature: 10-20 °C
Data storage: 1.5 GByte
S/C Cost: 195 US M$ (FY1996)
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</table>

Table 1*: Continued*

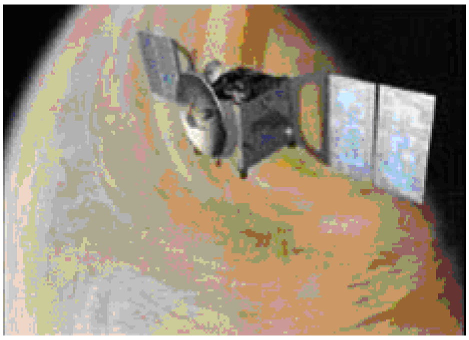
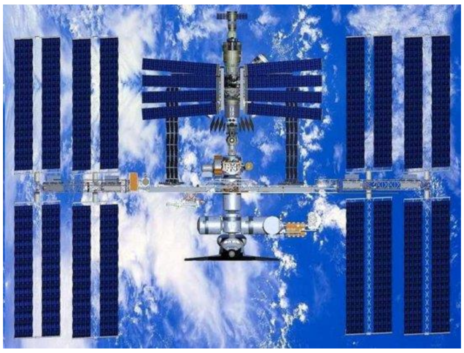
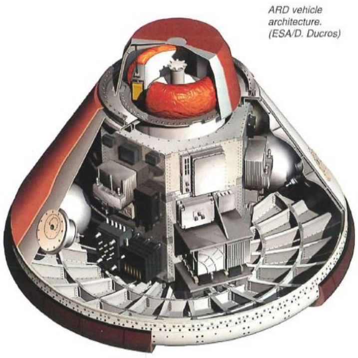
	Venus Express (ESA) • Launch mass: 1,244 kg (104 kg payload) • 1.65 m x 1.7 m x 1.4 m • Aluminum structure • Power: 650 W at max. distance from the Sun + batteries for eclipse periods • Propulsion: 414 N main engine (530 kg propellant) • Attitude thrusters: 2 x 4 10 N thrusters • Communication: S-band (5 W) and X-band (65 W) • Data storage: 1.5 GByte • Cost: 262 million US$ (2005)
	SMART-1 (mission to the Moon, ESA): • 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 • Cost: 100 M€ (FY2001)
	International Space Station (ISS, NASA) • 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, NASA) • 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, ESA) • Mass: 20,750 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, ESA) • 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 (NASA) • 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)

Most images are credited to NASA and ESA, Meteosat image credit Meteosat, STAR One C1 image credit STAR One, Delfi C^3 image credit TU-Delft

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.
- Cost generally is made up of recurrent cost (production cost, operations cost, insurance cost) and non-recurring (development) cost.
- 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.

Why do spacecraft differ or what makes them different?

To explain why spacecraft do differ, we first should realize that a spacecraft consists of two main elements:

1. **Payloads.** Payloads are the useful load of a spacecraft and as such they are the justification of the space mission. NO PAYLOAD means NO MISSION. Different payloads exist because of the different functions/tasks they have to fulfil, like communications, broadcasting (TV, radio), direct and/or remote[1] sensing, science, navigation, space station supply, sample return or even other spacecraft. An overview of current applications of typical mission payloads is given in Figure 1 [Fortescue, chapter 1.1].

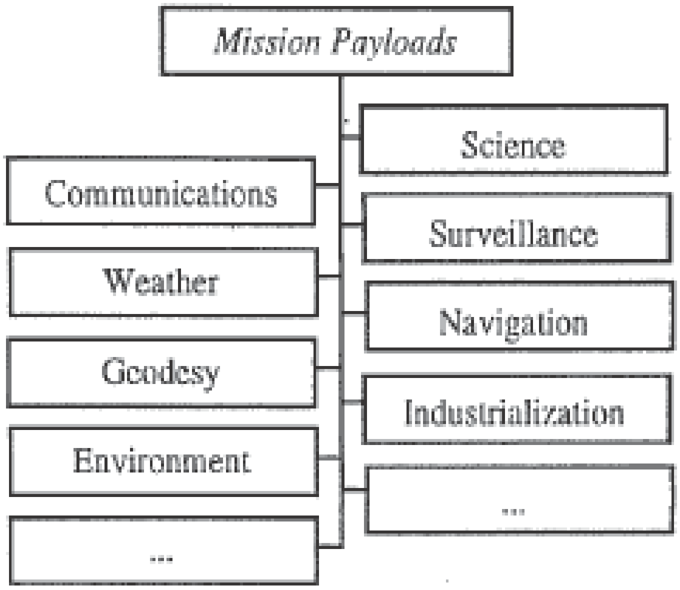


Figure 1: Mission payloads

2. **Spacecraft bus or service module**. The bus or service module 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. It essentially is the spacecraft without payload.

[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.

The type of payload to be carried on board strongly influences the design of the spacecraft (how it looks, its mass, etc.). For instance:

- To supply water and food to the international space station requires a spacecraft that has tanks to carry water and shelves or cases to store food. It can also be considered a cooling system is needed to keep the food fresh.
- A camera continuously taking images from Earth surface may have a need for highly accurate and stable pointing, which for other payloads may not be that important.
- A camera used to observe Earth from a platform in Earth orbit may require quite different means to control its temperature as for the same camera on a platform in orbit about Mars or Venus. 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.

Next to that the payloads have a strong influence on the design, they may also effect the system operational concept, orbit selection, spacecraft and ground segment architecture, and the types of launch services required. This again will have an effect on how the spacecraft looks like.

Because the type of payload influences how a spacecraft looks like, 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 hence may fall into more categories.

Other reasons for why spacecraft differ are because of:

- Different solutions to conduct a mission: Often there are many different ways to perform a mission. In practice we find that different design teams tend to come up with different solutions for the same problem. This may include differences in the launch vehicle and/or the mission orbit selected, as well as differences in vehicle design vehicle size and mass. It may also affect the level of autonomy of the spacecraft or how the vehicle communicates to ground (directly or via a relay station on ground or a satellite relay station), whether the S/C is propelled by its own integral propulsion system or one or more separate propulsive stages, etc.
- Different mission duration: Some missions require a spacecraft life of many years, like for telecommunications, Earth observation and navigation. Other missions only require a very short spacecraft life like for the ATV that is transporting cargo to the space station and after about 6 months separates from the space station and burns on re-entry in the atmosphere.
- Different mission 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.

- Different transfer orbits: For instance for interplanetary spacecraft we can select either a fast transfer or a more energy optimal transfer thereby using one or more fly-by's.
- Payload improvement: Payloads are improved continuously, 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.
- Vehicle technology improvement: The technology that enables us to perform spaceflight and to survive the harsh environment is continuously improving. Examples include the use of ion engines instead of chemical rocket engines for propulsion, or the use of novel Gallium-Arsenide solar cells instead of Silicon solar cells.
- Differences in available budget (money): We can use low- or high grade-equipment. Low grade equipment is less expensive than high grade equipment, but they have shorter life and/or fail more often and hence have a lower availability.
- Different risk strategies with risk being defined as the probability that design and development failures are incurred times the consequence of the failures. Some systems aim for high reliability (low probability of failure) whereas other systems aim for lower cost. Typically to attain a high reliability rather than a low reliability leads to high cost.
- Different political and ethical constraints. As examples are mentioned that in the US the export of space technologies is regulated under the International Traffic In Arms Regulations (ITAR) and that recently a number of space fairing nations have agreed that spent satellites will be transferred to a so-called graveyard orbit so as not to occupy scarce space in orbit and not to endanger collision with still operational spacecraft in case of a mishap.
- Etc.

Summarizing, we find that spacecraft differ because they have different payloads, mission duration, target, mission orbit, are operated differently, use a different launch vehicle to get into space, work on their own or in unison with others (e.g. in a constellation), etc.

The design process

Compared to a car most S/C are highly complex vehicles that bring high cost and long development times as well as a high design and development risk. To attain a successful design, a proper design process is needed.

According to [ABET]:
"Engineering design is the process of devising a system, component, or process to meet desired need. It is a decision-making process – often iterative – in which the basic science and mathematics and engineering sciences are applied to convert engineering resources optimally to meet a stated objective."
ABET uses the term **Engineering Design** to distinguish our concept of design from the common idea of design as unconstrained artistic and emotional expression.

As objective in the design, we can for instance select highest performance, lowest cost, and/or highest performance per unit of cost.

The design process for a complex vehicle as a spacecraft normally requires several design cycles even for preliminary designs. The spacecraft design process as depicted in *Figure 2* starts by first defining the spacecraft objectives and the spacecraft requirements. This usually happens in a so-called **space mission design study**, wherein we start with the all-important mission objective(s). Following the establishment of the mission objective(s), top functional requirements are established, taking into consideration the available capabilities and the imposed constraints. A solution—the mission design—is synthesized by an iterative process of analysis and design. Subordinate requirements on the launch vehicle, spacecraft, and its subsystems, along with mission operations, are then "flowed down" from the top-level requirements [Hofman et al, 1999].

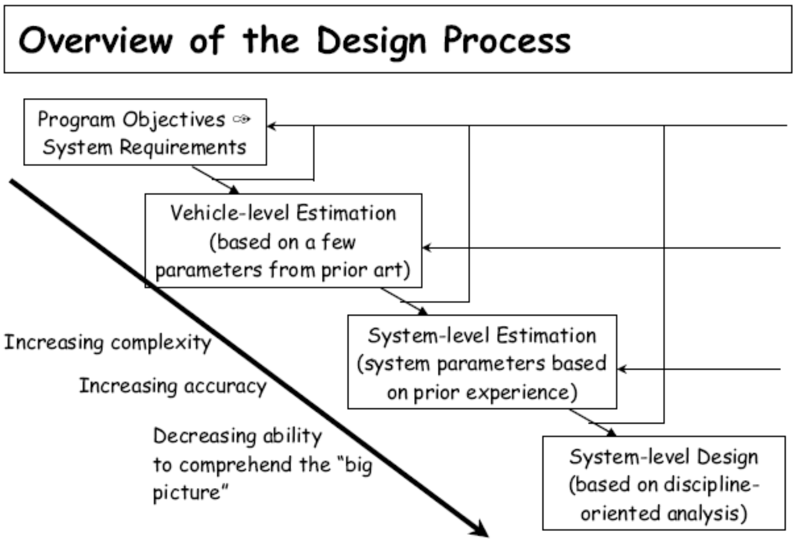


Figure 2: The Design Process [Maryland]

Objective: A short statement that tells what the design is to accomplish. An objective should refrain from indicating a solution to accomplish the objective as this may unnecessarily limit the search for solutions.
Requirement: A statement that provides a measure for how well a certain obligation/task needs to be performed.

After defining the objectives and requirements the actual design takes place in several rounds or design phases with clear milestones to achieve set by the designers in agreement with the customer. In this course, we focus on vehicle-level estimation and system-level estimation based on prior experience combined with simple (analytical) relations stemming from physics and other engineering disciplines. For more detailed system-level design based on discipline oriented analysis (with focus on performance) 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 focus on spacecraft design to a level of detail sufficient for level 2 designs and the design of the major spacecraft elements (level 3 design).

With each design phase (or 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 phase 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 a decreasing ability to comprehend the big picture. In S/C design, we typically distinguish between three/four design phases [Brown]:

- Preliminary analysis or feasibility studies (not actually considered a design phase, but more of an exploratory phase), wherein we aim to determine a reasonable spacecraft configuration that is able to do the mission, and to identify critical areas, important trades to be made and what it is going to cost. Estimation accuracy of vehicle level parameters on average is in range 30-50%.
- Conceptual[2] design, wherein we aim to select the 'best' concept, i.e. the best possible way of applying technical means to get a job done. Typical outputs are mainly study reports, models and mock ups next to analysis and simulation results. Estimation accuracy of vehicle level parameters on average is in range 20-35%.
- Preliminary[3] design (and technology completion), wherein the concepts selected are worked out into more detail using engineering models, mock-ups, prototypes as well as more detailed analysis and simulations. Estimation inaccuracy of the vehicle level parameters decreases to 15-25%.
- Detailed (or final) design and fabrication, which ends with end product detailed designs and end product component fabrication. Typical estimation inaccuracy is 5-15%.

Actual percentage values as mentioned above tend to depend on the parameter considered as some parameters are more easily estimated than others. The values may also differ depending on whether we are dealing with a completely new design or one that incorporates certain amounts of existing hardware. Values indicated are for a completely new design and are taken from the work of [Brown].

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 requiring more detailed analysis. 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) 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.
2. Establish requirements; According to the dictionary, a requirement is something that is imposed as an obligation; a necessity. They define what to design for (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.

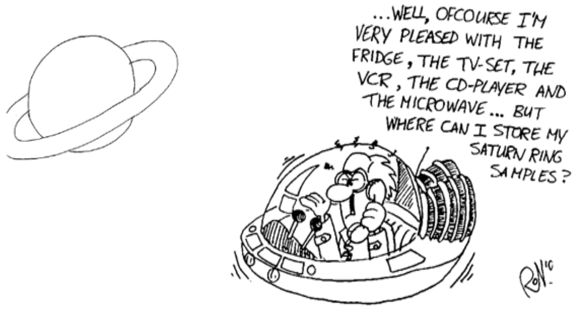


Figure 3: Illustration of bad problem definition / requirements generation

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.

[2] 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 may result. 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

[3] 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.

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/I. In common literature, the above is also referred to as the engineering design process or simply the design cycle.

In the lectures S/C design we focus on the sizing of the spacecraft (excluding the payload) 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)
- Add design margins (Chapter 3.7) and perform budgeting (Chapter 3.10)
- Evaluate the spacecraft sizing and budgeting results (Chapter 3.11)
- Perform spacecraft (sub)system sizing (Chapter 4)

Course material

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

Problems

1. List the various elements/segments of a space mission that a spacecraft interacts with and discuss the role/functions of these elements/segments (see AE1110-II).

2. Describe in your own words what is meant with a Space Mission Design study in relation to space vehicle / spacecraft design.

3. Describe in your own words the definition of engineering design according to ABET.

4. List the main phases in spacecraft design and explain how design accuracy and design effort vary for the different phases.

5. List the basic steps in engineering and shortly describe the purpose of each step.
   Hint: Consider the meaning of the acronym DESACME/I.

6. Describe in your own words the difference between a space mission objective and a spacecraft objective.

7. Describe in your own words what characterizes a proper objective.

8. Generate your own mission objective for a space mission that you consider worthwhile conducting in future.

2 Generating a spacecraft requirements list

Once we know what the spacecraft has to do (the functions that we need to design for), we can start defining how well these functions should be performed and generate a spacecraft requirements list (SRL). How to generate such a list is the topic of this section.

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	Payload dimensions: 1150 x 1410 x 1950 mm^3	
	2	Payload mass: 296 kg	
	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?)	

Note that in the table the space system element is identified where the requirement originated from, see under category.

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[4]. In addition we discuss how constraints are placed on the project as we have limitations concerning available resources (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.

2.1 Understand the role of the spacecraft in the mission concept

Before starting the requirements generation process, it is important to understand the role of the spacecraft in the mission concept and the purpose of the mission. A spacecraft typically forms part of the space element of a space mission concept next to for instance the launch vehicle, the mission orbit, the ground element and the mission operations. The space element of a space mission concept may consist of various different or multiple identical spacecraft. For instance, for some planetary missions, we may need both a kick stage and an orbiter. For others, we may even need a lander and/or an ascender (for instance for a sample return mission). For each of these spacecraft we need to understand its role in the mission concept. Once we understand the role of each spacecraft in the space element both in respect to each other and the other elements in the mission concept, the requirements for a specific spacecraft can be generated. As said earlier, this is usually done in a Space Mission Design study.

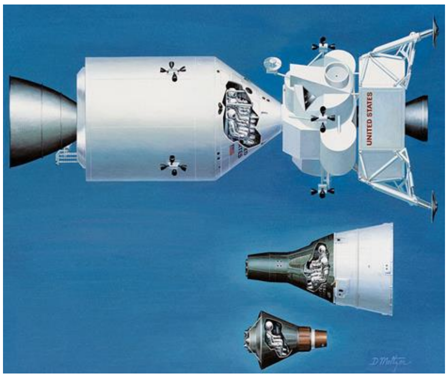


Figure 4: Part of space element for Apollo Moon mission (image credit NASA)

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

2.2 Spacecraft requirements from payloads

The mission (goal) of a spacecraft is to support the payload at the least expense or in the most optimum way. In more detail, it means that the S/C provides for amongst others the following functions:

- Transportation; It allows for transporting the payload to the required designation.
- Pointing of the payload; this is for instance to direct cameras or antennas.
- Protection against harsh (space/launch) environment; Protection against for instance space debris, and radiation.
- Attaching the payloads.
- Power; without electrical power most instruments will not function.
- Etc.

As indicated earlier, a spacecraft essentially is a platform carrying/supporting one or more payloads. Figure 5 shows 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).

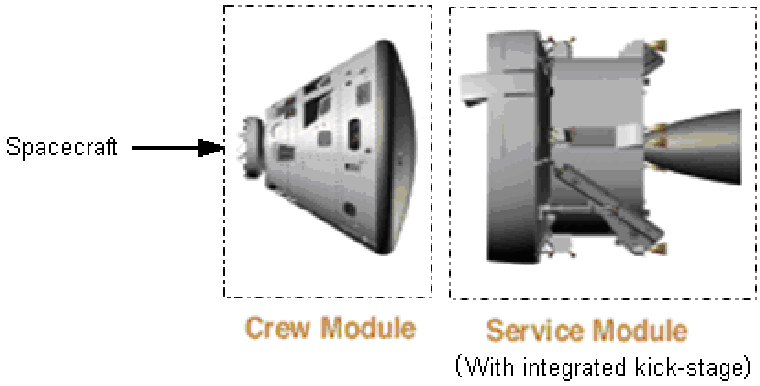


Figure 5: Service (bus) module carrying a crew module (image credit NASA)

Another example is ESA's Mars Express orbiter S/C, which forms part of the space element of Europe's first mission to Mars. The Mars Express orbiter hosts seven scientific instruments. HRSC (High Resolution Stereo Camera, OMEGA (Visible and Infrared Mineralogical Mapping Spectrometer) and MARSIS (Sub-surface Sounding Radar Altimeter) are used to image and study the surface and subsurface. For atmospheric and plasma studies there is PFS (Planetary Fourier Spectrometer), SPICAM (Ultraviolet and Infrared Atmospheric Spectrometer) and ASPERA (Energetic Neutral Atoms Analyser). OMEGA is also used to the study the atmosphere, and plasma is also studied with MARSIS. A radio link to convey data between the spacecraft and Earth is provided by MaRS, the Mars Radio Science Experiment, and used to study the gravity of Mars, the atmosphere and ionosphere, surface roughness and solar corona.

Each payload brings its own requirements to the spacecraft bus or service module. In the requirements generation process, S/C engineers may have to cooperate with payload instrument design engineers to uncover the S/C requirements and to formulate them in a way that is agreeable to them both

An important step in the determination of the requirements that stem from the payload(s) is to list the characteristics of the payload(s) considered, draw them out and gather and list its/their needs for support.

Table 4 provides an overview of some important payload parameters to be considered when generating S/C requirements.

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

Mechanical
Mass
Size (outline and mounting dimensions)
Moments of inertia
Uncompensated momentum (due to moving elements in the payload)
Launch loads (shock and vibration)
Disturbances
Thermal
Operating temperature range
Conducted and radiated heat flux to/from payload
Thermal gradients and base plate distortion
Electrical
Power usage (peak/average, AC/DC)
Output data rate
Command, control, and telemetry
Electromagnetic interference
Optical
Sensor orientation and clear fields of view
Pointing stability, agility
Contamination: particulates, outgassing
Other
Reliability
Life
Cost

As an example of a listing of payload characteristics, 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 relating to the parameters mentioned in the above table. 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. Output data rate/bandwidth
7. Operating temperature range
8. Reliability
9. Life
10. Etc.

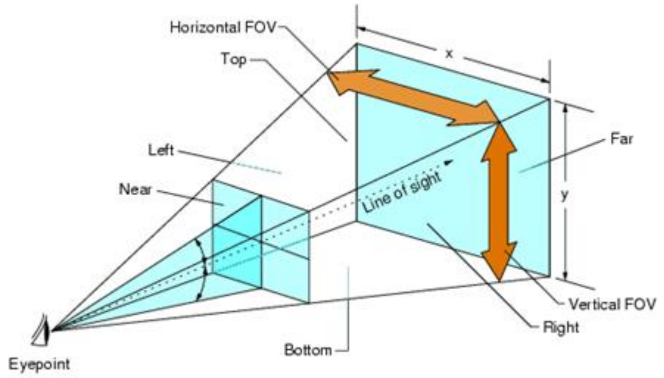
Below, some of the above parameters are commented upon.

Ad 1/2) All instruments are characterized by a certain mass and size. Since the spacecraft should allow for carrying the mass and provide for sufficient space to carry the instrument, it is important to know the mass and size of the instruments to be carried as payload. Typical values for some instruments/payloads have been collected in e.g. Table 5 and Table 6 as to provide the S/C designer with a starting point for S/C design. For instance, in case we are considering a communications payload/repeater[5] carrying 28 transponders each of which produces a power output of 100 W(only about 20 are actively used and 8 are used as backup; typically 1 out of 3 is used as backup) with a power output of 100 W, it can be found a repeater mass of 165 kg (5.9 kg/transponder) and a total antenna mass of 60 kg.

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. Payload power levels can be obtained from Table 5 and Table 6. For communication payloads, payload power level can be determined based on number of operating transponders times the power output divided by the transponder efficiency. The latter is in range 50-60%.

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 arc sec (1 arc sec 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 arc sec 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 6: Definition of Field of View, Line of Sight and Aspect ratio

Ad 6) Data rate: Measure of amount of data generated representative for the information gathered. It can be viewed as the speed of travel of a given amount of data from one place to another. Data transfer is usually measured in bits (or bytes; 1 byte is 8 bits) per second or a multiple thereof. For example, a typical low-speed connection to the Internet may be 33.6 kilobits per second (kbps). Some background information on bit rate can be obtained from https://en.wikipedia.org/wiki/Bit_rate.

[5] Repeater of a communications payload essentially consists of various receivers and transmitters, which together are referred to as transponders. A communications satellite can have up to 200 transponders on board.

Typical data rates related to voice/music/video communications can be found in Table 58. Combining the data rates per voice[6]/music/video (TV) channels with the required number of such channels gives the overall data rate for a communications payload. Using relations from the section on telecommunications then allows determining the required bandwidth in case of digital transmission. In case of analogue transmission, these data are already included in the indicated table be it that they still have to be multiplied by the number of channels.

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, see Table 58.

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.

Noteworthy of mentioning is the support issue of (preventing) electromagnetic interference and contamination. This issue can affect the design greatly, but is not dealt with in this course. For this you are referred to the general lectures on physics.

[6] Voice channel is also referred to as a phone line.

Table 5: Overview of specific payloads and their characteristics

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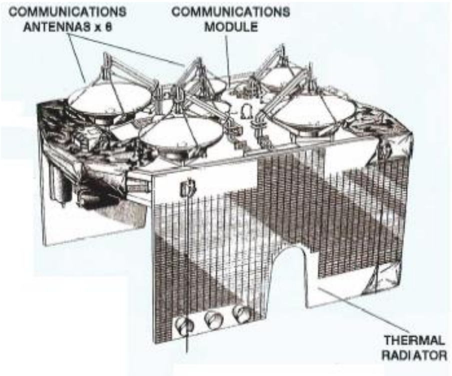
Communications Payload



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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) + associated antennas (transponders may share a single antenna)
- 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
- Antenna mass: See section on communications later in this work
- Transponder mass (depending on transmitter power)[7]:
  - 1.2 kg/transponder at a transmitter power of 5W
  - 2.67 kg/transponder at 13 W
  - 5.9 kg/transponder at 103 W
  - 9 kg/transponder at 150 W
  - 14 kg/ transponder at 180 W
- 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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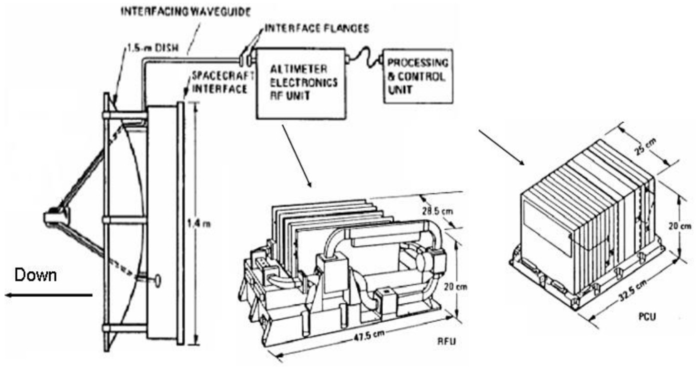
Altimeter



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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 the three modules in the figure
- 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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[7] Some variation is possible depending on the specific design. For instance some 5 W transponders are known that have a transponder mass of 4 kg.

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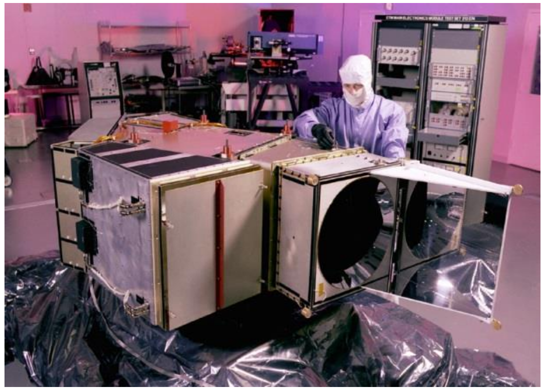
<td>Remote Sensing instrument (camera)
</td>
<td>Instrument: Landsat-7 Enhanced Thematic Mapper (ETM)
Instrument parameters

Mass: 425 kg
Power: 590 W (imaging), 175 W (standby)
Duty cycle: 15% imaging
Thermal control: 90 K (focal plane)
Pointing requirements:

Control: 60 arcsec (1 sigma)
Knowledge: 45 arcsec (1 sigma)

Jitter (jitter can be thought of as shaky motion) : 4 arcsec (1 sigma)
Physical Size:

Scanner Assembly: 196 x 114 x 66 cm
Auxiliary Electronics: 90 x 66 x 35 cm

Data taken from:
http://eospso.gsfc.nasa.gov/eos_homepage/Instruments/ETM/</td>
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<tr>
<td>Observation camera
</td>
<td>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)</td>
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<tr>
<td>Navigation payload

GIOVE B payload module on sled ready for thermal testing</td>
<td>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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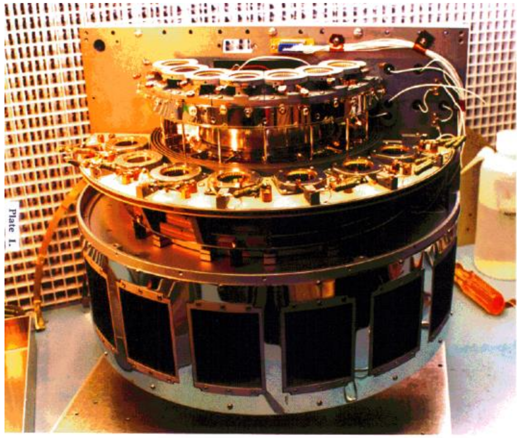
<td>Science payload
</td>
<td>Thermal Ion Dynamics Experiment (TIDE)/Plasma Source Instrument

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 (based on [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	2.5
	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		< 0.3[8]
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

[8] Based on ERS data

As indicated earlier, some S/C carry multiple payloads/instruments with each having its 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 [ESA, 2011]

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

To further illustrate the foregoing, the next few tables provide for a tabular overview of the Mars Express orbiter payloads (7 in total) and their main characteristics. Data has been taken from [ESA, 1997].

Table 8: Overview of Mars Express Orbiter Payload [ESA 1997]

MARS EXPRESS ORBITER MODEL PAYLOAD

Objectives	**Instrument**	**Acronym**	**Ref. Existing Instrument**	**Mass** (kg)	**Power** (W)
Surface Imaging	High-Resolution Stereo Colour Imager	HSCI	HRSC	21.4	40
Surface Composition	IR Mapping Spectrometer	IRMS	OMEGA	32.6	42
Atmosphere	Atmospheric Spectrometer	ATSP	PFS	42.6	45
Subsurface Sounding	Subsurface-Sounding Radar/Altimeter	SSRA		15.0	60
Environment	Energetic Neutral Atoms Analyser	ENAA		5.0	6
TOTAL MASS				**116.6**	
Design Goal				*120.0*	

HIGH-RESOLUTION STEREO CAMERA

HRSC	**Mass** (kg)	**Dimensions** (cm)	**Location**	**Op. Power** (W)	**Data Rate** (Gbit)	**Duty Cycle**
Camera Head	9.83	51×26.2×20.7	Body-fixed on Orbiter	40.4	1 per day	one imaging sequence every orbit
Digital Unit	9.7	36×26.2×15.4				
Thermal and cabling	1.85					
Total Mass	**21.4**					

ORBITER PAYLOAD DATA BUDGET

Instrument	Mode	Data volume (Gbit)	Data rate (Mbit/day)
HSCI	Orbit	>0.2	~1000
	Mission	~800	
IRMS	Orbit	<0.3	~100
	Mission	~500	
ATSP	Orbit	<0.16	?
	Mission	?	
SSRA	Orbit	?	~285
	Mission	?	
ENAA	Orbit	?	<864
	Mission	?	

ORBITER PAYLOAD OPERATIONAL REQUIREMENTS

Instrument	Operational Profile (day/night)	Operational Profile (summer/winter)	Sampling Rate (orbit)	Capable Lifetime (years)	(°C)
HSCI	4 to 30 min daytime	all seasons	every orbit	>3	Operations: −70/+50 Stand-by: −70/+50
IRMS	5 to 10 min daytime	all OK	every orbit	>2000 hours	Operations: −120/+40 Stand-by: ±50
ATSP	2 h/orbit all OK	both at all local times	one spectrum /10 sec	nominal	Operations: −20/+40 Stand-by: −20/+40
SSRA	all OK	all OK	one sample /sec	several years	Operations: −20/+40 Stand-by: −20/+40
ENAA	all OK	all OK	depends	several years	Operations: ±40 Stand-by: −40/+70

The foregoing table clearly shows the various instruments that make up the payload of the Mars Express Orbiter and their characteristics with respect to mass, size, power consumption (power need), data rates as well as duty cycle, and some more operational requirements. Also clear is that the payload consist of a number of already existing instruments (not all payloads for this mission had to be developed from scratch.

Data on payloads of importance for generating a spacecraft requirements list can be obtained from:

- Spacecraft Systems Engineering [Fortescue]
- 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

The importance of having a clear overview on the payload(s) carried by a spacecraft is, that it is immediately clear what support the S/C should provide for.

2.3 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 9.

Table 9: 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. What should be kept in mind that in the requirements generation process, S/C engineers have to work together with engineers from many other disciplines to ensure identification of proper S/C requirements. Of course, this will also allow identifying proper requirements for all of the other elements that make up a space mission.

In the ensuing discussion we have to keep in mind the various elements that make up a space mission of which the spacecraft is only one of the elements. We will consider how each of the mission elements (except for the spacecraft itself) may constrain the S/C design.

Spacecraft shall fit in launch vehicle

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 Automated Transfer Vehicle with Payload Transport Vehicle[9] (PTV) fitted into Ariane 5 payload bay. Figure shows a somewhat peculiar shape of the dynamic envelope (dotted line encompassing ATV with PTV), not uncommon to most launch vehicles. Maximum available diameter is 4.57 m. It also shows that the whole vehicle is connected to the launch vehicle via a payload adapter (conical ring in figure). The cylindrical section of the payload bay of the launch vehicle is ~8 m high. Parabolic section of the payload bay is not really used. Information on dynamic envelope of a launch vehicle generally is contained in the launch vehicle manual, a many page document describing the launch vehicle, launch operations, launch site, launch performances, etc. A good secondary source may be the Launch Vehicle Catalogue (available on course Brightspace web site).

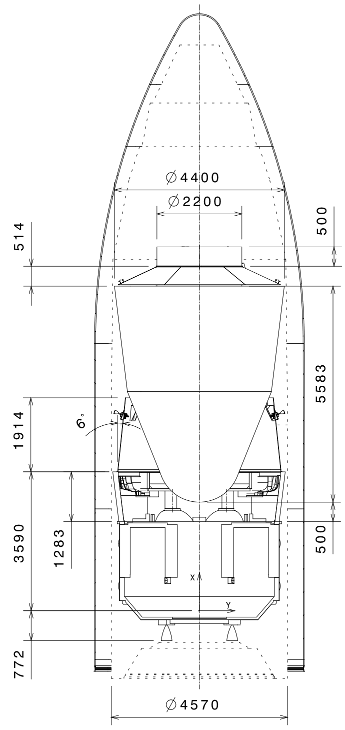


Figure 7: PTV fitted into Ariane 5 payload bay [Heidman 2004]

Spacecraft mass limited by maximum mass that can be carried by launcher in to the designated orbit

A launch vehicle can only launch a certain payload mass into some orbit. For the Taurus rocket, this is illustrated in Figure 8 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 an adapter[10] and maybe even the mass of a kick and/or upper stage. Sometimes even more than 1 vehicle is launched with the launch vehicle.

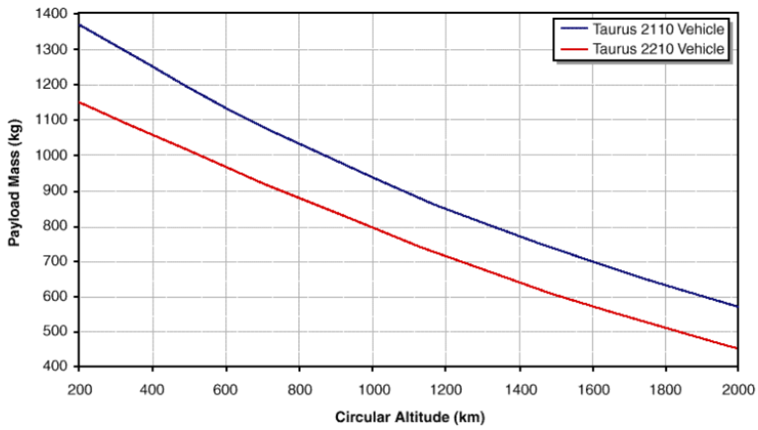


Figure 8: Taurus performance to 28.5° LEO orbits [LVC]

The spacecraft designer should make sure the spacecraft will not exceed the maximum mass that can be carried by the designated launch vehicle into the designated orbit. If the spacecraft turns out to be

[9] PTV is a redesign of ESA's ATV (Automated Transfer Vehicle), adapted for carrying the Payload Transfer Vehicle (PTV) which is capable of transferring payload from the Space Station to ground. The top segment (~1 m in height) is intended to allow for docking with the Space Station. This configuration was studied for ESA by EADS Space Transportation around 2004 {Heidman 2004].

[10] **Launch Vehicle Adapter.** The launch vehicle adapter, also referred to as a *payload attach fitting*, provides the structural connection between the space vehicle and any associated dispensers, or kick motors. It usually is discarded off right after the launch and hence its effect on the design of the spacecraft and its operations can be limited. Still, it should not be neglected.

more heavy, another launch vehicle (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 (Δv, velocity change) performance, the spacecraft may need to perform one or more maneuvers before reaching its final orbit. This is illustrated in Figure 9 for a mission to the Moon.

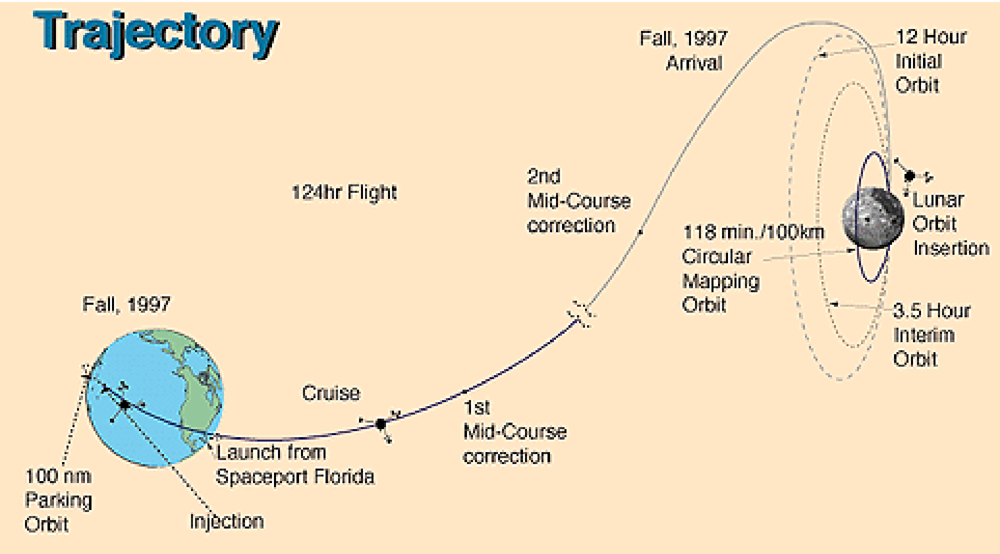


Figure 9: Maneuvers needed for spacecraft to travel from parking orbit to lunar orbit (image credit NASA)

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 from orbit analysis. For illustration, Figure 10 provides a detailed overview of typical velocity changes required to accomplish a Moon mission including landing and return to Earth divided over a number of maneuvers, like insertion from low Earth orbit (LEO) into lunar transfer orbit (LTO) 3.107 km/s, injection from LTO into low lunar orbit (LLO) 0.837 km/s and lunar landing 3.140 km/s. Figure also provides required Δv for the return flight.

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 10: Overview of typical velocity changes required to accomplish a lander mission to the Moon and back

Next figure shows a typical way of representing a delta-v budget. It again shows the manoeuvres and the associated Δv as in above figure, but also allows for providing information on the number of manoeuvres needed, as well as taking into account best/worst case situation. It also would allow for incorporating a Δv margin. The sum of all manoeuvre velocity changes (absolute value) is referred to as *mission characteristic velocity*. Notice that in many cases the velocity change accomplished by the launch vehicle is left out of consideration.

Mission orbit	Geostationary	(Allowable deviation from nominal position 0,1 deg)			
Launcher	Proton				
Launch in GTO					
Mission duration (yrs)	15				
Maneuvre	**delta v/maneuvre (m/s)**	**cycle time (days)**	**no. of maneuvers (-)**	**delta v/yr (m/s)**	**total delta V (m/s)**
Apogee kick	1836,49	*	1,0	*	1836,5
10 yr average NSSK	10,73	86,1	63,6	45,5	682,0
Worst Case NSSK	10,90	77,4	70,7	51,4	770,7
EWSK	0,13	35,3	155,3	1,33	19,9
Worst Case EWSK	NA	NA	NA	1,74	26,1
Orbit Maneuvres	0,00	*	0,0	*	0,0
Disposal	10,88	*	1,0	*	10,9
		Total Delta V (most favourable)			2549,3
		Total Delta V (worst case EWSK)			2555,5
		Total Delta V (worst case NSSK & EWSK)			2644,2

Figure 11: Typical Δv *budget for a GEO telecommunications satellite*

An important question that needs to be dealt with during the design is what *Δv* shall be provided for by the various elements that make up the space mission (launch vehicle, kick stage, orbiter, ascender, etc.) and what can be accomplished using fly-by and aero-braking or aero-capture? The more that can be done using either fly-by and/or aero-braking the lower the mass of the spacecraft(s) to be designed. The downside may be that the duration of the transfer flight increases. It is for this reason that S/C engineers generally work together with orbital analysts in the Space Mission Design phase as to allow for quick studies of different vehicles and orbits to find out what is the best option for the mission considered.

An important distinction to be made for all propelled phases when generating a *Δv* budget is that whether we are dealing with high or low thrust. This is because the thrust level affects the required *Δv*, mainly because it has an effect on the gravitational loss incurred. This is illustrated in Figure 12, which shows typical values of *Δv* for altitude and inclination change in Earth environment from an initial altitude of 400 km for both high thrust (acceleration levels of order 1 m/s^2 (figure on left) and low thrust (acceleration levels below 0.001 m/s^2) rocket propulsion systems [Sanchez]. From this figure it follows for a typical transfer flight from LEO (28.5°) to GEO that high thrust requires a *Δv* capability of 4220 m/s, whereas the value attained with low-thrust (thrust value not given) is 5900 m/s, which demonstrates a large increase.

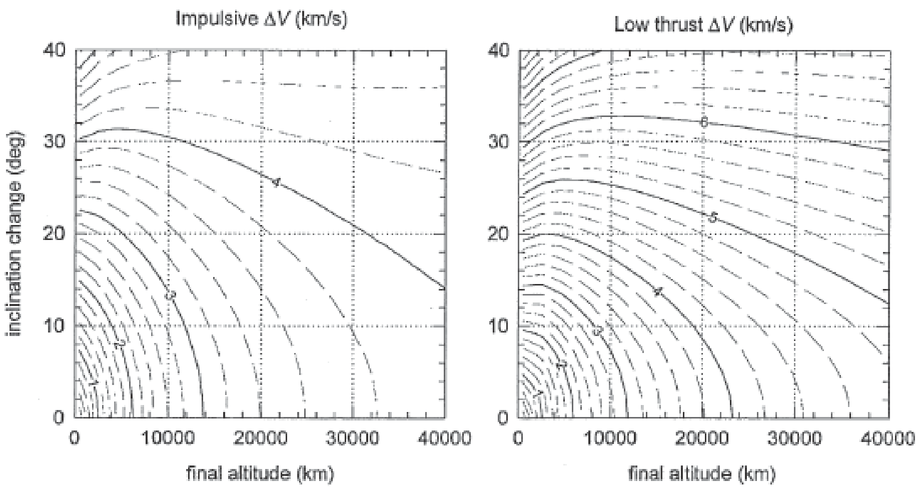


Figure 12: Contour maps of Δv for altitude and inclination change (Initial altitude is 400 km)

Next figure shows effect of the thrust level on the velocity increment for a Mars mission as determined by [Turner]. Again, we see that with decreasing thrust level the required Δv capability increases.

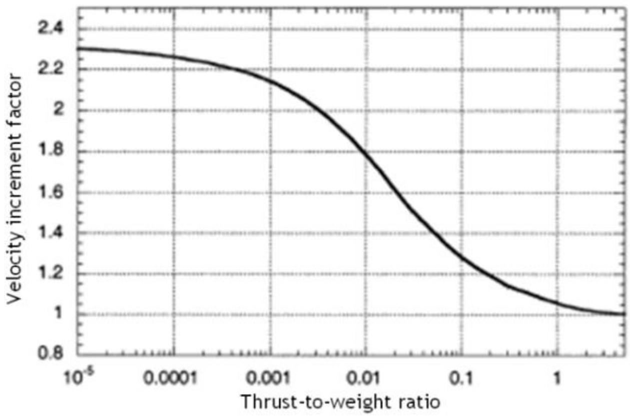


Figure 13: Effect of thrust to weight ratio on mission characteristic velocity (Mars mission)

In general, to generate a Δv budget, we use orbit/trajectory analysis. However, in case of a first guestimate, we resort to using historical data from past missions. Representative historical data for different types of maneuvers can be found for instance in Appendix A.

An important question is why to consider low thrust propulsion options as in this case the required Δv increases. The reason for considering low thrust options with the associated increase in gravity loss is because some low thrust options allow for significantly higher exhaust velocities (up to 100 km/s) then the high thrust options. This than allows for a much reduced propellant load and hence also increased payload mass. This will be discussed later in more detail.

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 14. 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. For other load factors, see http://en.wikipedia.org/wiki/Load_factor_(aeronautics). For a a discussion on human tolerance of G-force, consult http://en.wikipedia.org/wiki/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, 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.

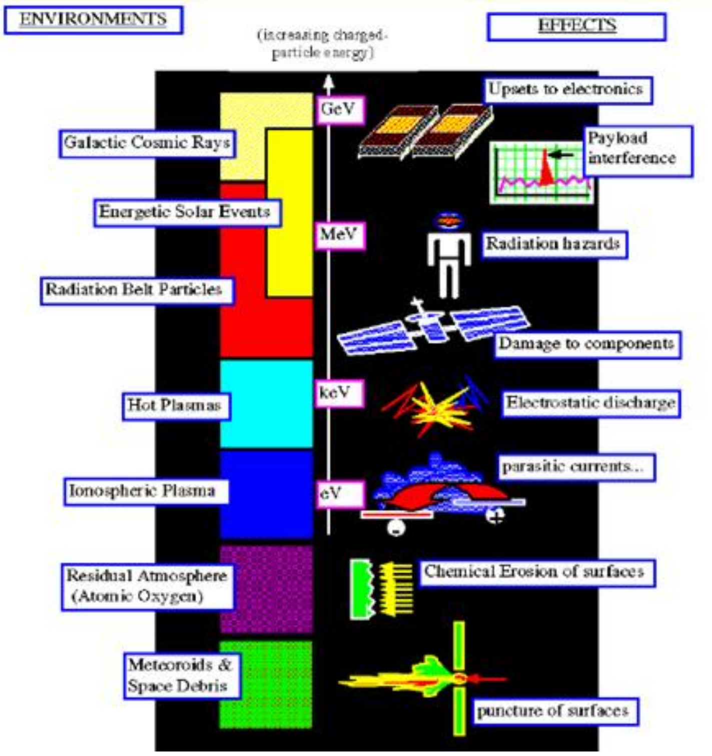


Figure 14: Space environment and effects on spacecraft (courtesy of ESA SME initiative training course)

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 10 shows characteristics of an ESA Deep Space Network ground station.

Table 10: 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 this data one learns that essentially two frequency bands[11] 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.

The table also gives a normal data rate of maximum 1 Mbps, but also a higher transmission rate of up to 105 Mbps can be accommodated for. Selecting this ground station, it more or less determines that the space mission should be designed with this maximum in mind and hence also the spacecraft should be designed with this in mind.

The very large antenna size of 15 m and 35 m of the ground station 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, one may say that the larger the antenna on ground, the smaller the antenna in space and vice versa.

[11] From Wikipedia: A **frequency band** is an interval in the frequency domain, delimited by a lower frequency and an upper frequency. The term may refer to a radio band or an interval of some other spectrum. Consider in this respect also our earlier discussion on available bandwidth.

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.

2.4 Requirements from financial 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.

2.5 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 shall:

- 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 manoeuvres 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>". For instance, the spacecraft shall be:

- Low cost
- Reliable
- Maintainable
- Safe

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.

2.6 Steps in spacecraft requirements generation

Steps in spacecraft requirements generation include:

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

Ad1) S/C is to provide for e.g.:

- pointing knowledge and control (attitude)
- position knowledge and control (navigation)
- electric power
- ruggedness (structure)
- intelligence (command & data handling)
- communications with ground and/or other S/C
- ability to manoeuver (change the orbit/maintain position) in space

Important constraints and operations requirements applicable to almost all spacecraft are listed in Table 11.

Table 11: Typical spacecraft constraints and operations requirements

Spacecraft shall • fit in launcher (both in terms of mass and size) • withstand launch loads • be reliable • be safe • ensure a certain life (duration)	• be of limited cost • be available when needed • be maintainable (Space Shuttle, Hubble Space Telescope) • be transportable (over ground, air, etc.)

Ad 4) To keep a clear overview of the requirements and not to forget any, they are usually collected in a so-called requirements list, see also 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 should provide for understanding why the requirement is as is and 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. A more complete overview of a requirements list is provided in Table 12.

Table 12: 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 $\Delta v \geq 700$ 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 (number of S/C): 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.

An example requirements document can be obtained from: Three Corner Sat, 2008, retrieved 27-01-2015, from http://spacegrant.colorado.edu/COSGC_Projects/Past_Projects/3cs/software/srs.html.
This example may be a little bit overwhelming as it shows many requirements. For now though it suffices to say that we do not expect you to be able to generate such a document at this stage of your study, but that you at least should understand the idea behind requirements generation, know about some definitions and that you have an initial capability of developing spacecraft requirements from scratch and that you know about the requirements (SMART) on requirements, see later entry, and a basic ability to do so.

Requirements flow down
Some spacecraft 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).

2.7 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 or Attainable – Are the requirements’ objectives achievable and attainable, i.e. is it possible physically for the system to exhibit that requirement under the given conditions. Some requirements may be beyond the bounds of human knowledge. Others may have theoretical solutions but be beyond what is currently achievable, like:
  - "The system shall be 100% reliable".
  - "The system shall be 100% available".
  - "The system shall have a minimum response to a query of 1 second irrespective of system load".

  To determine whether a requirement is attainable, it is good to ask oneself whether it has been done before (e.g. in past missions) and if not, why not?
- Realistic – Are the requirements realistic with respect to available resources?

  To determine whether a requirement is realisable or not is the most difficult part of creating a SMART requirement. It is definitely linked with a requirement being attainable or not and one may say that if it is not attainable, than it is definitely not realisable. Still, if it is attainable, it does not necessarily be realisable. To determine whether an attainable requirement is realisable, we should ask ourselves whether this requirement can be satisfied given:
  - other system and physical constraints that we have?
  - project resource (funding, schedule, availability, etc.) constraints which we must work to?

  For example, if there is a requirement to have 99% reliability but the project budget does not permit the inclusion of the extensive reliability measures needed to satisfy that requirement then that requirement is not realistic. To decide whether a requirement is realistic sometimes extensive analyses are needed.
- Time-bound or Traceable – Time-bound refers to when the requirements’ objectives are to be realized? Time-bound may also be replaced by traceable, meaning that requirements on a lower level (more detailed design) depend on requirements on a higher level. As an example, we mention S/C-cost, -reliability, and -life (lower level), which depend on requirements concerning mission cost, mission reliability and mission life (higher level). Traceability requires that a system is in place that helps to find out to which higher level requirement a lower level requirement relates. For instance, this could be a numbering system where the higher level requirements are given a number X and the lower level requirements a number X.Y, where X denotes the higher level requirement and Y is a counter that tells how many lower level requirements are derived from this one higher level requirement.

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

To help you in your understanding of the above, the next table lists words that are considered forbidden to be included in proper requirements. The words have been taken from a presentation by ESA on requirements generation.

Table 13: Requirements - Forbidden words [Erd]

- and/or
- etc.
- goal
- shall be included but not limited to
- relevant
- necessary
- appropriate
- as far as possible
- optimise
- minimal, maximise
- typical
- rapid
- user friendly
- easy
- sufficient
- enough
- suitable
- satisfactory
- adequate
- quick
- first rate
- best possible
- great, small, large
- state of the art

2.8 Problems

1. Describe in your own words:
   a) The importance of requirements
   b) The main types of requirements that can be distinguished and explain their role in S/C design
   c) What a S/C requirements list is and discuss the main elements that constitute such a list.
   d) Requirements rational
   e) Requirements flow down

2. List and explain the three main sources from which requirements originate.

3. List and explain at least 4 criteria for proper requirements.
   Hint: Consider the meaning of the acronym SMART

4. List at least 5 general functions a spacecraft fulfils.

   Give an account of the general problem of reliability encountered in spacecraft and discuss the difference between reliability and risk as well as the measures used in quantifying reliability and risk.

5. Consider a space probe that is destined for Jupiter. Which of following requirements are definitely not defined in a way that is specific, measurable, attainable (realistic) and or relevant to the mission?
   1. Probe shall be 100% reliable
   2. Maximum mass of probe shall be 3000 kg
   3. Mass of probe shall be 3000 kg
   4. Probe shall be Sun-pointing with an accuracy of +/- 0.1 deg
   5. Probe flight time shall be limited to under 1 hour
   6. Probe shall be 100% efficient
   7. Probe shall be capable of performing manoeuvres with a total Δv of 2.3 km/s

   Answer: The requirements 1, 3 and 6 are not defined in a SMART way. For instance, 100% reliability is not attainable. Also it is quite unrealistic to design for a probe mass of exactly 3000 kg. It is better to include some range of values or still better to indicate a maximum allowable level as is done in requirement 2.

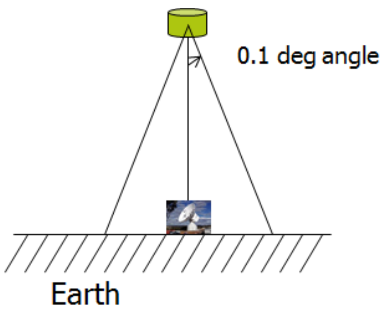
6. Which of the following requirements relating to a deep space probe do you consider to be a functional requirement for the probe?
    a. Probe mass shall be ≤ 300 kg
    b. Probe cost will be limited to not more than 100 million Euro in Fiscal Year (FY 2013 money)
    c. Probe shall be Sun-pointing during operational phase of the mission. Stability in all three directions shall be better than 0.1 deg/s
    d. Mission duration shall be in excess of 1 year
    e. Probe temperature shall be maintained within range -10 to + 20 ^{o}C
    f. Probe shall be safe to operate
    g. Probe shall be able to provide manoeuvres up to a total Δv capability of ≥ 2.3 km/s
    h. Probe reliability shall be better than 95%

    Answer: The requirements 3, 5 and 7 are functional requirements as these specify what the spacecraft shall do. All other requirements specify what the spacecraft "shall be".

7. You are assigned the task of designing a GEO spacecraft that is to be launched into GTO using Ariane 5G as the launch vehicle. Determine for this launch vehicle the limitations in payload mass, and volume as well as limitations with respect to the mounting of the S/C onto the launch vehicle. Also determine launch cost, and nominal payload separation velocity and rotational rate.

    Hint: Use ESA Launch Vehicle Catalogue or International Reference Guide to Space Launch Systems by Steven J. Isakowitz (both are available in LR Library). ESA Launch Vehicle Catalogue is also available on the course web pages.

8. An S/C in GEO has on board a small camera that is looking down (nadir) to Earth. This camera is pointed to nadir with an accuracy ±360 arc-sec[12], see figure. What is maximum distance between target and point where camera line of sight intersects with Earth (Earth curvature is negligible)?



    Answer: Maximum distance is 63 km.

9. An S/C is in orbit about Earth. It uses a camera to view Jupiter which is at 6 AU from Earth. The camera is pointed by the S/C. What pointing accuracy is needed to ensure that line of sight of the camera crosses Jupiter's disc at a point within 2000 km from the target (Jupiter centre)?

    Answer: Pointing accuracy is 0.45 arcsec.

More problems for exercising upon are available via the course web pages, whereas some are also contained in a separate workbook available from the TU-Delft online print shop. Of this workbook also an electronic copy is available via the universities online learning platform (Brightspace).

[12] 1 arc-sec is 1/3600th of a degree.

3 Spacecraft design: Vehicle level estimation

In the early stages of spacecraft design (like those phases that tend to be part of space mission design studies) we tend to investigate different vehicle 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 addition, in these early stages, alternative solutions (different concepts for the spacecraft on how to perform the spacecraft mission) may be generated which should be analysed and traded against each other so as to select the “best” design. To perform such trade(-off) studies evaluation criteria shall be established based on the requirements. Most common criteria are mass and cost, but also size and reliability do matter.

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 essentially consists of 5 steps:

1. Generate concepts
2. Estimate S/C characteristics (vehicle properties)
    a. Determine what S/C characteristics are to be estimated;
    b. Collect relevant data from comparable spacecraft;
    c. Generate estimation relationships or select relationships from literature;
    d. Estimate characteristics and document.
3. Determine S/C configuration and mass moments of inertia
4. Budget resources and set margins
5. Document, evaluate and if necessary iterate and/or improve

In the remainder of this chapter these steps are discussed in some more detail.

3.1 Generating concepts

First step in the design process is to generate vehicle concepts. For this, we generally study comparable past (from history) designs. Comparable is here defined as spacecraft that are more or less performing similar tasks/functions as envisioned for the vehicle that we are trying to design. Based on the vehicles studies, one or more concepts are generated that potentially can do the job. For illustration, Figure 15 (courtesy NASA) shows schematic representations of two deep space probes both targeted to orbit the planet Jupiter, which stands at an average distance of 5 AU from the Sun.

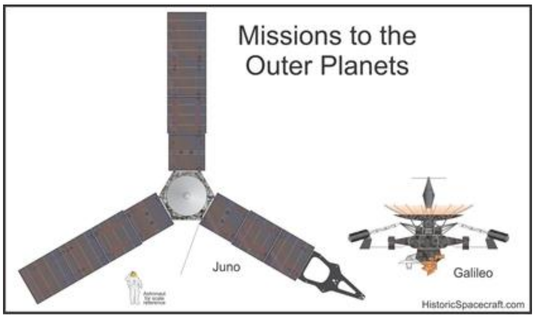


Figure 15: Two different deep space probes to Jupiter

Two very different approaches can be seen. This difference is mainly related to the power generation. The Juno spacecraft uses a large solar array (divided over 3 wings extending far from the S/C) to obtain the required energy, whereas the Galileo spacecraft is powered by Radio-isotope generators, leading to a more compact design.

Next figure shows two possible concepts for a solar power satellite taken from [Smitherman]. The mission of this satellite is to gather as much power as possible, hence the large size, which is then converted and transmitted to ground. So the satellite essentially functions as a space-based power station.

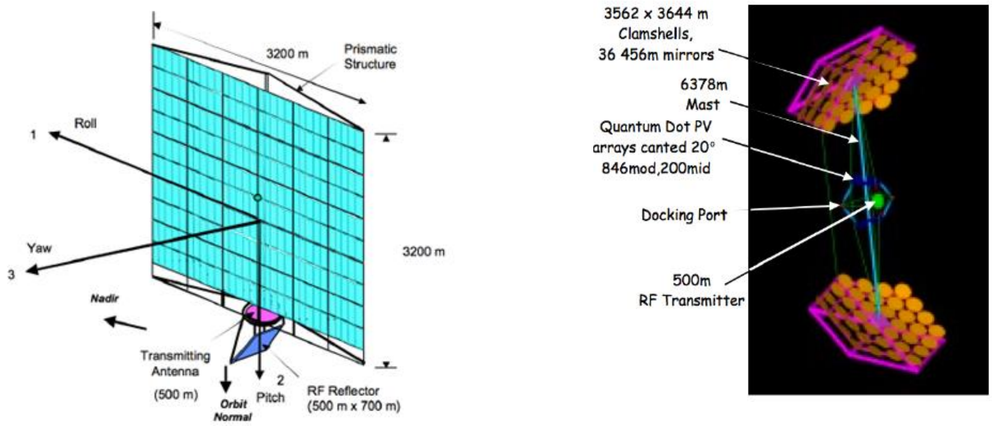


Figure 16: Two possible concepts for a solar power satellite (on left Abacus concept and on right Integrated Symmetrical Concentrator satellite concept)

In both concepts, solar energy is converted in radio-frequency (RF) energy that subsequently is transmitted down to some receiving station on ground. The Abacus concept is for a smaller version as to approach a more manageable mass and scale for assembly and operations. The Integrated Symmetrical Concentrator Satellite is more complex in appearance because it uses reflectors to concentrate sunlight onto a smaller PV array. Note that figure provides some details that can only be filled in once we have performed the first round of design.

Another example of different concepts, but this time for a Mercury orbiter, is given in Figure 17. Top left is Russian electric propelled Mercury P orbiter (with lander on top), Bottom left is European/Japanese Bepi Colombo, bottom right is US MESSENGER spacecraft.

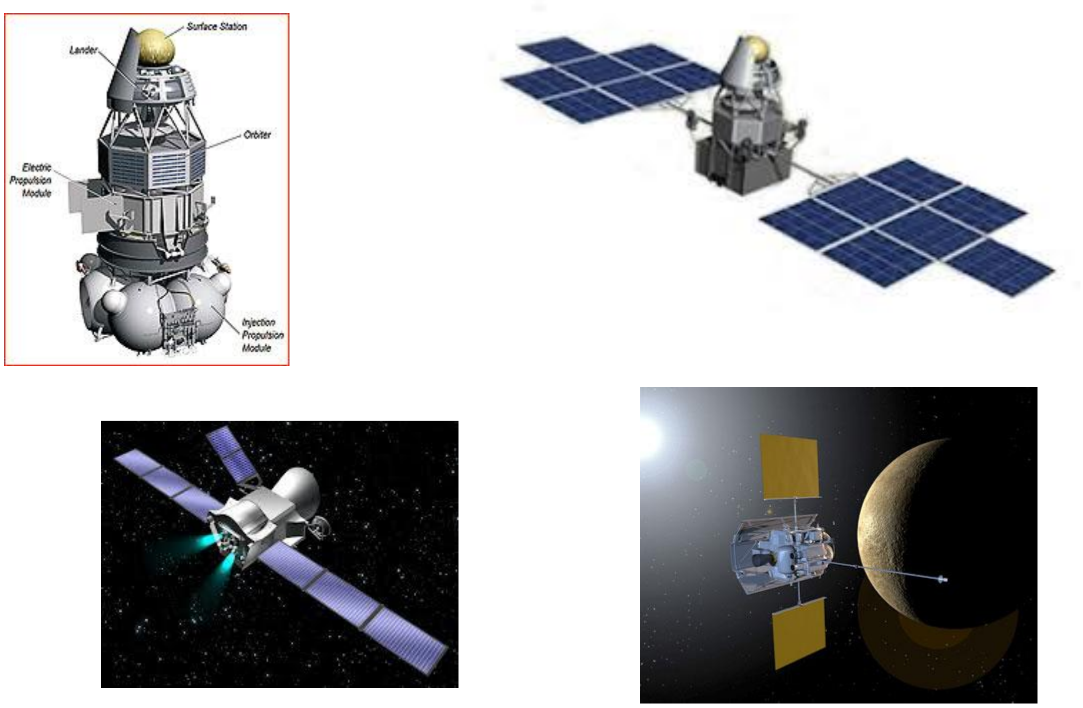


Figure 17: Mercury orbiter concepts (pictures by courtesy of IKI, ESA/JAXA, NASA)

The US MESSENGER spacecraft was injected into a heliocentric orbit by a Delta II rocket using a solid rocket upper stage. It used various gravity assists to reduce its speed relative to Mercury. Final orbit was attained after a 6 year flight by using a large rocket engine to enter into an elliptic orbit about the planet. The multi-flyby process greatly reduced the amount of propellant necessary to slow the spacecraft down, but at the cost of prolonging the trip by many years and extending the distance travelled to 4.9 billion miles.

In contrast the European/Japanese Bepi Colombo spacecraft will be launched by Ariane 5. After injection into a heliocentric orbit, the spacecraft will have a seven-year interplanetary cruise to Mercury using solar-electric propulsion and nine gravity assists to reach the final orbit. Propulsion during the transfer stage is provided by a specially developed electric propulsion transfer stage. This transfer stage makes up the bottom element of the vehicle and is equipped with large solar arrays to provide the necessary electric power.

Mercury P is a Russian project to Mercury that like Bepi Colombo will use electric propulsion. Hence the very large solar array (much larger than for MESSENGER) to provide the electric propulsion system with the required power.

From the foregoing, we should learn that by looking at past missions, we notice peculiarities. By investigating these, we can learn about different ways of achieving mission objectives. Each of these ways (choices/options) needs to be analysed and evaluated for its feasibility in the design at hand. Some important options are:

1. Primary propulsion:
    a. Integral propulsion, where the propulsion system is integrated inseparably in the vehicle
    b. kick stage/orbital transfer vehicle, where the main propulsion function is provided for by a separate stage or vehicle, which is separated after use.
2. EPS (for more details, see chapter 4, EPS section):
    a. Photovoltaic power system
        i. Body mounted array
        ii. Extendible array (rotatable or fixed, single or multiple wings)
    b. RTGs or still other
3. Body shape: Sphere, cylinder, rectangular box, hexagonal/octagonal box
4. 3-axis control, spin control or other (for more details, see chapter 4, section on ADCS)
5. Antennae (for more details, see chapter 4, Communications section)
    a. Rotatable
    b. Fixed (vehicle needs to be steered to allow for pointing)

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 or to perform trade studies include:

- Vehicle total mass and size; is a suitable launcher available? Is it manageable?
- Vehicle power; are the required power levels feasible?
- Vehicle cost; can we stay within the allocated budget?
- Vehicle reliability; can mission success probability be guaranteed?
- Development risk; is insurance needed?
- Vehicle configuration; can we fit on board all payloads such that each instrument has the proper viewing angles and Field of View?
- Etc.

Depending on the requirements that need to be fulfilled of course also other parameters can be included.

3.3 Collect relevant data

Next step is to collect relevant data from comparable spacecraft. For this to succeed, we need to determine what parameters do effect the parameters listed in the previous section, the modelling we are aiming for and what are considered comparable vehicles.

At the early design stages very little is known about the spacecraft. Estimation is done using simple estimation relationships developed based on historical (statistical) data combined with simple relationships from physics and other fields of engineering. This generally is referred to as a first level vehicle sizing.

For this first level sizing we are looking for the main parameters affecting the vehicle characteristics described in the previous section. Thus we are looking for those parameters that greatly determine spacecraft mass, size, cost, etc. Experience shows that for a first estimation of spacecraft parameters, at least the following parameters are important: Type of instrument(s) or type of spacecraft (according to application), number of instruments and per instrument/payload its dimensions (height, length, width), mass, and power. Other important parameters to be considered are mission orbit and mission duration. This is visualized in Figure 18.

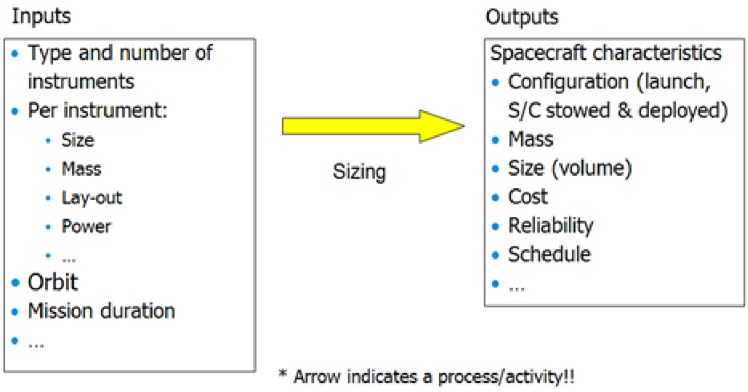


Figure 18: Spacecraft level dimensioning and sizing

How the above parameters relate to the spacecraft parameters to be determined is shown in next table. Note that we are looking for the main parameters only as otherwise estimation methods quickly become complex and time consuming.

Table 14: Some important parameters for first level sizing of spacecraft (orbiters, landers, fly by probes)

Parameter	Parameter affected	Rationale
Payload size;	Spacecraft size;	Large payload requires a large bus to carry it;
Payload mass;	Spacecraft dry mass;	Heavy payload requires a heavy bus to support it;
Payload power;	Spacecraft power;	Payload power is provided by spacecraft;
Spacecraft dry mass;	Spacecraft cost;	Propellants are relatively low cost;
Mission life;	Reliability;	Long life increases probability of failure;
Mission delta-v;	Propellant mass;	High delta-v increases propellant mass;
Failure rate;	Failure probability, reliability;	High failure rate leads to a high failure probability;
Complexity;	Reliability, cost, risk;	More complex vehicles have higher cost, lower reliability and hence higher risk.

Complexity as parameter is difficult to quantify. For first level estimations though it is sufficient to consider that complexity varies mostly with the type of spacecraft and to some extent also with spacecraft mass and spacecraft size. Hence to take into account complexity it is sufficient to distinguish between the various types of spacecraft, like landers, orbiters, and Earth satellites. For the important category of Earth satellites a further distinction can be made, as this category contains a large variety of different spacecraft, see chapter 1.

Classification by mass is also 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 15 provides a distinction of small satellites partly based on data taken from [Surrey].

Table 15: 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 finding answers is the focus of the investigation you are performing and that you develop 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.

An example of typical data collected for a first sizing is given in Figure 19. It is the author's experience that also data on delta-v accomplished by the vehicle in relation to total delta-v budget is important to include.

Past Mission Summary

Mission	S/C dry Mass (kg)	P/L Mass (kg)	Mass Ratio	S/C Pwr. (W)	P/L Pwr. (W)	Pwr. Ratio
		Observatory	**Type**	**Missions**		
XMM	3234	2147	0.62	1000	675	0.68
Integral	3414	2013	0.59	2377	719 (max)	0.30
		Planetary	**Type**	**Missions**		
MEX	510 (71)	116	0.26	1500 [650]	140	0.21
Rosetta*	1322 (~110)	170 (27)	0.11	850@ 5 AU	190	0.22
VEX	633	93	0.15	1100(Venus)	150	0.13

(*) The additional Lander mass is included in the total dry spacecraft mass.
[] Power at maximum distance from Sun. Power available varies depending on Mars position.

Mission Overall Summary

Mission	Launcher	Launch wet Mass (kg)	Orbit (km)	Launch date	Cost (e.c. 2006)	TM (kb/s)
		Observatory	**Type**	**Missions**		
XMM	A5	3800	114000x7000	1999	919	66
Integral**	Proton	3954	153000x9000	2002	397	113
GAIA	Soyuz Fregat-2B	2030	L2	2011	550	5000
		Planetary	**Type**	**Missions**		
MEX	Soyuz Fregat	1223	11560x258	2000	204	38-230
Rosetta	A5 G+	2900	N/A	2004	825	22
VEX	Soyuz Fregat	1241	66000x250	2005	203	28-262

** *Launcher provided by RSA (Russian Space Agency) as part of an international collaboration*

Figure 19: Technical data useful for first sizing [ESA, 2015]

Appendix B provides for a handy collection of spacecraft data that can be used for deriving estimation relationships, see next entry. 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. For instance, data for Earth satellites should not be used to design a lander or deep space prove 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? In this respect, consider also the following remarks:

    - Do not confuse bus-, spacecraft-, payload- and launch mass and dry and wet mass.
    - Do not confuse peak power, average power and Begin Of Life power with End Of Life power.
    - Do not confuse development and production cost
    - Do include year cost with cost data. The cost of an item expressed in year 2000 dollars is quite different from the cost of the same item in year 2014 dollars, due to for instance inflation.
    - Do not confuse bus-, spacecraft-, payload- and launch cost;
    - Do not confuse development and production (recurrent) cost.
    - Etc.

An overview of parameters used in spacecraft engineering and their definitions is given in Table 16.

Table 16: Some vehicle parameter definitions

Parameter	Definition
Vehicle launch mass (VLM)	Gross vehicle mass plus mass of kick stage (if applicable) and mass of launch vehicle adapter[13] (LVA)
Vehicle injected mass (VIM)	Vehicle mass just after separation from the launch vehicle. Launch mass minus mass of LVA
Vehicle gross/loaded/wet mass (VWM)	Total vehicle mass (sum of vehicle dry mass and propellant mass)
Vehicle on-station mass also referred to as vehicle mass in orbit	Total vehicle mass when arriving on-station, i.e. in target orbit when starting operational life (Begin Of Life, BOL)
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 (VEM)	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
Bus or platform (dry) mass	Vehicle (dry) 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
S/C power	Payload power + power needed by S/C subsystems
Payload power	Power required by the payload(s)
EOL power	End of Life power
BOL power	Begin Of Life power
Development cost	Cost associated with the development of a S/C (is one-time or non-recurring cost)
Production cost	Average cost associated with the production of a S/C (is recurring cost)
Operations cost	Cost associated with the operations of a S/C
S/C production time	(Average) time it takes to produce 1 S/C
S/C development time	Time it takes to develop the S/C
Spacecraft life	Maximum duration that the vehicle can function in a satisfactory way. In most missions, life is limited due to a limited propellant load carried on board
Spacecraft reliability (R)	Probability that a spacecraft functions properly over the spacecraft life when used under prescribed conditions
Spacecraft failure probability (F)	Probability that a spacecraft fails over the spacecraft life when used under prescribed conditions
Risk	Probability of failure (F) times impact or severity (S) of failure (Risk = F x S)

3.4 Estimation by analogy

One method to obtain values for the important design characteristics as outlined in the foregoing is through analogy or system similarity based estimating. This is an estimating *technique* that uses the values of parameters and measures of scale from a previous, similar *activity* as the basis for estimating the same parameter or measure for a NEW activity/design. 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*).

As an example, consider the case that we want to design a LEO Earth Observation spacecraft capable of producing black/white (B/S) and color (C) imagery with a spatial resolution better than 1 m, starting in year 2020. The camera to be carried on board of the S/C is estimated to have a mass of 380 kg. Literature shows us that there exists a more or less similar spacecraft, being the Worldview 1 spacecraft. This spacecraft carries the Worldview 60 camera, which is capable of producing black/white and color

[13] Launch Vehicle Adapter (LVA) is the physical structure used to connect the S/C to the launch vehicle. Most launch vehicles offer standard LVA, but sometimes an additional adapter may be needed to connect the S/C to the LVA. This may be referred to as payload adapter as it is considered the responsibility of the S/C engineer.

imagery with a spatial resolution better than 2 m at an orbital altitude of ~500 km. The camera also has a mass of 380 kg. From literature, the following characteristics of the Worldview 1 spacecraft are obtained:

- Launch date: 2007
- Launch mass: 2500 kg
- Mission design life; 7.25 years
- S/C dimensions: 3.6 m (high) and 2.5 m in diameter, span of the deployed panels is 7.1 m.

Analogous estimation than would indicate that for the NEW project, we should also consider that the vehicle has a launch mass of 2500 kg, has the same size, cost, etc.

Another example of analogous estimation may be found in the Mars Express and the Venus Express mission conducted by ESA. After going to Mars (an outer planet), ESA member states decided to also conduct a comparable mission to Venus (an inner planet) thereby conducting the same investigations (reuse of Mars Express payloads) for about the same duration. It would be logical to say that both missions are quite similar and hence we could use the data of Mars Express to generate a first estimate of the mass, size, cost, reliability, etc. of the Venus Express S/C.

Analogous estimation would be almost perfect if the NEW spacecraft would be truly identical to the comparable historical vehicle. However, this would only work if the two spacecraft are truly similar. This in reality is never true as this would come down to the rebuilding of an earlier developed satellite and hence one may doubt about how new the design is? Also over the years, payloads tend to become smaller and smaller and use less power, which further complicates matters. This makes that analogous estimating actually is a form of *expert judgment*. Experts need to assess to what degree some NEW design will be analogous to older designs, most likely using older (obsolete) technologies. 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 (realism/accuracy).

3.5 Parametric estimation

Parametric estimating refers to an estimation technique which is based on the premise that changes in some output (dependent) parameter can be related to the value of an independent variable. For instance, vehicle mass and size depend largely on payload mass and size or vehicle size depends on vehicle mass or spacecraft cost can be related to spacecraft dry mass, etc. In general, we are looking for relations of the form:

$$y = f(x)$$

Here y is the dependent variable and x is the independent variable. Above relation may be of different forms, including linear, power, logarithmic, exponential, etc.

Parametric estimation relies on the use of relatively simple statistical analysis methods, like averaging and 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:

- Statistical analysis that provides a simple average (arithmeticetic or equally weighted mean), and a sample standard deviation (SSD). The latter being a measure for the spread in the data.
- Regression analysis (a means of statistical analysis) that provides for a Most Likely Estimate (MLE), i.e. "best curve fit of data", and standard (percent) error of estimate (SEE) sometimes also referred to as relative standard error (RSE).

Averaging generally is limited to the generation of linear relationships, whereas regression analysis also allows for other types of relations to be developed.

A 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. A further advantage of parametric estimation is that the estimation relationships can be nicely combined with (more fundamental) relationships from physics (e.g. orbital mechanics), RAMS engineering, etc.

Both similarity based and parametric estimation use data of more or less similar designs as a starting point for the estimations. Important is 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). Moreover, the latter method can also be applied in case no historic missions can be found for estimation by similarity.

In this section, attention is focussed on developing/generation of estimations and/or estimation relationships using historical data. The use of simple relationships from physics etc. will be illustrated in later sections. Below, we will introduce first estimation using averaging and second using regression analysis, i.e. "best curve fit of data".

Estimation based on using arithmetic mean and standard deviation
In this section, we will introduce the use of averaging to develop a linear relation between an independent and a dependent variable. We will do this by way of examples.

The first example is on the development of a relationship between S/C dry mass (the dependent variable) and payload mass (the independent variable). We will do this based on the presumption that payload mass is a fixed percentage or ratio of vehicle dry mass.

$$M_{dry} = A \cdot M_{payload} \qquad [1]$$

Please consider what other parameters may affect the dry mass of the S/C. Consider e.g. the propellant mass carried and more specifically the delta-v capability of the craft.

To allow for estimation of the value of the parameter A (which is here defined as the payload mass-to-dry mass ratio), we have collected historical data on payload mass and dry mass as can be found in literature. These data are given in Table 17. The table also provides the calculated value of the payload mass-to-dry mass ratio.

Table 17: Sample mass data

Spacecraft	Payload mass	Vehicle dry mass	Mass fraction
	(kg)	(kg)	(-)
A	220	750	0.29
B	36	250	0.14
C	43	420	0.10
D	12	112	0.11
E	25	140	0.18

Using the data from the foregoing table, we find an average payload mass to spacecraft dry mass ratio of:

$$\bar{x} = \frac{(0.29 + 0.14 + 0.10 + 0.11 + 0.18)}{5} = 0.16$$

An average mass fraction means that the constant “A” in the relation = 1/0.16 = 6.25 kg of dry mass for every 1 kg of payload mass. Hence, it follows that vehicle dry mass is 6.25 times payload mass. So for a payload mass of 100 kg, we find a vehicle dry mass of 625 kg.

Comparing the average mass fraction of 0.16 with the mass fractions of the individual vehicles some spread is noticed. As a measure for the spread we determine the SSD using:

$$SSD = \sqrt{\frac{1}{n-1} \cdot \sum (x_i - \bar{x})^2} \qquad [2]$$

Filling in numbers gives:

$$SSD = \sqrt{\frac{1}{5-1} \cdot \left[(0.13)^2 + (-0.02)^2 + (-0.06)^2 + (-0.05)^2 + (0.02)^2\right]} = 0.08$$

It follows an SSD of ~0.08 (more exactly 0.077) or roughly 50% of the average value of the fraction. When assuming a normal distribution[14] of the data, see appendix E, this means that roughly 65% of all predicted values are within 50% (once the sample standard deviation) of the predicted value. So for a 100 kg payload mass, we find a dry vehicle mass of 625 kg. With a 65% probability, the dry mass is in the mass range 417 kg – 1250 kg. That is how accurate it is!

Now when collecting data the issue might arise on how many data points are needed. For instance, when taking only the first data point of Table 17, we have a fraction of 0.29, when adding the second, the mean value is 0.22 and so on. The issue is when to stop or what is a statistically meaningful number? This question is dealt with in numerous text books on statistics. For now, it is advised to use at least 10 data points.

A second example of using averaging is based on the presumption that insurance cost is related to the insured value. Some typical data for the period 1980 to 2002 is shown in Figure 20. From the data, we find an average insurance rate over the period up to 2002 of 16.5% of the insured value. The figure also clearly shows that from year to year variations do occur. To take into account these variations, the SSD has been determined. A very approximate (exact figures are hard to obtain from the figure) determination gives an SSD of 5.5%, meaning that 95% of all interest rates are in the range 5.5 – 27.5% (average value ± 2 times SSD).

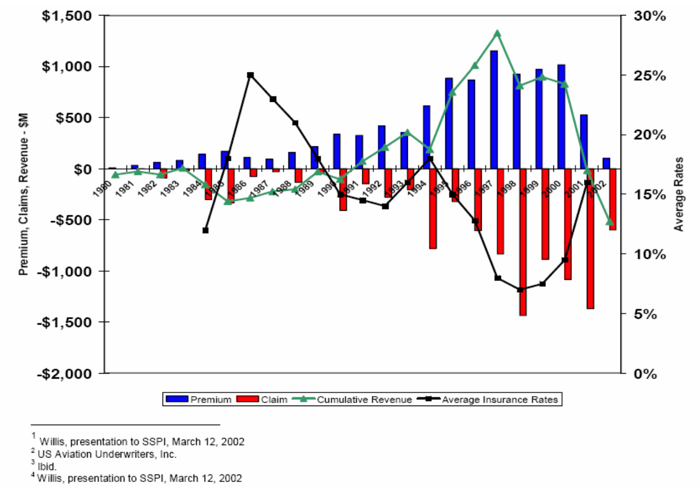


Figure 20: Insurance rates [Futron, 2002]

Above two examples are just two examples. Many more relations can be considered and some may even be better than the ones presented here. This though is left for the reader to explore for her/himself.

[14] Other distributions may exist. For the purpose of this work this will be left out of consideration.

Estimation using regression analysis and relative standard error (RSE)
Next to simple averaging, we can 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 21, which shows S/C gross (or total) mass (dry mass + propellant mass) versus payload mass for interplanetary orbiter S/C. Below, we first discuss the regression line itself. Following, we will discuss the data spread about this line.

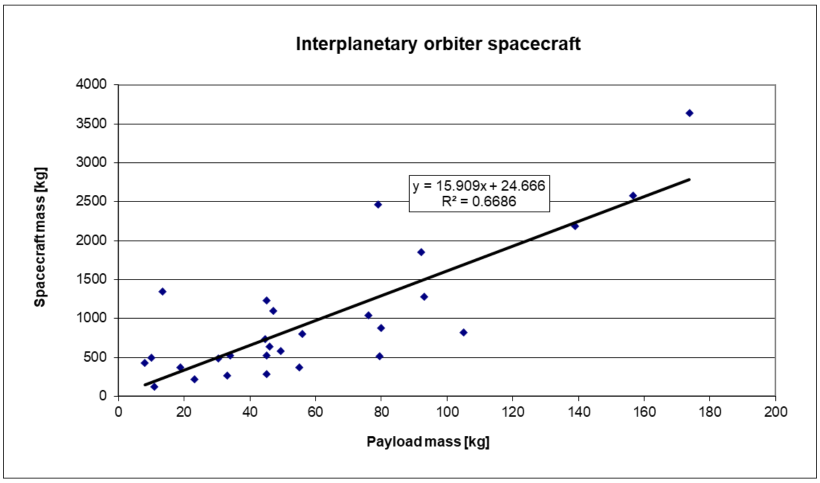


Figure 21: Gross mass of some planetary spacecraft

The relationship found is referred to as estimating relationship and in this case it is a Mass Estimation Relationship (MER) as the dependent variable in the relationship is mass. 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. More details on this relationship can be found in appendix C.

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 (relative) standard error of estimate (RSE), defined as:

$$RSE = \sqrt{\frac{1}{n-m} \cdot \sum_{i=1}^{n} \left(\frac{y_i}{f(x_i)} - 1 \right)^2} \qquad [3]$$

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 RSE. 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 RSE is zero.

Notice that in the way that the RSE is defined here, it represents 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, as for the SSD, 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 22 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 RSE is calculated as a relative error. For the data plotted in the figure, the RSE is ±38%.

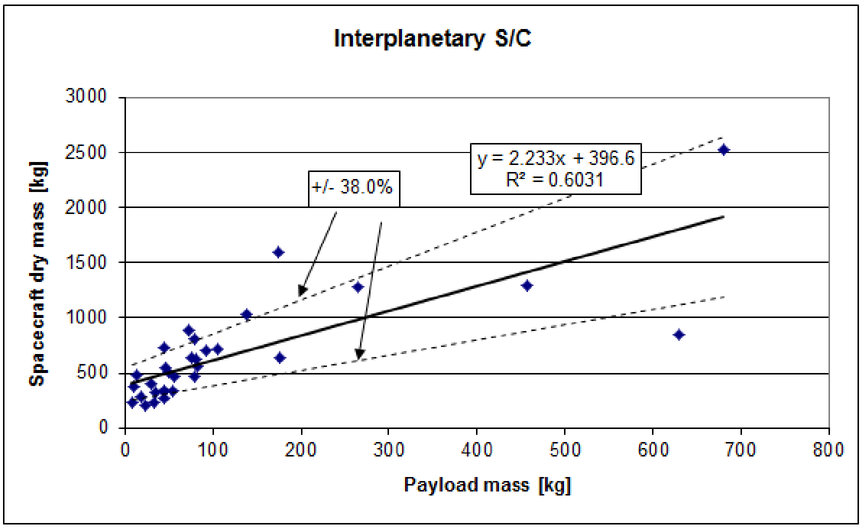


Figure 22: Interplanetary spacecraft dry mass

For improved understanding, we have also plotted two partly dotted lines in the figure showing the ±38% range about the most likely value. Notice that almost all points lie neatly within this range except for 6 points (out of 23). The percentage number of spacecraft that lies within these two lines agrees nicely with the important 68-95-99.7 rule holding for a normal distribution. It essentially says that 68% of all data points are within one standard error of the mean, 95% is within two standard errors and 99.7% is within 3 standard errors of the mean. So the lower the standard error is the closer the solutions are to the MER. For further explanation of this much used rule, read appendix E. later in this work, we will use this rule to determine appropriate margins (or contingencies).

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 = (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.

Now that we understand how estimation relationships can be generated using statistical techniques, we can turn to discussing a method for spacecraft preliminary sizing.

3.6 Method for spacecraft preliminary sizing

In this section, we will present a method for estimating a number of important characteristics of spacecraft. Characteristics include such parameters like spacecraft launch mass, spacecraft loaded mass, spacecraft power, spacecraft size, cost, reliability and risk. Note that all these parameters are linked to the earlier mentioned requirements. Key is that in our estimations we aim to ensure selecting values for the independent variable that ensure that the dependent variable fulfils the requirement(s). 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.

Launch mass estimation

Launch mass (in combination with the target orbit) is critical for launch vehicle selection. This in turn is important for ensuring that launch cost are in line with the general cost requirement. 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 5, and a launch vehicle (or payload) adapter (LVA), see Figure 23.

Figure 23: Payload adapter (courtesy RUAG)

Of these, a kick stage (or service module) 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.

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:

$$M_L = M_{SC} + M_{KM} + M_{LVA} \qquad [4]$$

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

$$M_L = M_{SC} + M_{LVA} \qquad [5]$$

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 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 1which 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.

Method A: Estimating wet mass based on payload mass only
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 24, some of these relations are 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.

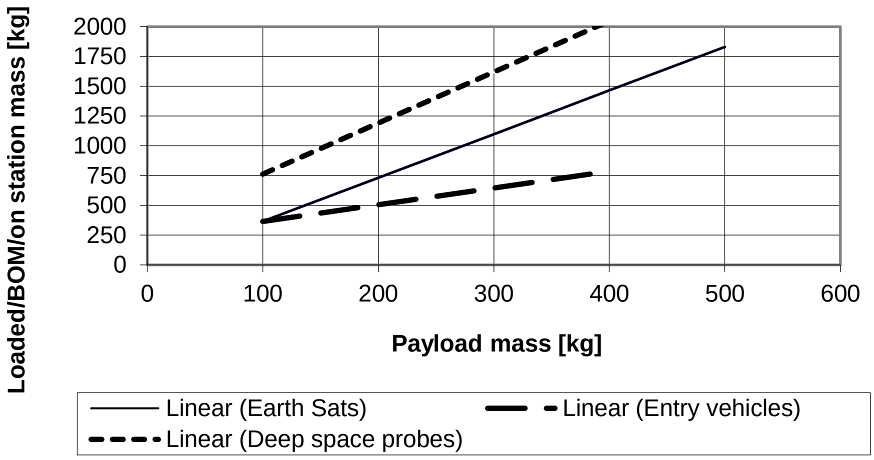


Figure 24: Wet mass versus payload mass for various types of spacecraft

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 Satellites 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:

$$\left(M_{SC}\right)_{wet} = \left(M_{SC}\right)_{dry} + M_{propellant} \qquad [6]$$

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 these masses do show small differences. 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:

$$\left(M_{SC}\right)_{dry} = a \cdot M_{payload} + b \qquad [7]$$

The constants a and b 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 25.

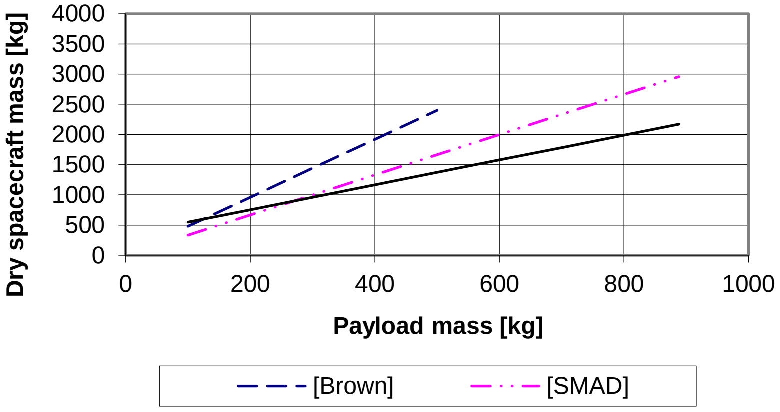


Figure 25: Comparison of 3 relationships for estimating dry mass of Earth Satellites

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 18 the estimated results are compared for the ERS-1 spacecraft to the actual value as reported on [eoPortal Directory].

Table 18: 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 is estimated using the rocket equation[15] as given in Equation [8].

$$\Delta v = w \cdot \ln\left(\frac{M_o}{M}\right) \quad \rightarrow \quad \left(\Delta v\right)_e = w \cdot \ln\left(\Lambda\right) \qquad [8]$$

[15] 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).

$$\Lambda = \frac{M_o}{M_e} \qquad [9]$$

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 system used. For details on the derivation of the rocket equation, you are referred to the Section on "Launch vehicle trajectories" in AE1110-II.

Using:

$$M_o = M_e + M_{propellant} \qquad [10]$$

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

$$M_{propellant} = M_o \left(1 - e^{-(\Delta v/w)}\right) \qquad [11]$$

Or:

$$M_{propellant} = M_e \left(e^{\Delta v/w} - 1\right) \qquad [12]$$

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 – 2200 m/s

For more details on attainable rocket exhaust velocities see the later section on propulsion.

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 needs 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 a 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 reaction control system (RCS). 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 26 and is valid over a spacecraft wet mass range of 500-2400 kg.

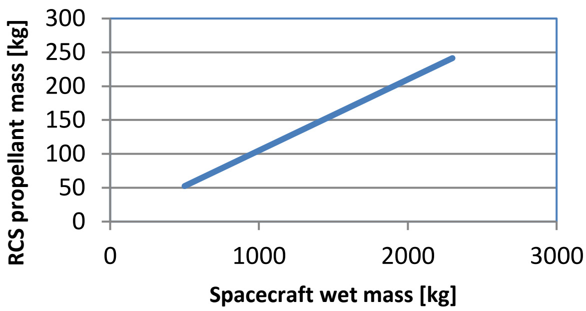


Figure 26: RCS propellant mass versus spacecraft wet mass

In case of a large propellant load, i.e. large Δv maneuvers (in excess of say 300 m/s) the approach described so far may not be realistic as we implicitly assume that we can increase propellant mass without any effect on dry mass. However, with increasing propellant load, tanks get larger and hence heavier and hence S/C dry mass increases. This can be taken into account by assuming that the bus dry mass consists of one part that depends on propellant mass and one part that is independent of propellant mass. We obtain:

$$M_e = M_{pay} + M_{bus} \wedge M_{bus} = M_{propulsion} + M_{other\ subsystems} \quad [13]$$

Relations for propulsion subsystem (dry) mass in relation to propellant mass can be taken from e.g. the "Propulsion" section, see chapter 4. Later sections also allow to validate our assumption that the mass of the other systems is (largely) invariant with propellant mass. One issue remaining is how to determine the mass of the other S/C subsystems. For now, it is suggested to use the dry mass estimated earlier as a starting point. In later design iterations, when more knowledge is available, this estimate can be updated.

B) Kick stage mass and adapter mass estimation

In case of large maneuvers (in terms of Δv), we may also consider adding a separate propulsive stage (kick stage) that carries the required propellant. This has as advantage that when arriving on station, the stage can be discarded reducing the mass of the S/C itself. To determine the mass of such a stage, we consider kick stage mass as the sum of kick stage propellant mass and kick stage dry mass:

$$M_{KM} = \left(M_{KM}\right)_{Dry} + \left(M_p\right)_{KM} \quad [14]$$

One relation for the dry mass estimation of chemical rocket propelled kick stages is [SMAD]:

$$\left(M_{KM}\right)_{Dry} = 17.5\% \text{ of } \left(M_p\right)_{KM} \quad ; \quad \% \text{ range is 10-25\%} \quad [15]$$

This relation essentially means that kick stage dry mass typically is in range 10-25% of propellant mass carried by the stage. For estimation, it is advised to use the mid-range percentage value (here 17.5%) to estimate kick stage dry mass. Assuming a 1 sigma accuracy, it shows a 1-sigma uncertainty of about 40-45%. For further info, see later section discussing means of propulsion.
Propellant mass can be estimated using the rocket equation, similar to as described earlier for the spacecraft. The one critical element is to determine how much of the velocity change is to be delivered by the kick stage and how much by the spacecraft itself.

Adapter mass can be estimated using:

- [SMAD]:

$$M_{LVA} = 1\text{-}2\% \text{ of injected mass} \quad [16]$$

- [Brown]:

$$M_{LVA} = 0.0755 \cdot M_L + 50 \quad (M_L \text{ is in the range 200-3500 kg}) \quad [17]$$

Power estimation

S/C 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 S/C, 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 taken from this appendix are plotted in Figure 27 over their range of validity. For high payload power they are plotted in Figure 28.

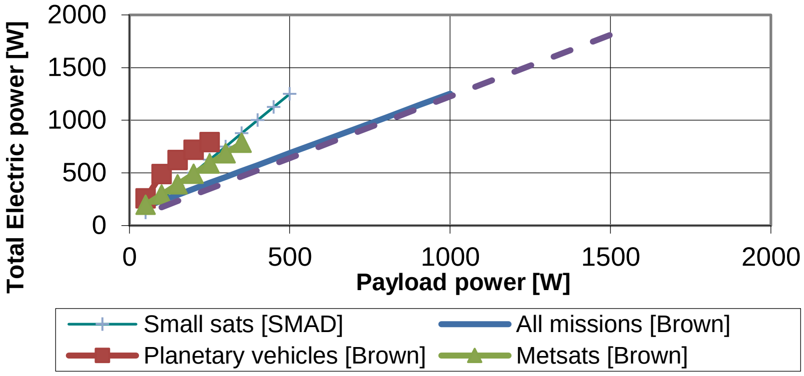


Figure 27: Total electric power versus payload power for various S/C types (low power regime)

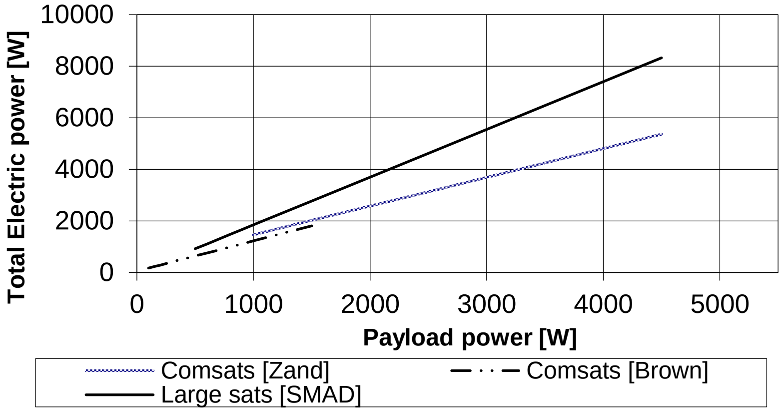


Figure 28: Total electric power versus payload power for various high power S/C types

Results again show the differences that exist between the various spacecraft types. Hence, it is essential that when you select a relationship that you also check its validity, i.e. does the PER give sufficiently accurate results for vehicles similar to the one being designed.

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:

$$V = \frac{(M_{S/C})_{wet}}{\rho} \qquad [18]$$

Here 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 envelope.

For instance:

- Large Earth (orbiting) satellites ([SMAD], 75 S/C, 136 kg < total mass < 3625 kg): Mass density ranges from 20-179 kg/m^3, average mass density is 79 kg/m^3

Various different (and more accurate) size estimation relationships are collected in appendix C, see section on size estimation.

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].

$$L_b = V^{1/3} \text{ or } \mathrm{L_b} = \left(\frac{M}{\rho}\right)^{1/3} \qquad [19]$$

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

$$L_b = 0.25 \cdot M^{1/3} \qquad [20]$$

Above relations are relatively simple and easy to reproduce. However, in later design iterations, once the body shape becomes clear, more exact relations can be used. For now, we will limit ourselves to using the above defined body linear dimension.

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:

$$A_b = L^2 \qquad [21]$$

Solar array area A_a (also of importance for drag estimation) and solar array mass M_a for <u>Earth</u> orbiting spacecraft can be estimated using next relations taken from [SMAD]:

$$A_a = \frac{P}{P_\delta} = \frac{P}{100\frac{W_e}{m^2}} \qquad [22]$$

$$M_a = \frac{P}{P_{sp}} = \frac{1}{25\frac{W_e}{kg}} \cdot P = 0.04\frac{kg}{W_e} \cdot P \qquad [23]$$

Here P_δ and P_{sp} are array power density and array specific power, respectively.

Both relations provide a very conservative estimate and are valid for low efficiency Silicon panels taken @ 1 AU at end of life conditions for which we have a power density of 100 W_e/m^2 and an array specific power of 25 W_e/kg. Values for other panel types in relation with panel life can be obtained from the section on electrical power subsystem design in Chapter 4.

Values for other distances to the Sun can be determined using Equations [24] and [25], With d is distance to Sun expressed in AU.

$$P_\delta = \frac{100}{d^2} \ [\mathrm{W}_e/\mathrm{m}^2] \qquad [24]$$

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

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. A more generic discussion can be found in chapter 4.

From the array area now also the length (in span wise direction) and the array height can be determined:

$$A_a = \text{array length x array height (L x H).}$$

Array height usually depends on spacecraft height, whereas length may be distributed over two or more wings.

Cost estimation

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

$$C_{total} = C_{development} + N \cdot C_{production} \qquad [26]$$

And:

$$C_{SC} = \frac{C_{development} + N \cdot C_{production}}{N} \qquad [27]$$

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 finds 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 data obtained from literature (see e.g. appendix B) 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:

$$C_{SC} = 0.3531 \cdot \left(M_{S/C}\right)_{dry}^{0.839} \qquad [28]$$

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. Notice that if we divide the above relationship by S/C dry mass, we obtain spacecraft specific cost, i.e. spacecraft cost per unit of spacecraft dry mass). It follows that small S/C have a higher specific cost than large S/C. 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 larger and hence heavier, this leads to lower cost per kg.
In appendix C various other (including more accurate) CERs can be found as well as information on specific cost of NASA planetary and science S/C. To obtain S/C cost from specific cost we simply need to multiply specific cost times S/C dry mass.

When costing spacecraft, we need 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:

$$Infl = \left(1 + \text{inflation rate / yr}\right)^{\text{\# of years}} \qquad [29]$$

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. So over a 12 year period the cost of some item has been increased with a factor 1.268. So using above relationship we can take into account inflation at least assuming that inflation rate is constant over the period considered. In case inflation rates vary from year to year, the above simple relation is to be used with caution.

> *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* [28]*, gives a cost estimate of 116 M$. Of these the latter is considered less accurate as the RSE for this estimate is much larger than the RSE of the former.*

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 29 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.

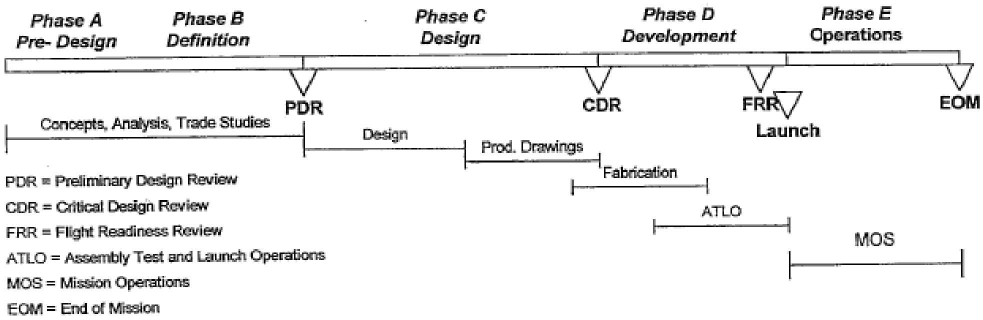


Figure 29: Typical phases in the life cycle of a spacecraft and their distribution in time [Brown]

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 19.

Table 19: 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 Science S/C	30 months 36-72 months

Reliability

Most items do fail at some point in time, and hence 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. Hereafter, we will limit ourselves to those failures that bring immediate failure of the total spacecraft, i.e. the major or catastrophic failures only.

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).

It can be reasoned that the probability of failure (F) of a spacecraft during the mission life is given by:

$$R = 1 - F \qquad [30]$$

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) [Fortescue]. Mathematically, assuming a constant failure rate, 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[16]:

$$R = e^{-\lambda t} \qquad [31]$$

From Eq.[31], 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.

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:

$$\lambda_{total} = \sum \lambda_i \qquad [32]$$

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.[31], it can be shown that spacecraft reliability follows from payload reliability and bus reliability and vice versa according to:

$$R_{SC} = R_{Payload} \cdot R_{Bus} \qquad [33]$$

Equation [33] actually means that for the spacecraft to operate, both the payload and bus should function correctly. If one of the two fails, the spacecraft fails.

Another important point is that if we have 1000 of the same (non-repairable) items, then 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

[16] In case the failure rate is given in number of failures per use cycle, the time t in relation [31] is replaced by the total number of use cycles during its life

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.[31] 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 [31] 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):

$$Risk = F \cdot Severity \qquad [34]$$

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 30 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.

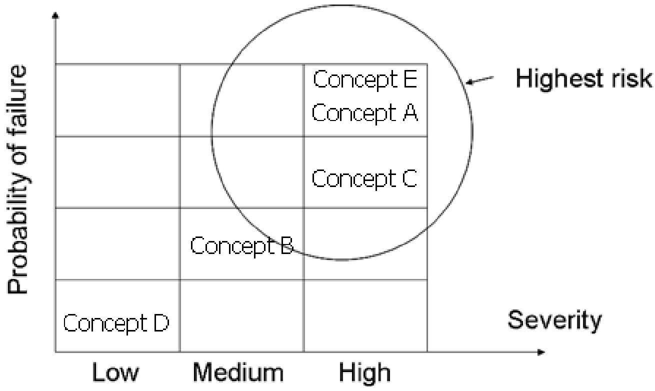


Figure 30: Risk map

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 [34] 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:

$$Insurance = 0.175 \cdot C_{S/C} \qquad [35]$$

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.

3.7 Design margins

In the foregoing, estimation of spacecraft parameters in an early design phase has been dealt with. Given the nature of the estimations, all estimations have some uncertainty/spread. To allow for incorporating these uncertainties in the design, the designer usually adds a design margin (contingency or safety factor), thereby providing room for growth resulting from design definition and development without the need of major redesign, i.e. extensive design modifications. Nobody wants a spacecraft that is too heavy or too large for a launch vehicle to carry into space and/or to reach its destination.

Design margins can be applied to all parameters of interest introduced in the foregoing, including mass, power, size (volume) cost, reliability (or life), Δv, I_{sp}, etc.

A common definition for design margin is:

$$\text{Design Margin} = \text{Total Capability} - \text{Current Best Estimate} \qquad [36]$$

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

Margins can be reduced in later stages of the design when uncertainty decreases.

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. Still others, like NASA, JPL, ESA, etc. have done the same. Next figure shows three tables that provide S/C design margin percentages as recommended by different organizations, including AIAA. From this figure we clearly see that margins decrease towards the later design stages. The main reason for this is the reduced design uncertainty due to the increasing design maturity. Another point that can be taken from the data is that the recommended margin may differ per parameter. This has to do with that some parameters are easier to determine accurately than others. For instance, it is much easier to estimate mass, volume and power of some item than to estimate its cost, reliability, life, risk and or development time. Hence the latter usually are provided with larger margins.

Mass Contingency Guidelines

Source	Relative to:	Phase B Start	At PDR	At CDR
NASA "Green Book" [6]	Flight System	35%	30%	25%
Goddard Gold Rules [7]	Flight System	25%	20%	15%
JPL Design Principles [8]	Flight System	30%	20%	10%
AIAA Standard [9]	Flight System	30%	21%	12%

Power Contingency Guidelines

Source	Relative to:	Phase B Start	At PDR	At CDR
NASA "Green Book" [6]	Flight System	35%	30%	20%
Goddard Gold Rules [7]	Flight System	25%	20%	15%
JPL Design Principles [8]	Flight System	30%	20%	15%
AIAA Standard [10]	Flight System	22%	15%	10%

Cost Contingency Guidelines

Source	Relative to:	Phase B Start	At PDR	At CDR
NASA "Green Book" [6]	Mission	35%	30%	20%
GSFC GPR 7120.7 [11]	Mission	30%	25%	25%
JPL Design Principles [8]	Mission	30%	25%	20%
JSC Cost Handbook (Within SOTA) [12]	Flight System	35%	25%	20%
JSC Cost Handbook (Beyond SOTA) [12]	Flight System	50%	25%	20%

Figure 31: Typical S/C design margins [NASA]

Noteworthy from the data in the figure is also that recommendations with respect to the margin percentage value to be selected may differ depending on the recommending organization. This says two things. The first one is that, when designing a S/C for some organization like NASA, and ESA or company, the designer should make sure to known the customer's recommendations. The second thing is that apparently uncertainty is thus uncertain that different recommendations exist next to each other.

Not clear from the data, but also noteworthy is that for a completely new and unique spacecraft larger margins are needed than for a S/C that is partially based on an existing S/C. This is left for later studies.

A more basic way of determining margins is to use available data on SSD or RSE to determine appropriate margins and assuming e.g. that data values are normally distributed about the average. In the case of a normal distribution, we find that 68% of all probable outcomes are in a range of +/- 1 SSD (or RSE) about the average and 95% of all probable outcomes are in a range of +/- 2 SSD (or RSE) about the average, and 99.7% are in a range of +/- 3 SSD (or RSE). This is referred to as the 68-95-99.7% rule also referred to as the empirical rule. See appendix E for some further information on the normal distribution and read about double-sided and single-sided confidence bounds. In case of a single sided confidence bound as for example when considering maximum spacecraft mass, we could select a margin equal to two times the SSD (or RSE) to ensure that only in 16% 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 2 SSD or even 3 SSD.

3.8 Example sizing

In this section an example is given of a first 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 should 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*
- *Launch vehicle data:*
  - *Maximum diameter under fairing: 3 m*
  - *Vehicle is injected into final orbit by launcher, so no separate kick stage is needed.*
  - *Maximum launch mass into designated orbit: 2500 kg*

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.[16] or [17]. For now we use [16] 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. *Design margin (available): The launch vehicle selected is capable of launching a S/C of 2500 kg in the designated orbit. The spacecraft currently being designed has a launch mass of 1908 kg. The design margin for this spacecraft is 2500 kg – 1908 kg = 592 kg or 592 kg/1908 kg = 31.0%.*
7. *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*
8. *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.*
9. *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)*
10. *Solar panel mass: M_a = 0.04 (518 W) = 20.7 kg*
11. *For the body moment of inertia (kg-m^2), we find using Equation* [38]*:*

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

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

Note that when using the more exact relation [37] 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%.

12. *Cost estimate is determined using the relation* [28]. *For a dry mass of 1468 kg (including the LVA), we obtain: $0.3531\ (1468)^{0.839}$ = US$ 160.2 million (FY 2000)*
13. *Development time is estimated using Table 19. Text indicates that for a science mission phase C/D typically ranges from 36-72 months. Taking the mean value, we find for the duration of the C/D phase a value 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.*
14. *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. In case we use 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.*
15. *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).*
16. *Insurance cost: Insurance cost is estimated based on* Figure 20, *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).*
17. *Design margin (required): For all the parameters calculated in the foregoing, we introduce a design margin. In case RSE and/or SSD are not known, these margins can be taken equal to the recommended values, see earlier given figure. However, in case RSE (or SSD) are known, we can use these values to determine an appropriate margin. Here we will demonstrate the method by assuming an RSE of 15% on the estimated dry mass of the spacecraft and an SSD of 10% on propellant mass. Selecting a 1 SSD (or RSE) range, we find for the dry mass (1440 kg) a mass margin 15% x 1440 kg = 216 kg. For the propellant mass of 440 kg, we find a mass margin of 10% x 440 kg = 44 kg. Assuming that the uncertainty on inert mass and propellant mass are not correlated, see appendix E, it follows for the total margin a value of 220 kg. In case the two estimates are correlated, we can simply add up the mass values and find a mass margin of 260 kg. In case experience shows that a 1 RSE (or 1 SSD) single confidence bound is not sufficient, you can always select a 2 RSE (or SSD) confidence bound. In that case the total margin increases to 440 kg (uncorrelated) or 476 kg (correlated).Since the correlated case is most critical, this is usually used to set the margin.*
18. *Comparison of this required (recommended) design margin with the available design margin shows that both cases are within the available margin. This is deemed sufficient for the phase we are in now.*

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.

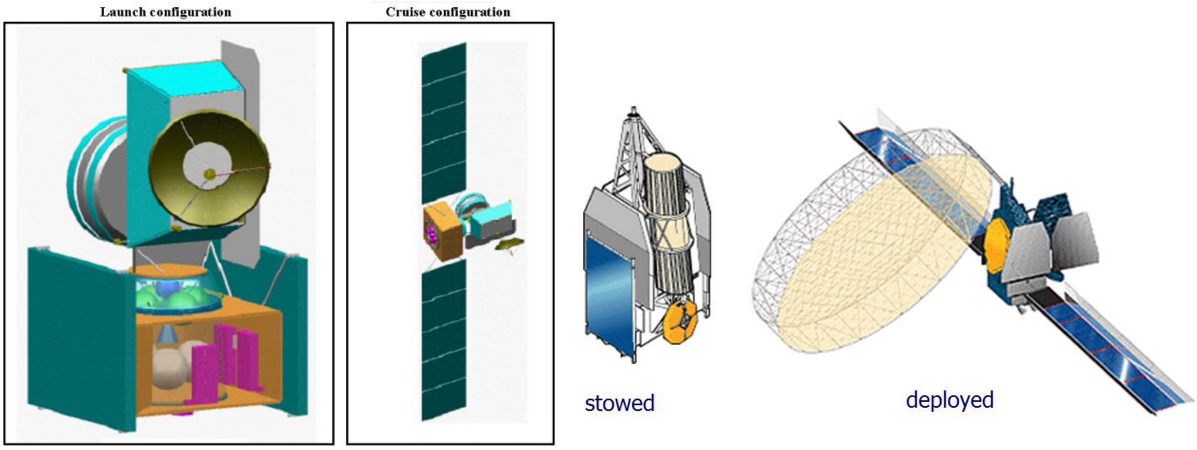
3.9 Quick-look spacecraft configuration

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. How a spacecraft looks like depends largely on:

- The mission and more specific the payload carried on board

- How the vehicle is powered and the amount of power needed
- How the vehicle is stabilized
- Whether a propulsion system is needed or not and how much propellant is to be carried on board
- Position of target, Earth and the Sun relative to the spacecraft
- Etc.

For all spacecraft at least 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 32 show launch configuration of Bepi Colombo S/C and Thuraya S/C as well as the cruise configuration (Bepi Colombo) and the on-station configuration of Thuraya S/C. Clearly noticeable is that in launch configuration, panels and appendages are in retracted position, which helps for the spacecraft to survive the launch loads. When in space, panels and antennae are deployed.



Bepi Colombo S/C (Image credit: ESA) Thuraya S/C (image credit: Boeing)

Figure 32: Different configurations of specific spacecraft.

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. Next figure shows drawing of configuration of an Earth Observation S/C with indicated in figure the direction in which the vehicle is moving as well as the direction of the Sun and Earth. Notice that the solar panels are pointed to the Sun, so that they are able to receive power. Antennae of various instruments are all pointing to Earth, which is of course related to its being an Earth Observation satellite. The main body measures 982 mm x 982 mm x 650 mm.

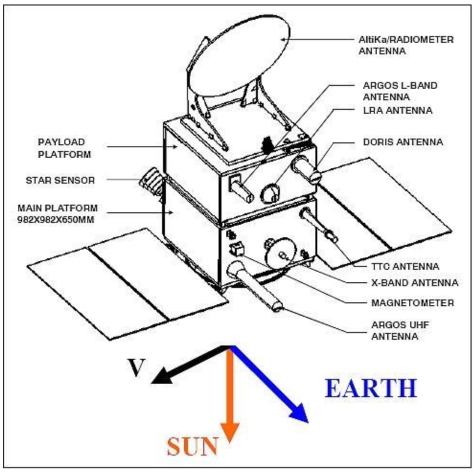


Figure 33: Configuration of Earth observation S/C (SARAL) in relation to Earth and Sun direction (Image credit: ISRO and CNES)

Initially simple sketches will do, like the ones shown in Figure 34, but some measure of the size of the spacecraft (its main dimensions) shall be incorporated.

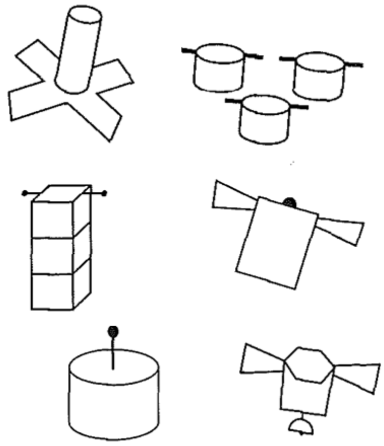


Figure 34: Simple sketches showing spacecraft configuration options

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. More detailed information will follow later in the course reader.
- 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*	*$\pi/4$ 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.

A general goal when configuring the spacecraft is to limit the MMOI as much as possible. For spinning spacecraft, it is important that the MMOI about the spin axis is largest as otherwise the spin motion is not stable, see e.g. [Larson].

For more details on how MMOI affect the rotational motion of the spacecraft, see later in this lecture series (section on ADCS). For now we just focus on determining these properties for a simplified S/C geometry. We will determine S/C MMOI for both the un-deployed state (on the launch vehicle) and the deployed state (after separation with all extendible items extended. In the method described hereafter, we will assume:

1. S/C body is of cubical plan form with CoM in geometric center for both the deployed and un-deployed state. Effect of deployment on main body MMOI and CoM is neglected;
2. S/C is equipped with two identical wings on opposite sides of the spacecraft and with rotational axis going through the S/C CoM.
3. x-, y-, and z-axes, see figure hereafter, are principal axes;
4. Mass of body (un-deployed state) is homogeneously distributed;
5. Solar array is of square plan form with CoM in geometric center and negligible thickness;
6. Mass of array (when deployed) is homogeneously distributed;
7. Each wing is held by a yoke of length L (equal to linear dimension of body) holding the array at a distance from the S/C body
8. 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.

1) MMOI un-deployed S/C
For a cubical spacecraft (or spacecraft body) of mass M (homogeneously distributed) and linear dimension L follows for the body moment of inertia (kg-m^2):

$$I_b = \frac{1}{6} \cdot M \cdot L^2 \qquad [37]$$

For a mass density of say 79 kg/m^3 (a realistic average value, see earlier section on spacecraft size estimation), we obtain:

$$I_b \approx \frac{1}{6} \cdot M \cdot \left(0.233 \cdot M^{1/3}\right)^2 \approx 0.01 \cdot M^{5/3} \qquad [38]$$

2) MMOI deployed S/C
Deploying solar wings or other deployable items (antennae, etc.) can substantially increase the MMOI of a spacecraft as compared to the un-deployed state.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 35.

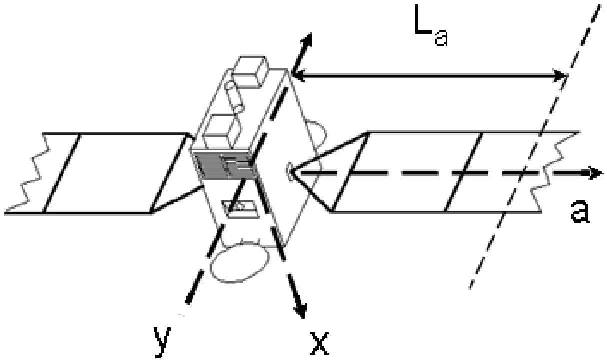


Figure 35: Schematic spacecraft representation for mass moments of inertia calculation (adapted from [SMAD])

First we estimate solar array mass and size using the relations [22] and [23]. Next we compute:

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

$$L_a = \frac{3}{2} \cdot L + \frac{1}{2}\left(\frac{A_a}{2}\right)^{1/2} \qquad [39]$$

Where L is the earlier introduced linear dimension of the S/C and A_a is the total solar array area. Note that here we assume the solar array area is distributed over two symmetrical wings and that each wing is of a square planform. The latter is rare, but for now serves to illustrate the method. We also assume that, e.g. to prevent shadowing of the solar array by the S/C body, a beam (yoke) is holding the wings at some distance from the S/C body. The length of this yoke has been taken here equal to the linear dimension of the spacecraft.

Moment of Inertia (kg-m^2) perpendicular to plane of solar array (see for MMOI relations of simple shapes appendix E):

$$I_{ax} = \left(L_a{}^2 + \frac{A_a}{12}\right) \cdot M_a \qquad [40]$$

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

$$I_{ay} = \left(L_a{}^2 + \frac{A}{24}\right) \cdot M_a \qquad [41]$$

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

$$I_{aa} = \left(\frac{A_a}{24}\right) \cdot M_a \qquad [42]$$

MMOI of spacecraft in deployed condition can now be estimated by summing the S/C body MMOI and the solar array MMOI. This is not 100% accurate as the mass of the array is also included in the S/C body MMOI calculation and when deploying the array this may affect the position of the center of mass of the S/C. For the purpose of this course both effects are hereafter neglected unless specifically indicated otherwise.

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

Inputs (see results from page 63):
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) = $(5.12^2 + 5.2/12) \times 20.7 = 553$ kgm^2.*
4. *MMOI about S/C body y-axis (perpendicular to array axis) = $(5.12^2 + 5.2/24) \times 20.7 = 549$ kgm^2.*
5. *MMOI about S/C body z-axis (parallel to array axis) = $(5.2/24) \times 20.7 = 4.5$ 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.

From the simple example we can see the effect of extendible items on the MMOI is not to be neglected. Still it remains a simplified case. In reality many variations exist. For instance there are many different shapes of the body varying from cubical to cylindrical and rectangular boxes with rectangular or hexagonal or octagonal ground plane. Also some S/C have just a single wing or more than two wings and the wings may be attached directly to the body using hinges and sometimes even fixed on to the body. Also other extendible items, like large radar and/or communications antennae) may need to be accounted for. In case of different vehicle shapes (cylindrical, sphere, etc.) and different array shapes (rectangular, single wing, quadruple wings, etc., the relations in annex E allow for calculating more appropriate values. In case the assumption of a homogenous mass distribution is questionable, further refinements to the estimation can be obtained from for instance Engineering Mechanics by Meriam & Kraige, 1993 or http://emweb.unl.edu/negahban/em373/note18/note18.htm

3.10 Budgeting

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. An example mass budget is given in Table 20 and an example (electric) power budget is shown in Table 21.

Table 20: Example mass budget (courtesy ESA)

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**

Table 21: Example power budget (adapted from MSG spacecraft [Haines])

Element	**Nominal conditions (W)**	**Peak (W)**
Payload 1 (SEVIRI)	166	324
Payload 2	154	154
Antennas	12	12
Command and data handling	62	62
Attitude and Orbit Control System (AOCS)	20	20
Thermal control	38	50
Communications	32	32
Total Power	**484**	**654**
Total with 6% margin	**513**	**693**

Both budgets clearly show the various elements accounted for and the allocation made. The elements may vary from S/C to S/C depending on the functions the S/C need to fulfill. Also some elements accounted for in one budget may be missing from another. For instance, the structures system is accounted for in the mass budget, but not in the power budget. This is because the structures usually do not require any electrical power. Note that in the power budget as shown in Table 21 the Electric Power Subsystem (EPS) is absent, so this power budget is indicating the net power that the EPS should deliver. Depending on the design of the EPS, the total power produced (including the losses incurred in the EPS itself) should be established.

Both budgets also show the inclusion of margin, i.e. contingency. In Table 20 a margin of 20% is taken on both dry mass and propellant (here mistakenly referred to as fuel). Given that both margins are summed with the dry mas and fuel mass, it follows that the designer considered the two well correlated, i.e. if dry mass goes up than also propellant mass goes up.

Budgeting and the setting of margins generally starts with the collection of historic data on mass, power, size, cost, reliability, failure rate, etc. of the spacecraft and its many constituting parts. Once this is done, the data needs to be elaborated upon to allow for generating budgets and to determine estimation accuracy (or estimation uncertainty). 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 outlined for determining a first budget for some spacecraft design related parameters based on historic data using a top down approach, i.e. working from the highest level downwards (starting at the spacecraft level and in later phases progressing to the subsystems and their components).

Budgeting for mass

Mass budgeting (like for any parameter) starts with collecting mass data of spacecraft that are of the same category as the spacecraft to be designed. 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[17]. The table provides percentage of spacecraft subsystem (dry) mass in relation to total spacecraft dry mass.

Table 22: 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

Next we can determine average (percentage) values and sample standard deviation (SSD) on the mass ratios for each element that needs to be budgeted. These data are also included in the table; see the appropriate rows in the table. In the final row, a measure is provided for how important each component is relative to the payload.

Using the average percentage values, subsystem estimation relationships can be derived. An example of how to generate a mass budget is given hereafter.

Note that, 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. This is left for future work.

[17] 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 22****.*

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 for now is considered to hold no uncertainty.

Solution

- *As a first step, we compute bus dry mass. It follows a bus dry mass of 1440 kg – 300 kg = 1140 kg.*
- *From Table 22, we learn that the structure system mass 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.*
- *Result for the various subsystems are given in the third column of the next table:*

Subsystem	*% contribution*	*Subsystem mass*	*SSD*
Structures	*29.6%*	*337 kg*	*47.5 kg*
Thermal	*5.9%*	*67 kg*	*44.6 kg*
Power	*40.6%*	*463 kg*	*80.6 kg*
TT&C	*5.9%*	*67 kg*	*23.0 kg*
AOCS	*8.5%*	*97 kg*	*30.2 kg*
Propulsion	*7.2%*	*82 kg*	*38.9 kg*
Miscellaneous	***2.3%***	***26 kg***	
Total excluding margin	*100%*	*1140 kg*	
Margin	-	*117 kg*	-
Total	-	*1257 kg*	

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 22*** *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 correlated estimates, see appendix E, it follows a total SSD of 265 kg. Taking two SSD as the margin, i.e. margin is 530 kg, it follows a total dry vehicle mass including margin of 1670 kg. This is worse than the margin estimated in section 0. This is a bit unexpected as we are improving the estimates. In general, we need to improve our estimates so that margins do not explode.*

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 2 SSD,*
- *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 can be seen from the example mass budget in Table 20, 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 the generation of a power-, volume-, and cost budget. To give you a head start, we have collected related budgeting data for various types of spacecraft in appendix D.

Budgeting for reliability

Reliability budgeting is a bit different as compared to mass, power, volume and cost budgeting. For reliability, this is because it depends on both the failure rate and the life of the S/C. 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 [42] to compute the reliability of the various subsystems. Typical such percentages are given in Figure 36. For more details, see appendix D.

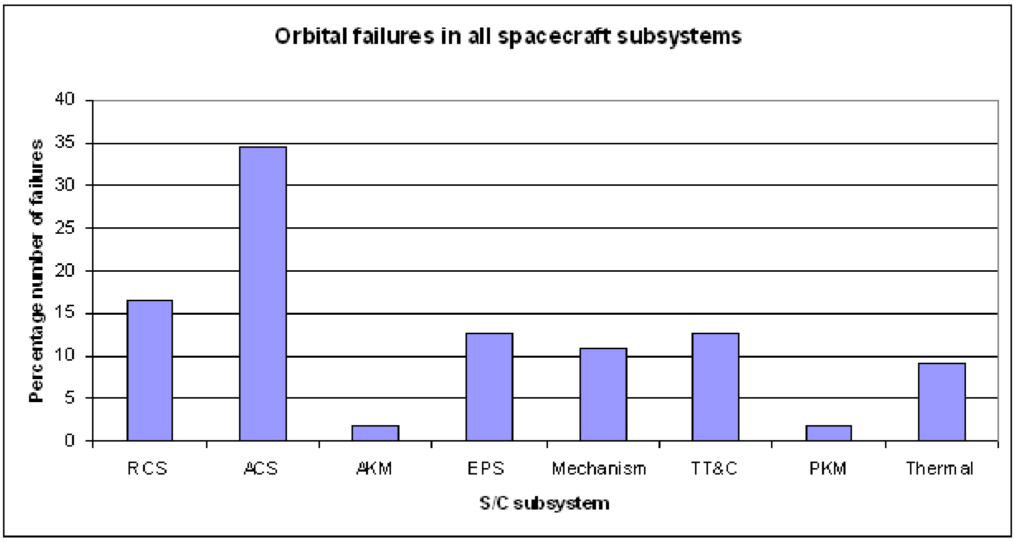


Figure 36: Spacecraft failure rate data per subsystem

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

$$R = e^{-\sum \lambda_i t} = \prod_i e^{-\lambda_i t} = \prod_i R_i \qquad [43]$$

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 per year it follows for the RCS a failure rate of 0.00169 failures per year.*
- *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. Given the low percentage failure rate, this has little consequence. However, in case the spacecraft has little or no mechanisms, the consequence is much more pronounced. This though is left for self-study.

A margin on reliability can be determined by taking a margin on the failure rate. When setting a margin on failure rate of say 30%, this means that the failure rates to be designed for should be 30% less than the failure rate that follows from the required reliability. This is included in the example reliability budget given in Table 23.

Table 23: Example reliability budget (AKM and PKM not included) with and without margin

S/C reliability	0.9			
S/C life	10 year			
Element	**Failure rate (w/o margin)**	**Reliability (w/o margin)**	**Failure rate (with 30% margin)**	**Reliability (with margin)**
	[failures/year]	[-]	[failures/year]	[-]
RCS	0.001847	0.9817	0.001293	0.9872
ACS	0.003802	0.9627	0.002661	0.9737
EPS	0.001412	0.9860	0.000988	0.9902
Mechanisms	0.001195	0.9881	0.000836	0.9917
TT&C	0.001303	0.9871	0.000912	0.9901
TCS	0.000978	0.9903	0.000684	0.9932
Total S/C		**0.9**		**0.929**

It follows that with margin, vehicle design reliability needs to increase from 0.9 to 0.0.929.

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:

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

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 manoeuvres require most propellant as a few percentage savings on a large amount of propellant may be much more easily accomplished than a large saving on a small amount. 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 appendix A.
- 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

3.11 Evaluate (and if necessary iterate) design

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

In the evaluation, it is key to determine the key S/C requirements and the key characteristics of the S/C.

- Key requirements are requirements that drive/dominate the design. They cannot be influenced by the designer.
- 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 should also focus on getting answers on below questions:

- Is launch vehicle 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.

In case 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.

Key for the success of the method described in this document is in identifying the spacecraft type and selecting 'comparable' or 'similar vehicles. For instance, it is clear that a space launch vehicle 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 a planetary orbiter, that we compare it to other planetary orbiters and so on. It is also very critical that we make sure that subsystems are unambiguously defined, i.e. it is clear what items belong to a subsystem and which not.

3.12 Problems

1. Explain in your own words the meaning of the following terms:
   a. Data credibility
   b. Parametric estimation
   c. Margin
   d. Budgeting
   e. Spacecraft configuration
   f. Key requirement (of design driver)

2. You are designing a microsatellite to carry a payload that requires 9.0 W of electrical power. To allow for a very first power estimate of the spacecraft, you have collected data on payload power versus total power for a number of similar spacecraft. These data are summarized in table on the side.

Spacecraft	Payload power [W]	Total power [W]
Delfi C3	0.29	2.6015
XI-IV	0.12	0.619
HauSat-I	0.5	1.06
Orsted	11.1	31
PoSat	7.9	18.2
Saga	10	70

   Using the data collected, you are to determine the total spacecraft power to be generated on board to allow for successful operation of the spacecraft using:
   a) Analogous estimation
   b) Average payload to total power ratio
   c) Describe in your own words which of the two estimates you consider better and why.

Answer: a) 9.0 W payload power specified is close to the payload power of PoSat. So it follows that payload power of the microsatellite is close or somewhat in excess of 18.2 W. However, it is also close to that of Saga, which would lead to a total power of 70 W or slightly less. This then would give an estimated power range of 18.2-70W; b) Average payload power to total power ratio is 0.285 (28.5 %) with an SSD of 0.156. Based on the 9.0 W of power we find for the spacecraft total power a value of 31.6 W (MLE). With a probability of 65%, the spacecraft power is in the range 21.4 W to 60.4 W.

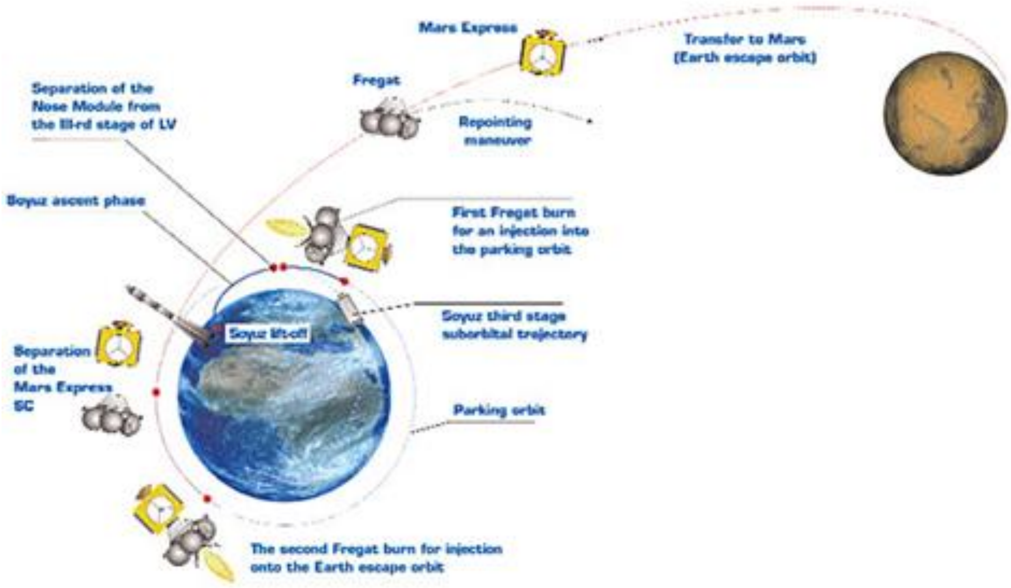
3. You are designing a spacecraft capable of carrying a payload mass of 150 kg.
   a. Select arbitrarily 6 S/C from the table on the right and determine for this selection the average payload mass to on station mass ratio and the sample standard deviation (SSD).
   b. Use the data obtained to estimate the on-station mass of your spacecraft

   Answer: Varies depending on the 6 S/C selected from the table.

S/C	Orbit	payload mass	on station mass	In GTO orbit mass
	(-)	(kg)	(kg)	(kg)
Meteor M	LEO	1200	2700	NA
Bird	LEO	24	72	NA
Resurs DK	LEO	1200	6650	NA
Electro L	LEO	370	1500	NA
Meteosat	GEO	63	322	720
MSG	GEO	270	1000	2010
GOES-6	GEO	63	326,3	660
Earlybird	LEO	150	310	NA
Quickbird-1	LEO	290	910	NA
EROS-A	LEO	36	250	NA
DMSP 5D-2	LEO	243,2	833,6	NA
DMSP 5D-3	LEO	308,2	1045,5	NA
Envisat	LEO	2150	8140	NA
SMEX-SWAS	LEO	102	287	NA
SMEX-TRACE	LEO	58	213	NA

4. Consider a deep space probe of <u>dry</u> mass 600 kg using two means of rocket propulsion, each with its own specific impulse) to perform:
   - Transfer flight: $\Delta v = 5600$ m/s ($I_{sp} = 340$ s)
   - On-station orbit maintenance: $\Delta v = 900$ m/s ($I_{sp} = 300$ s)

   The propellant for the transfer flight is stored on board of a kick stage (a bit similar to the European Mars Express deep space probe, see figure (figure with courtesy of ESA).



   Given that the dry mass of the kick stage equals 10% of the propellant mass carried on board of the kick stage, determine the total vehicle (kick stage + probe) loaded mass in case no margin is taken into account. Consider whether the order of calculation (first the transfer flight and then orbit maintenance or vice versa) has an effect on the answer obtained.

   Answer: 7.74 ton (1 ton = 1000 kg)

5. Consider that you are in the process of designing a spacecraft to carry a payload mass to its final destination. Essentially two different methods have been discussed in the foregoing text tot to determine spacecraft total mass. The first method uses data on total mass and payload mass of historic spacecraft to determine total S/C mass, whereas the second method estimates loaded mass based on the sum of S/C dry mass and propellant mass using available data on dry mass and payload mass of historic spacecraft, mission Δv and specific impulse. Explain in your own words which method you consider more accurate and why?

6. You are to design a spacecraft with a reliability of 0.8 over a mission duration of 4 year. The payload demonstrated reliability is 0.7 over a period of 10 year. Given a constant (but different) failure rate for payload and spacecraft, determine for this spacecraft the required bus reliability.

   Answer: 0.923

7. A deep space probe costs 100 M$. Design life is 10 yrs. In case we estimate an average catastrophic failure rate of 0.02 S/C failures per year, what is the risk associated with this S/C?
   Risk is … (select best value)

   a) 82 M$ b) 80 M$ c) 20 M$ d) 18 M$ e) 25 M$

   Answer: 18 M$

8. You are considering the best vehicle to launch a S/C costing 100 M$. Total mass of this S/C is estimated at 3000 kg with an RSE of 10%. You have collected data of 5 different launch vehicle options, see table below, all capable of reaching the target orbit.

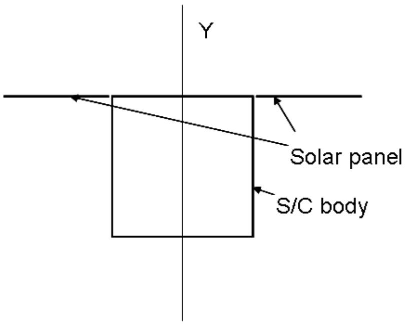
Launch Vehicle	A	B	C	D	E
Launch cost [Million dollar]	45	40	40	38	38
Launch vehicle reliability [-]	0.965	0.9	0.95	0.85	0.85
Payload mass [kg]	4000	3600	3300	3600	3300

   Additionally is given that:
   - in case of a launch failure a new launch (not the S/C) is provided for free, and
   - the probability of needing to select a new launch vehicle because of a too heavy S/C (consider the given RSE) should be equal or less than 2.5% (use "68-95-99.7" rule).

   Which launch vehicle provides lowest total cost when taking into account the associated risk due to a launch failure?

   Answer: Launch Vehicle A.

9. You are designing a space probe targeted for Venus. Using data on similar probes you have estimated a total vehicle mass of 300 kg and a maximum power need from the solar panels of 600 W at 1 AU. The current vehicle configuration is given in figure on right.



   For this vehicle are given:
   - Total solar array power at 1 AU: 600 W (about 1200 W at Venus)
   - Total vehicle mass (including solar array): 300 kg
   - Vehicle mass density as computed from comparable vehicles: 200 kg/m^3
   - Vehicle has a cubic shaped body with 2 solar panels directly attached (one on each side) to the S/C body

- Solar panel side attached to the body is equal to body linear dimension
- Solar incidence angle on solar panel is 90 deg.
- Power density and specific mass @ 1 AU is 150 W/m^2 and 50 W/kg, respectively
- Mass moment of inertia of cube about centre of mass is 1/6 $M*L^2$ with 'M' is body mass (including solar array) and 'L' is length of sides of cube.
- Mass moment of inertia about centre of mass of thin rectangular panel = 1/12 * M * ($l^2 + w^2$) with M is panel mass and 'l' and 'w' are panel length and width, respectively.

Estimate for this vehicle:

a. Mass moment of inertia of S/C before deploying the panels (panels are usually thin and hence are considered of negligible thickness)
b. Mass moment of inertia of S/C about y- axis after deploying the solar panels, see figure.

Hint 1: For answering question a first determine the linear dimension of the S/C.
Hint 2: For answering question b, first determine solar array panel mass, length and width.

Answers: a) 65.6 kg-m ^{2}(L = 1.145 m); b) 29.6 kg-m^2(Venus is at an average distance of 0.7 AU from the Sun; When in Venus orbit, the solar array should deliver about 1200 W, when in Earth orbit, this is 600 W; Array mass is 600 W / 50 W/kg = 12 kg or 6 kg per panel; Array area is 600 W/ 150 W/m^2 = 4 m^2 or 2 m^2 per panel; Assuming that width of panel is equal to linear dimension, we find w = 1.145 m and hence l = 2 m^2 / 1.145 m = 1.75 m; MMOI of single panel is ~2.19 kg/m^2; Taking into account two panels and using parallel axis theorem, we find MMOI of panels about y-axis = 2 (2.19 kgm^2 + (6 kg(0.5*(1.75 m+1.145 m))^2 = 29.62 kgm^2)*

Consider that now also the MMOI of the S/C about the y-axis can be determined in the deployed state by simply adding the MMOI of S/C body and of the panels. A quick estimate can be obtained by neglecting the effect of panel deployment on the position of the S/C center of mass and neglecting the effect of panel deployment on S/C body MMOI. This though is left for another time.

Further problems for exercising upon are available via the course Brightspace pages (look for Mobius exercises) or are provided in the tutorials.

4 System level sizing

The spacecraft (bus) is traditionally divided into 8 subsystems as given in Table 24 and Figure 37[18] (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 24: 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	Up link and downlink & Spacecraft tracking	Tracking, Telemetry & Command (TT&C)[19]
Command & Data Handling (C&DH)	Command processing Data processing/formatting	
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

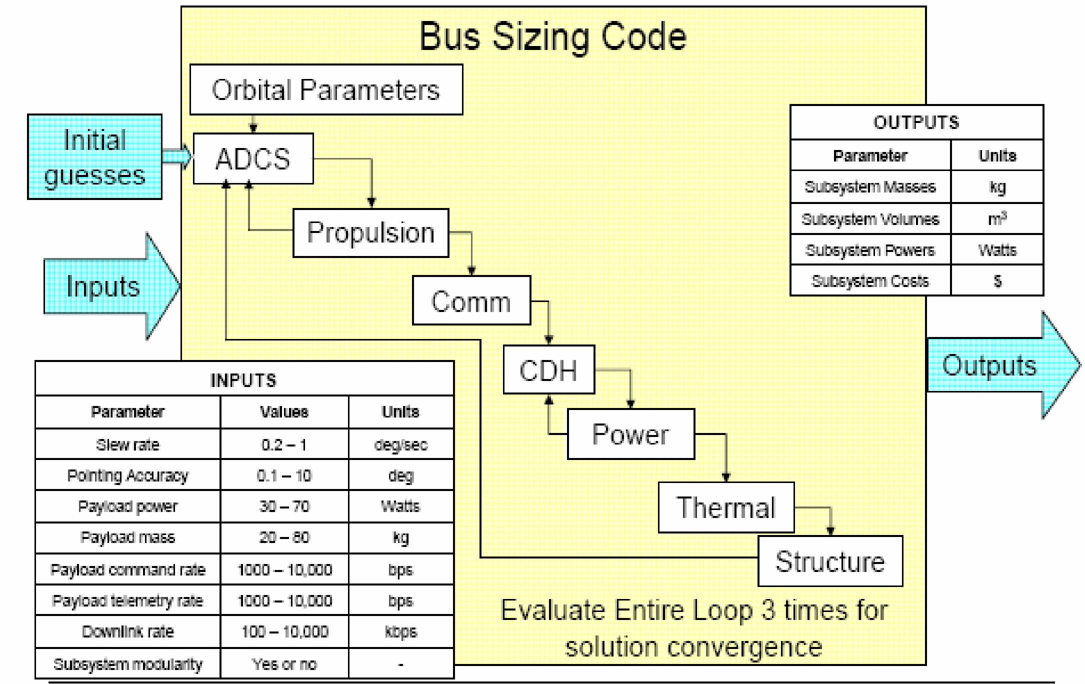


Figure 37: The spacecraft design process

[18] Figure 37 shows the classical waterfall approach to design where each of the subsystems are designed in a sequential manor. Another, more modern approach, uses concurrent engineering where all of the subsystems are designed at the same time (in parallel). This is more difficult to manage, but can significantly reduce the required design time.

[19] In some spacecraft the command and data handling (C&DH) system and the communications system sometimes are combined into a single system: the tracking, telemetry and command (TT&C) system.

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)
- How the system looks (what elements make up the system), what major design options exist and possible configuration issues
- Key design features
- Key (design) issues
- Fundamental design relationships
- Major steps in structures (and mechanisms) design

No attention in this work is given to design verification. This is considered out of scope for the present work.

Problems

1. Give an account of the eight traditional spacecraft bus subsystems, identifying the main functions provided by each.

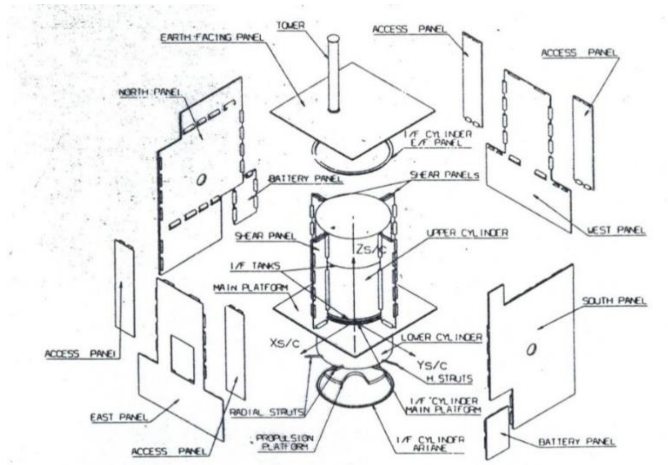
2. Describe in your own words the meaning of the terms:
    a. Waterfall method
    b. Concurrent engineering

4.1 Structures and Mechanisms

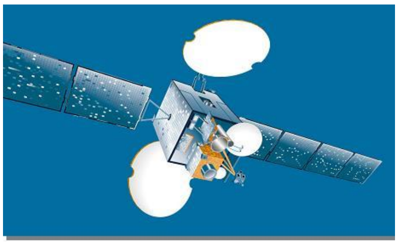
In this section, we will first describe various examples of spacecraft structures and mechanisms used. Next we will discuss important design features of structures and mechanisms followed by why we need them and a discussion of important design issues. Next various simple methods are presented that allow illustrating the basic approach towards the design of spacecraft structures. Some of the methods presented relate to the material offered in the course Introduction to Aerospace Engineering II and hence a good understanding of this material is needed. Finally, we will discuss material selection and provide an overview of the most important design loads and on how Factors of Safety are used to ensure a safe design.

Spacecraft structures (examples)

Figure 38 shows the main structural elements of the Artemis S/C. This is a large GEO data relay telecommunications spacecraft of mass 3100 kg designed to withstand 8.25g in longitudinal and 2.70g in lateral direction. 1st longitudinal frequency is 42 Hz and first lateral frequency is 15 Hz. Its height (bottom to top) is 4.8 m. Body dimensions are 3.1 x 1.86 x 1.9 m^3.



Artemis S/C structural elements [Gaudenzi]



Artemis spacecraft (courtesy ESA)

Figure 38: Artemis spacecraft structural elements

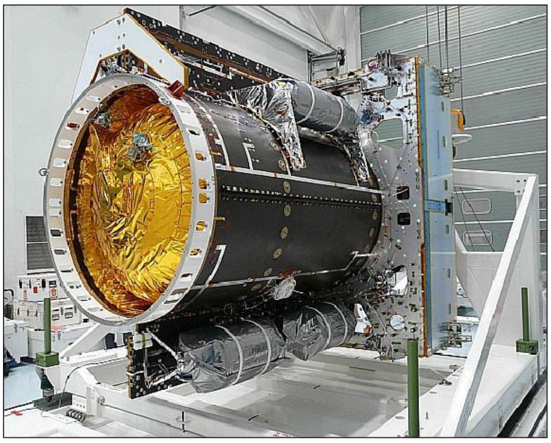
Key features of the design are [Gaudenzi] the high stiff box shape consisting of a central cylinder of diameter ~1.19 m, a main platform, a propulsion platform and various panels. Lower part of the cylinder provides for a 1194 mm interface with the launcher (Ariane 5). The North-South panels are carrying the radiators and the batteries (which produce a lot of heat) and provide for attachment of the solar arrays. Embedded in the central cylinder are two large propellant tanks. Tank loads are transmitted directly to the central cylinder. All panels are of a sandwich design with aluminum skins. Total structures mass is 185 kg of which the central cylinder and the bottom and top panel make up 80.5 kg (43.5%) and the wall panels 55.10 kg (29.7%), see table on side.

MASS BREAKDOWN

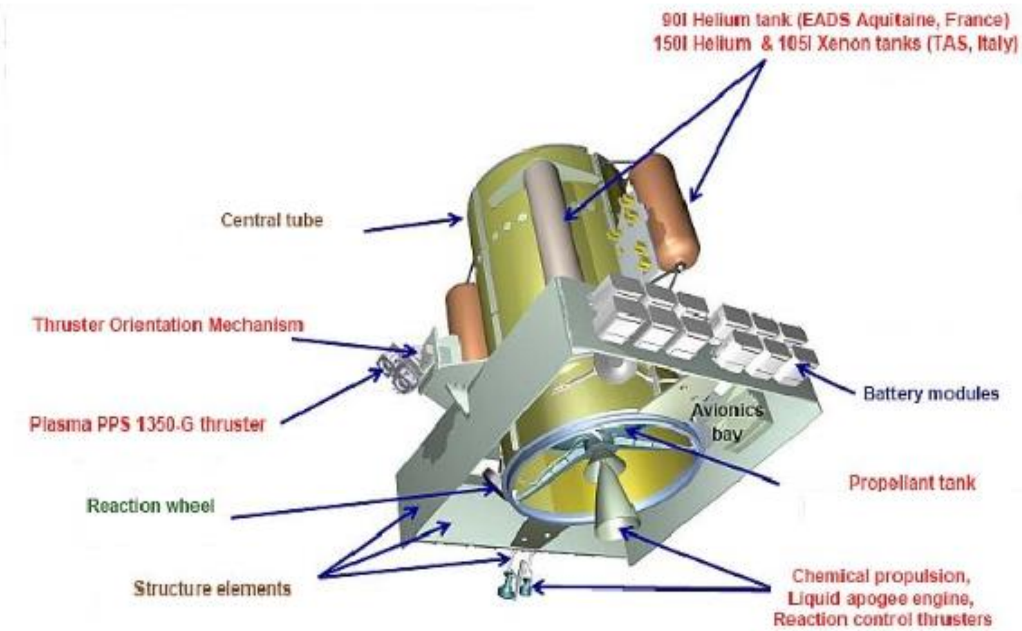
EQUIPMENT DENOMINATION	Q.TY	TOTAL MASS (Kg)
Access Panel	4	6.80
Balancing Massess Support	8	0.50
Battery Panel	2	4.00
-Z chem. Thrust. Support	(4)	—
Central Cylinder	1	58.50
Earth Facing Panel	1	11.70
East Panel (+X)	1	8.70
Horizontal Strut	4	0.80
Lower Cylinder	(1)	—
Main Platform	1	10.30
North Panel (-Y)	1	17.80
Propulsion Platform	1	4.70
Shear Web	2	3.20
STR Miscellaneous (SET)	1	23.50
Sensor Support	7	0.50
South Panel (+Y)	1	17.40
SADM Strut	2	0.20
Thruster Support	16	2.40
TT&C Support	1	1.00
Tower	(1)	—
Upper Cylinder	(1)	—
Vertical Strut	4	1.80
West Panel (-X)	1	11.20

Next figure shows the Alphabus spacecraft service module (SM) next to the Alphasat repeater module (RM) with antenna module (AM) on top. The SM is built around a large central tube (1.6 m diameter) embedding two large propellant tanks with a maximum capacity of 4200 kg, and provides the mechanical interface (1666 mm) with the launch vehicle for a launch mass of up to 8,800 kg including a payload mass of up to 1500 kg. Attached to the central tube is an inner deck/platform of outer dimensions 2800 x 2490 mm^2 as well as some panels that allow for mounting of the batteries and various other devices. The central tube also supports the repeater module and carries within the propellant tanks needed for the propulsion system. The central tube is a carbon fiber composite structure offering high strength and low mass. Alphasat RM has three floors and consists of two halves to allow for ease of assembly of the payload. Floor size is 2.5 x 2.8 m^2.

Alphabus payload/repeater module (l) and bus/service module



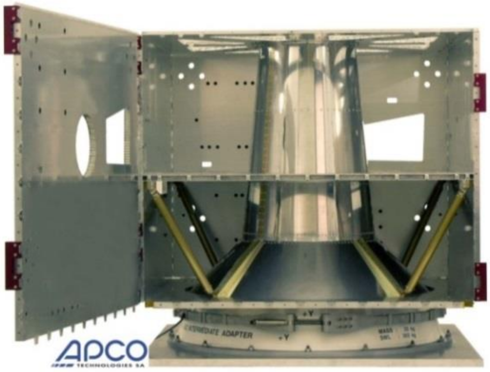
Alphabus service module



Schematic of service module

Figure 39: Alphabus spacecraft payload module and service (bus) module (courtesy ESA)

Figure 40 shows the main structural elements of the SMART-1 structure. SMART-1 is a 370 kg spacecraft intended as a Spacecraft & payload technology demonstration for future European cornerstone missions. The S/C carries a 19 kg payload. Its structure is built around an aluminium sandwich central cone, which supports 3 decks and four sidewall panels on which equipment are mounted. Only aluminium alloys are used for face-sheets, honeycomb cores, struts and secondary structures. The mass of the structure is 45 kg. Figure (left) shows primary structure of SMART-1 with the central cone mounted on ground-handling adapter. Special panel hinge and attachment allows easy access inside the structure. Central cone, see figure bottom right, holds plasma thruster. Also 82 kg of supercritical xenon propellant will be carried aboard SMART-1 in a tank mounted in the centre of the structure above the thruster.



Aluminium box structure (volume of ~1 m^3) with cut-outs for solar array drive and star trackers (front and aft panel) and 3 decks.

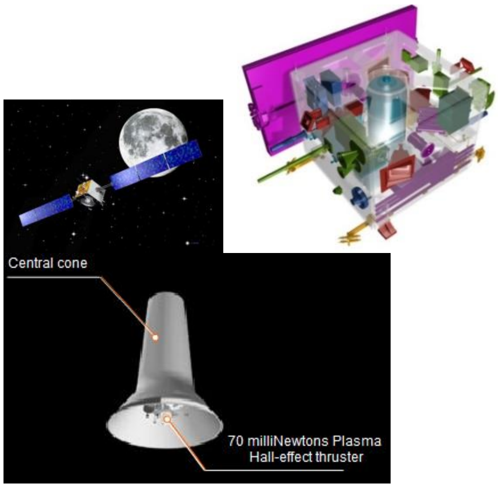


Figure 40: SMART structure (courtesy ESA)

Figure 41 gives a schematic view of the European ATV and the ATV S/C structure subsystem. Its mission is to periodically bring supplies (up to 7500 kg) to the International Space Station and to dispose waste (6500 kg). Overall launch mass is about 20.7 ton. The vehicle has a diameter of 4.48 m and an overall length of 9.79 m (10.77 meters including the Separation and Distancing Module). Four solar panel wings are deployed in orbit, with a span of 22.28 meters.

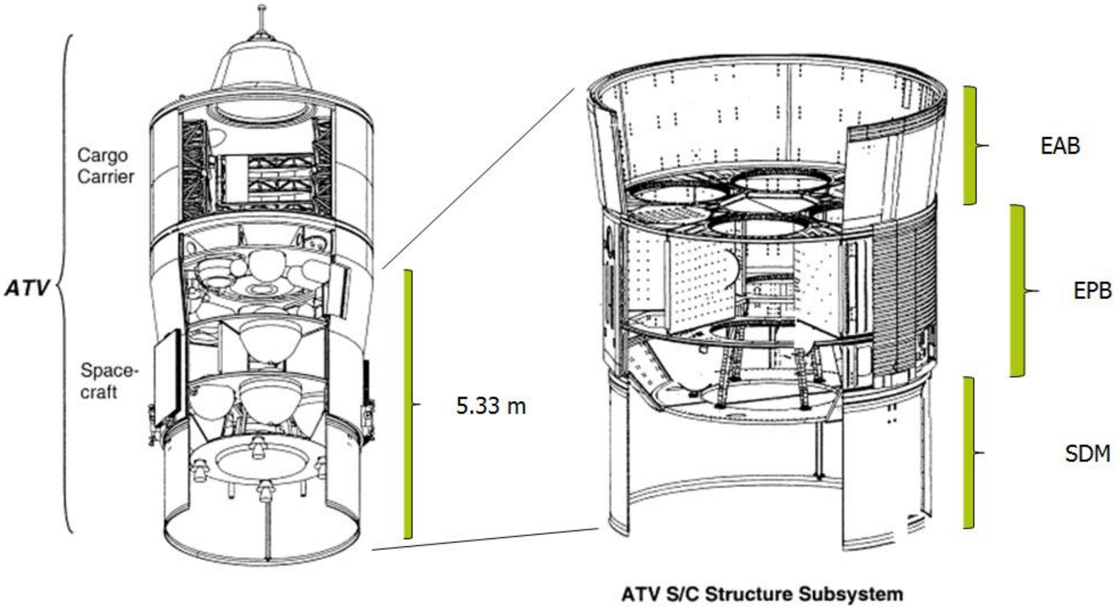


Figure 41: Automated Transfer Vehicle structure subsystem [Kogl]

The total launch mass of 20.7 ton includes a vehicle dry mass of 9,784 kg, 5,858 kg of propellant for in-orbit maneuvers (including raising the orbit of ISS), and 4,597 kg cargo in support of the ISS. The design drivers for the structure include the ability to withstand 5.45 g axial loading and 1.5 g lateral loading. The resulting design as shown in figure comprises:

- Separation and Distancing Module (SDM) with a height of 2.0 m and a mass of 345 kg, which provides the mechanical interface with Ariane-5 and ATV’s separation and distancing from the launcher;

• Propulsion Bay (EPB) with a height of 2.033 m, which accommodates most of the Propulsion and Re-boost Subsystem (some thrusters are also located on the forward part of the cargo carrier);
• Avionics Bay (EAB) with a height of 1.33 m, which accommodates most of the avionics equipment;
• Solar Generation System (SGS), which includes four 4-panel deployable solar wings, each with its own drive mechanism.

As can be seen from the figure, the design essentially is of a cylindrical planform with shelves/plates added to allow for mounting of tanks, equipment and instruments. Some plates are supported to limit bending. Structure mass of spacecraft (not including the cargo carrier) is 2.15 t.

Figure 42 shows the structure of the MSG (Meteosat Second Generation) satellite. It consists of two main parts, a 192-kg primary structure that supports the payload and the various subsystems and electronics boxes, and the secondary support structure weighing 27.5 kg and supporting the propulsion system and the electrical power system. Total spacecraft mass is approximately two tons, which indicates a structural mass of about 10% of the total S/C mass. The primary structure is comprised of the service module structure hosting the payloads and majority of the subsystems, and an antenna platform hosting the communications antenna. It consists of a conical central cylinder (the backbone) to which a number of circular-shaped platforms/decks are attached supported by struts, and a stringer stiffened shell. The secondary structure consists of a number of struts. Still other elements include lower closing support, sun shade, fluid tanks (pressure vessels), etc.

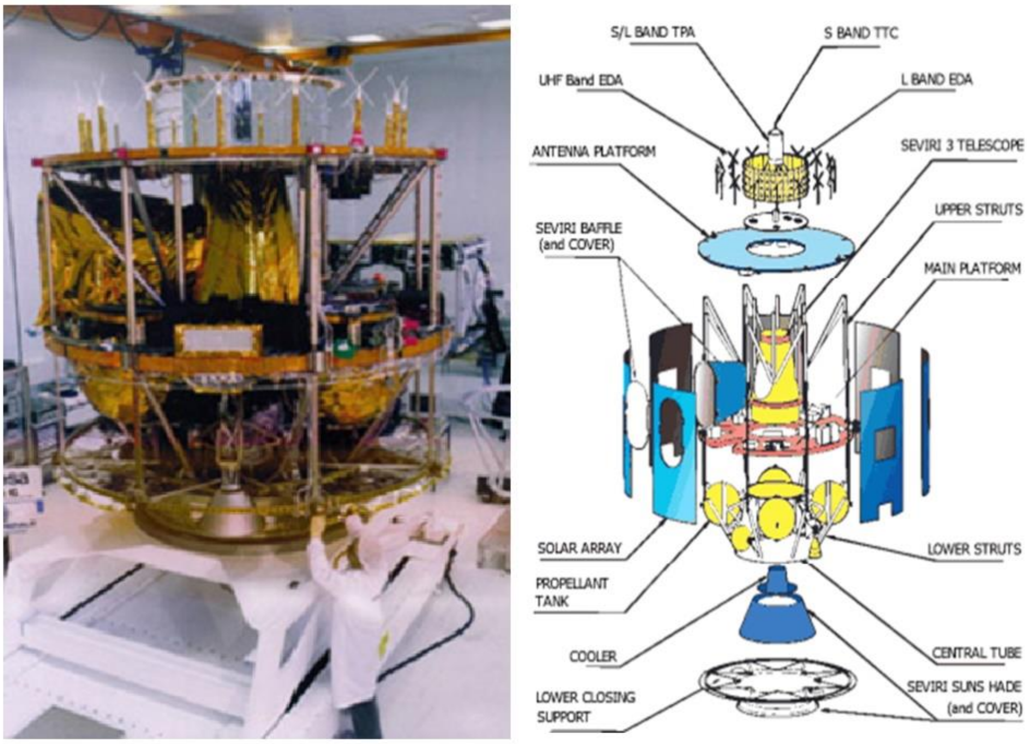
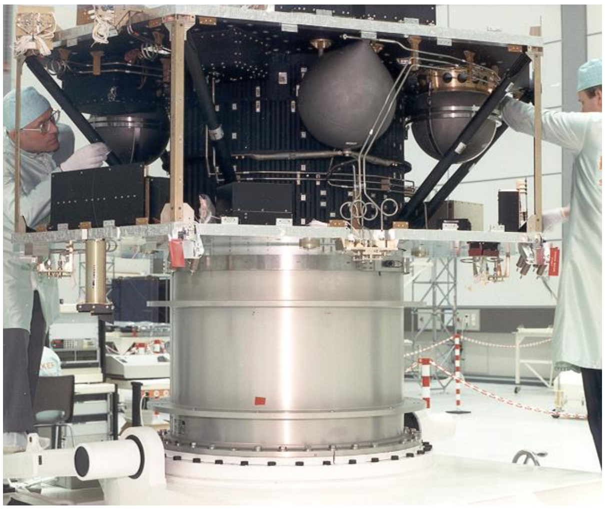


Figure 42: Meteosat Second Generation (MSG) Structure (courtesy ESA)

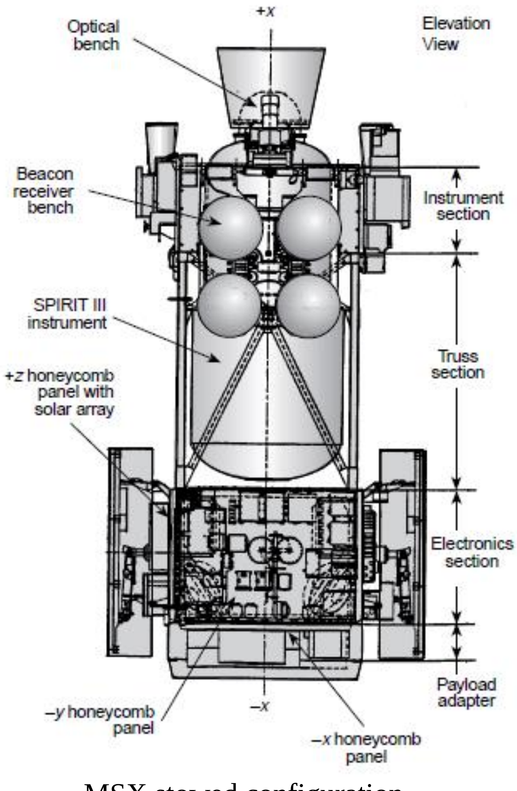
Figure 43 shows on left a central cylinder for a large space platform (Spacebus-4000) being placed on top of a payload adapter or LVA. The CFRP Central Cylinder has a diameter of 1.2 meters, a height of 4 meters and weighs less than 90 kg. Total mass supported is up to 6 tons. Figure on right shows the primary structure of the Hipparcos satellite (1140 kg launch mass) consisting of a central cylinder to which a number of horizontal platforms/decks are attached supported at the outer edge by struts and with LVA attachment ring on bottom. In the central cylinder an apogee kick engine of mass 463 kg is installed. Not shown in figure are the payload interface structure and the Sun-shade structure.

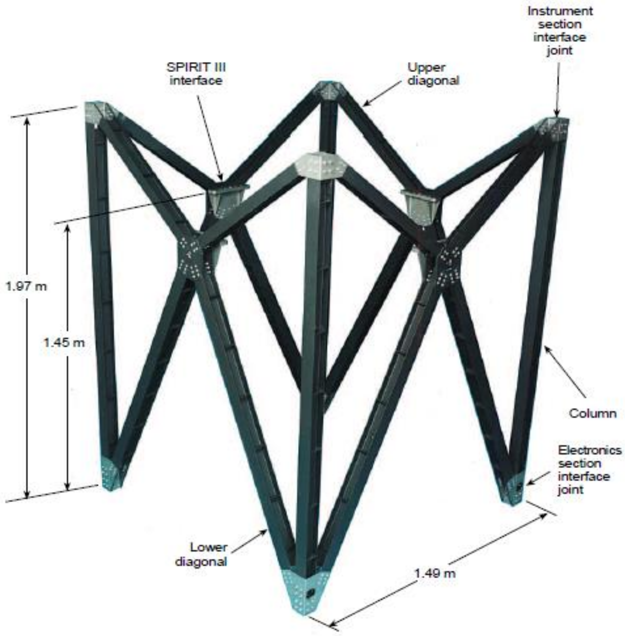


Alphabus 4000 central tube Hipparcos S/C structure

Figure 43: Spacecraft example structures

Next figure shows a schematic of the structure of the Midcourse Space Experiment (MSX) spacecraft. This observations spacecraft provides a multisensory platform for target detection and tracking. The spacecraft has an total mass of 2812 kg and has a length of 5.1 m and a diameter of 3.3 m.





MSX stowed configuration MSX truss structure

Figure 44: MSX structural build up [Skulney]

From the work of [Skulney] we learn that the vehicle consists of an instrument section, a centre (or truss) section and a bottom (or electronics) section. The instrument section has a box-style structure of size 1.59 x 1.59 x 0.72 m^3. It is made up of honeycomb panels (3.3 cm thick core and 0.5 mm Al face sheets) and "L" shaped corner elements. The honey comb panels contain embedded heat pipes for thermal control. The centre section of the spacecraft includes a graphite/epoxy composite truss structure that support the main observation instrument. It has a mass of less than 54.4 kg and was to offer a spacecraft frequency of 12 Hz lateral and 35 Hz axial. The structure also offers a low coefficient of thermal expansion to ensure proper alignment of the main instrument and a low thermal conductivity to minimize the heat transfer between the electronic and the instrument section. The bottom section essentially makes up the bus section. It essentially has a hollow (skeletal) rectangular structure of size 1.59 m^2 and 1.12 m long covered on all sides by 5.2 cm thick aluminium honeycomb panels with 0.6 mm Al face sheets. Part of the structure is also a 1.60 m diameter and 30 cm long cylindrical payload adapter for mating the spacecraft to the launch vehicle.

In Figure 45 the structure of a nanosatellite (mass below about 10 kg) is shown. It is the structure of The Delft developed Delfi-C³ satellite with a total mass of 2.1 kg. From http://www.delfispace.nl/delfi-c3/structure-mechanisms-thermal-control, we learn that its structure is based on the 3-unit (1 unit is 100 x 100 x 100 mm^3) solid chassis from Pumpkin Inc. A 3-unit CubeSat has a total structure length of 326.5 mm and a cross-section of 100x100 mm. The total length of the satellite is 340.5 mm, the extra length is from support feet at both ends (top and bottom), which thereby also allow room for the separation spring that ensures separation from the launch vehicle.

Figure 45: Delfi C³ structure

The primary (outer) structure consists out of the tube chassis and the top and bottom panels. Together these components absorb the mechanical loads during launch and provide the electrical ground. The primary structure components are made from Aluminium 5052-H32. The secondary (inner) structure supports the internal equipment boards. It consists out of four threaded rods that run the full length of the spacecraft. The rods are connected to the tube chassis and the top and bottom panels of the primary structure to increase the rigidity. The solar panel structure consists of two sections. The main section supports the GaAs solar cells, whereas a second section located at the end of the panel suspends a small solar cell payload. The primary structure has a mass of approximately 735 gram (~35% of total mass S/C mass) and the secondary structure 100 gram (~5 % of total S/C mass).

Mechanisms (examples)
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 each with their own application is given in Table 25.

Table 25: Typical space mechanisms

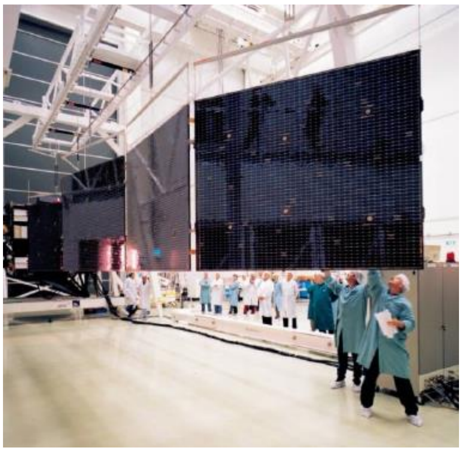
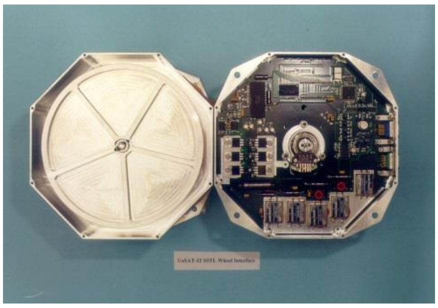
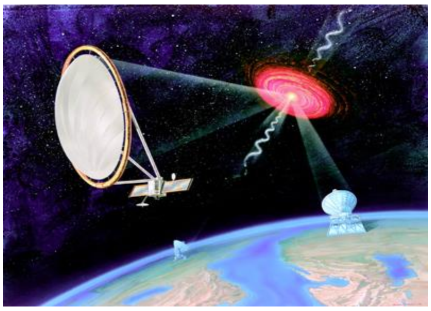
	**Solar panel deployment mechanism** To deploy the initially stowed (during launch) solar panel. After deployment, panels need to lock in place thereby increasing stiffness.
	**Solar array drive mechanism** To align the spacecraft's solar panels towards the Sun. It essentially is a large electric motor bolted onto the primary structure that connects to the arm of the solar array. It also ensure that the power from the array is transferred to the spacecraft via e.g. slip rings. Important for the design/or selection of the system is the rotational rate it should be able to sustain as well as the loads (including bending) it should be able to carry.
	**Separation mechanism** To separate the payload from the LV or a mother spacecraft. Mechanism shown was used for separation of a.o. YesSAT (a student designed spacecraft) thereby ensuring a 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.

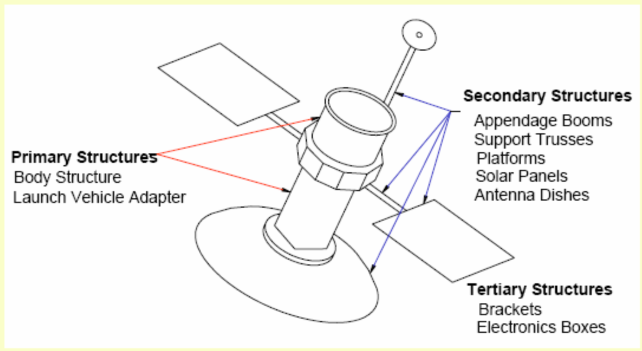
Table 25: continued

	**Pointing mechanism** For pointing of antennae, antenna dishes, thrusters, etc. Important is the number of axes (one- or two) about which a rotation can be accomplished. Shown in figure is an antenna pointing mechanism developed by RUAG which includes two electrical motors for changing the angular position about two axis. It has a mass of 10.5 kg and requires 6.5 W of power per motor (when active). Pointing precision is 0.005 deg and rotational speed is 0.3 deg/s per axis.
	**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.
	**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.

Key design features of spacecraft structures and mechanisms

When considering design features of existing spacecraft structures, we learn that the spacecraft structure can be divided in:

- ***Primary*** structure or main structure, whose purpose is to guarantee the overall structural integrity and transmit loads to the base of the satellite through specifically, designed components (central tube, honeycomb platform/deck, bar truss, etc.). This structure also provides the attachment points for the payload and the associated equipment and interfaces with the launch vehicle through a launch vehicle adaptor (usually cylindrical or conical shaped). Failure of the primary structure leads to a complete collapse of the satellite.



Ref: Thomas P. Sarafin, "Spacecraft Structures & Mechanisms"

- ***Secondary*** structure includes all essential appendages and support structures, such as solar arrays, antennas, thermal shield, pressure vessels (propellant/fuel/oxidizer and pressurant tank) and side panels. A secondary structure failure is not a problem for the structural integrity, but it could have some important impacts on the mission if it causes propellant or pressurant leakage, alters the thermal control, the electrical continuity, and the mechanisms or if it crosses an optical path. 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.
- ***Tertiary*** structure refers to the smallest structural elements and includes less-essential mounting hardware, like brackets, component housings, and electronic boxes.

In more detail, we find that the main structural elements essentially are cylinders, panels (rectangular or cylindrical shaped), and beams. This is an essential feature as it allows for employing relatively simple analytical methods to perform a first design of a spacecraft structure, see later in this chapter. When considering main structural elements in some more detail, we find three different types of structures being used. These structures include:

- Monocoque structures; Monocoque construction uses stressed skin to support almost all loads much like an aluminum beverage can. The wall may consists of a thin metallic sheet, but also sandwich panels. The latter allow to increase stiffness of the structure.
- Semi-moncoque; Semimonocoque construction uses a substructure to which a skin is attached. The substructure, which consists of frames of various sizes and stringers, reinforces the stressed skin by taking some of the bending stress.
- Truss structures; This is essentially a structure without a load carrying skin. In truss structures all loads are carried by tubes which are connected at their ends to form a well-braced framework.

A broad range of mechanisms can be distinguished based on their application. When looking more at the principles of how they operate, we can see various basic ways of operation:

- 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

Still, the main elements of mechanisms are:

- Actuator (Spring/stored energy devices, motor)
- Flexible joints (slip rings, hinges, bearings, etc.)
- Sensor for angular information or other
- Release devices
- Locking devices that stabilize the shape after deployment

An important feature of the structures and mechanisms subsystem is its mass. Typical mass data are provided in appendix D for different types of spacecraft. In appendix D, also a relation is given that provides the structural mass of the spacecraft in relation to its dry mass. It shows that for spacecraft in the dry mass range 50-1750 kg, the mass of the structures (and mechanisms) system is roughly about 20 25% of the dry mass with a relative standard error of 21.7% of the estimated value due to spread in the data. This spread is in part related to different design solutions and the use of different structural materials, see later. Also important is to consider that some secondary structure elements, like the reaction wheels, the solar panel structures and the propellant tanks, as well as the tertiary elements sometimes are included as part of another subsystem.

A commonly used material in spacecraft structures and mechanisms is aluminium. More recently though we see a trend towards the use of composite materials as it allows for reducing the mass of the structure. For some special structures, like pressure vessels (for instance for propellant storage) and truss corner sections, also titanium and steel are used. For mechanisms also some soft materials may be used to ensure closing of valves, as well as materials that provide low friction.

Other important features of spacecraft structures include there reliability and cost. Typical data on reliability and cost of S/C structures can be found in appendix D. In general, mechanisms are avoided as much as possible as they are prone to failure and are high cost items. However, this is not always possible. In case mechanisms are needed usually tests are used to determine whether the mechanism is able to function properly for the duration of the mission. This is what makes mechanisms high cost.

Why structures and mechanisms?

Need for structures:

From [Fortescue] we learn that the main function of the structures system is 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 S/C equipment, like batteries, thrusters, transmitters, solar panels, antennas, electronic boxes, radiators, etc.
- Provide interconnect to launch vehicle and or other vehicles, like orbital transfer vehicles (a kind of space tug)
- Provide handling hard points (hoisting, transportation)
- Provide protection (débris, radiation, etc.)

Need for mechanisms:

- For deployment of solar arrays, radiators, antennas and booms. Some spacecraft are thus large or hold thus large extensions that they do not fit in the launch vehicle payload bay unless specific measures (mechanisms) that allow for folding and stowing of these structures are employed.
- To allow for pointing of e.g. receive and transmit antennae, solar panels, thrusters, etc.
- To allow for a scanning motion

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

- Ability to withstand loads
  - Natural frequency sufficiently high to avoid resonance between launch vehicle and spacecraft (stiffness)
  - Strength both for tension and compression
  - Select materials that have low thermal expansion or include measures to reduce effects relating to differences in thermal expansion

  This is important as we do not want the structure to rupture, fracture, buckle, etc.
- Open structure to allow for good accessibility
- Structural mass: Reducing the structural mass generally allows for an increase in payload mass.
- Materials used shall be 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.
- Reliability and cost of the structural elements. Reliability must be high, whereas cost should be low. Note that a reduction in structural mass by replacing aluminum by a composite materials may lead to higher cost and longer development time.
- Interfacing with the launch vehicle (LV) structure. Details on interfacing with the LV can be obtained from for instance the LV catalogue and or LV manuals. The latter usually apply to a single type of LV. For some spacecraft the structure may be designed such that they can be launched with more than one launch vehicle. This is to allow for flexibility in case a certain launch vehicle is not available due to e.g. a recent launch failure.

Key design issues for mechanisms are:

- Friction: No lubricants allowed. Need to have sufficient torque to overcome friction or use of low friction materials.
- Energy must be stored (spring) or added to keep it working
- Mass, cost and reliability.

Structural analysis
Analysing a spacecraft structure can be quite complex requiring the use of finite element methods, see [Fortescue]. 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. To limit the scope of the present work, we will mainly focus on the primary structure (i.e. the load carrying structure) as well as panel like structures (solar arrays). Still for other elements the same approach may be used. Also in this work, we will only include a few simple load cases (mostly only point loads). For other more complex load cases you are referred to the course material taught in e.g. AE1110-II, but any good text book discussing the effect of loads on simple structures may do.

<u>Design for stiffness</u>
To avoid resonance, the spacecraft should be sufficiently stiff. For instance, when considering the case as depicted in Figure 46, everybody understands that the larger the base, the stiffer the structure will be. This stiffness is especially important during the launch, but also in space stiffness is an issue especially for large structures like the International Space Station.

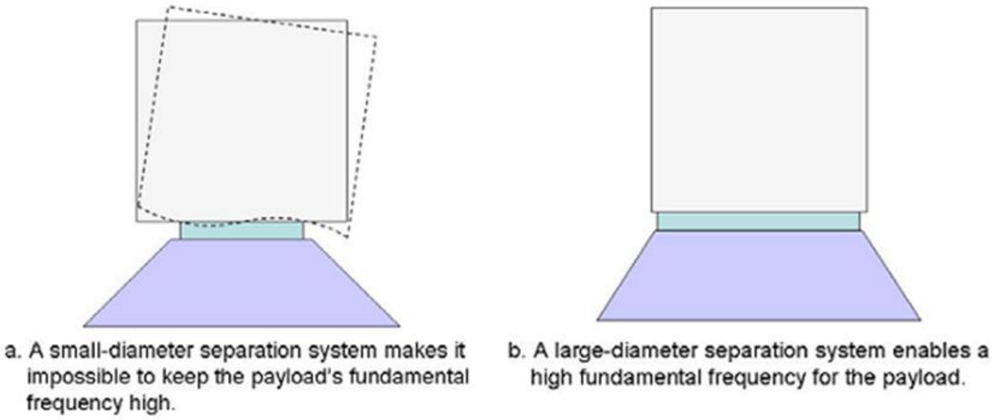


Figure 46: Effect of diameter of spacecraft separation system on stiffness

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 Figure 47 (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 of height L with mass of S/C concentrated in mass in center of geometry (beam length is 0.5L)
- S/C of height L with mass M concentrated in point mass on top (this situation is not really realistic, but the results are considered to allow for a conservative design, meaning a safe and maybe too heavy, design)
- Solar panel with panel mass M located at the panel geometrical center
- Etc.

Key for all cases is that one end is fixed and the other end is free to move.

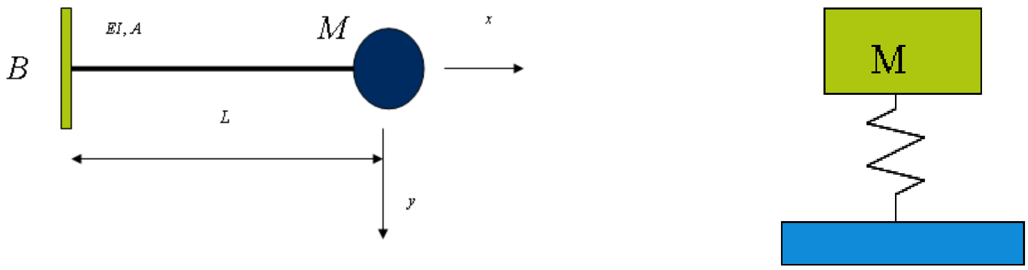


Figure 47: Beam approximation

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).

$$
\begin{aligned}
f_n &= \frac{1}{2\pi}\sqrt{\frac{k}{M}} \quad [\mathrm{Hz}] \\
f_n &= \sqrt{\frac{k}{M}} \quad [\mathrm{rad/s}]
\end{aligned} \qquad [44]
$$

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

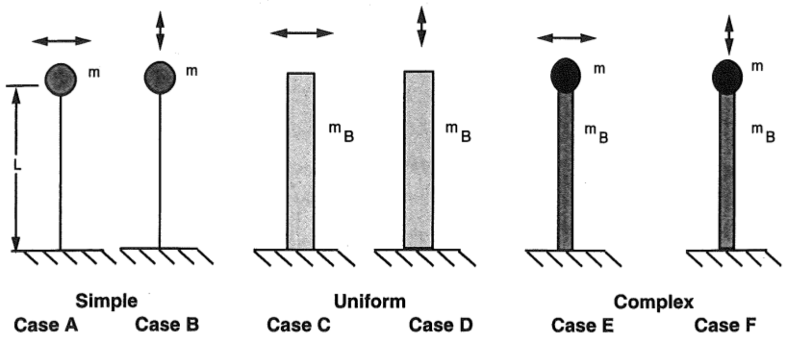
- lateral direction:

$$k_y = \frac{3EI}{L^3} \qquad [45]$$

- longitudinal direction:

$$k_x = \frac{EA}{L} \qquad [46]$$

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

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 48: Beam deflection δ and natural frequency f_{nat} (in Hz) as function of beam and tip mass [SMAD]

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 the satellite (consider all mass concentrated in the vehicle's center of geometry):

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:

$$f_{nat} = 0.160\sqrt{\frac{AE}{ML}} = 0.160\sqrt{\frac{0.025m^2 \times 70GPa}{1000kg \times 2m}} = 149.6Hz$$

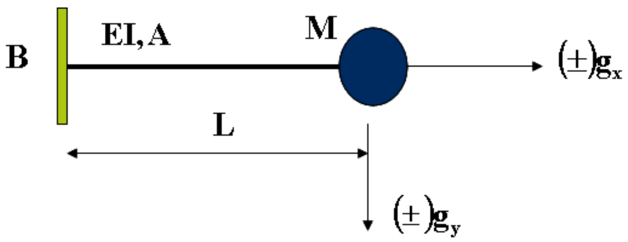
Lateral direction:
Area moment of inertia (see also annex F): $I = \pi r^3\, t = 0.0126\ m^4$
Spring constant 4.1E7 N/m
Natural frequency is 91.4 Hz or 574 rad/s

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. Compared to the total mass of the spacecraft (1000 kg) this is quite substantial. So we should seek ways to reduce spacecraft mass.

The launch vehicle is very important to consider with respect to excitation frequencies. However, next to launch vehicle 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 [Fortescue])</u>
When sizing for strength, we must consider stress at point B and buckling load. We use the same simple model (point mass at free end of mass less beam) 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 (is point of highest stress) can be calculated using:

$$\sigma_{tot} = \frac{g_y MLc}{I} + \frac{g_x M}{A} \qquad [47]$$

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:

$$\sigma_{ax} = \frac{P_{eq}}{A} \qquad [48]$$

Using axial stress, we can size the cylinder (thickness, diameter and length, whatever is appropriate) for maximum ultimate and yield stress using appropriate FoS. Thickness is highest at point B. For the purpose of this course, variation in thickness will be neglected and hence it is implicitly assumed that the thickness of the cylinder is constant along its length.

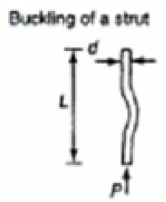
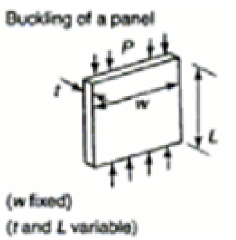
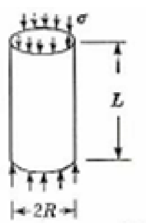
Method presented is based on assuming the cylindrical spacecraft can be modelled using a massless beam with a point load only. Solutions for several other commonly encountered configurations (for instance, multiple point loads + distributed load) are readily available in textbooks on mechanics of materials and engineering handbooks or have been dealt with in earlier classes on structures. Note that more detailed modelling generally leads to different dimensions of the structure and hence affects the mass of the structure. This is considered out of scope for this course. Here we aim for students to understand the key factors and how these will change the vehicle characteristic parameters.

Next to tension, we must also check if the cylinder can withstand buckling loads. If, for instance, we use the same simple beam structure as shown before, we find that the critical load is given by the Euler buckling load:

$$P_e = P_{cr} = \frac{\pi^2 EI}{4L^2} \qquad [49]$$

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

Table 26: Buckling relations of simple geometries

Case			
Characteristic equations $\sigma_{cr} = \frac{P_{cr}}{A}$	$P_{cr} = \frac{\pi^2 EI}{(L')^2}$ $I = \frac{\pi \cdot d^4}{64}$	$\sigma_{cr} = \text{constant}\frac{E}{(1-\nu)^2}\cdot\left(\frac{t}{w}\right)^2$	$\frac{\sigma_c}{E} = 9\left(\frac{t}{R}\right)^{1.6} + 0.16\left(\frac{t}{L}\right)^{1.3}$
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 [Fortescue]. 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 [49] 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. Vehicle mass is 1250 kg and is considered concentrated in the center of geometry of the cylindrical structure. 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,*

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 flight limit loads. We obtain:

- *In x-direction: Mg_x = 1250*9.81*6 = 73575 [N]*
- *About z-axis(bending load): MLg_y = 1250*9.81*2.5*1.5 = 45984 [Nm]*

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

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

Filling in numbers gives for the axial stress: σ = 164 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 safety factors, see later.

Checking for yield strength: For ductile materials (usually metals) we should also check whether the yield strength of the material is sufficient.

Checking for compressive strength, we find using the relation for critical stress for a cylinder from ***Table 26****, (use L = 2.5 m) a critical stress of 10.2 MPa. This is well below the compressive stress experienced by the cylinder and hence the cylinder fails if not strengthened. Please note that when strengthening the cylinder, the mass of the cylinder tends to increase.*

In the above example, we have assumed the wall of the cylinder to consist of a thin metallic wall (sheet). To allow for improved stiffness and buckling strength you may consider using sandwich materials. This is considered out of scope for the present work, but may be taken up by the interested reader to explore for him/herself.

Sizing for internal pressure
Pressure vessels must be sized to withstand the internal pressure. This will be dealt with in some detail when discussing launch vehicles. It compares well though with the sizing of a pressure cabin of an aircraft.

Materials selection
A commonly used material for spacecraft structures is aluminium. In more recent spacecraft also composite materials are used. For some special structures, like pressure vessels and truss corner sections, also titanium and steel are used. Next table taken from "Introduction to Aerospace Engineering – Structures and Materials" provides some important properties of metals used as structural material.

Table 27: Typical mechanical properties of some metallic materials used for space structures [AE1110]

Metal	Alloy	E	G	σ_y	σ_{ult}	ε_{ult}	ν	ρ
		[GPa]	[GPa]	[MPa]	[MPa]	[%]	[-]	[g/cm^3]
Steel	AISI 301	193	71	965	1275	40	0.3	8.00
	AISI 4340	205	80	470	745	22	0.29	7.85
	D6AC	210	84	1724	1931	7	0.32	7.87
Aluminium	AA 2024-T3	72	27	345	483	18	0.33	2.78
	AA 7475-T761	70	27	448	517	12	0.33	2.81
Titanium	Ti6Al-4V (5)	114	44	880	950	14	0.34	4.43
Magnesium	AZ31B-H24	45	17	221	290	15	0.35	1.78

A good measure of the strength performance of a material is its specific strength. It is defined as the yield strength divided by its density. It relates the strength of a material to its mass. A high specific mass means you are dealing with a material that offers high strength for low mass. Composite materials do have very high specific strength well in excess of values attained for aluminium or other metals.

Next to specific strength and stiffness, various other properties need to be considered when considering materials selection. This is shown in next table taken from the work of [Grastataro, 1995]. It compares how an aluminium alloy structure compares to a conventional composite structure.

Table 28: Manufacturing comparison matrix [Grastataro]

Parameter	Aluminum Alloy Structure	Conventional Composite Structure
Material	aluminum	graphite/epoxy
Material Cost	low	high
Material Advantage	cost	stiffness/weight/strength
Material Form	metal billet	molded parts
Manufacturing Process	machining	molding
Process Cost	moderate	high
Process Advantage	established technology	customized geometry
Fabrication Time	16 weeks	30 weeks
Tooling Cost	low	high
Structure Weight	64.4 kg (142 lb)	42.6 kg (94 lb)
Unit Fabrication Cost	\$133 k	\$400 k

Besides the lower mass of the composite structure it clearly is also much more expensive and takes much longer to produce.

Still, next to manufacturability, many more parameters need to be taken into account when selecting materials for structural applications. For some applications, also the materials thermal conductivity (how well the material can transfer heat by conduction) and its coefficient of thermal expansion are important. In addition, materials selected need to be compatible with space environment (extreme temperatures, presence of monatomic oxygen, etc.); some materials deteriorate in space due to UV radiation, ozone or atomic oxygen attack.

In case you would like to know more on materials selection for spacecraft applications, you are advised to read the section on material selection in [Fortescue]. Given the limited scope in the present work, we will mainly focus on how strength and stiffness aspects of materials influence the design of the structure.

Important structural 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 [Fortescue]) and include:

- Quasi-Static (or steady state) Loads (QSL), i.e. “Loads independent of time or which vary slowly, so that the dynamic response of the structure is not significant” (from ECSS E-32-T).
- 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 launch vehicle manual. For each launch vehicle 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 website.

An example of a load diagram is given in Figure 49. 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.

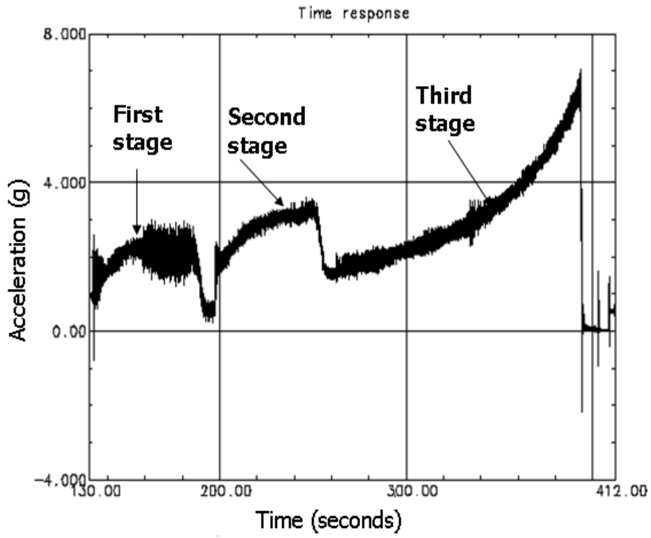


Figure 49: Typical acceleration loads during launch vehicle ascent flight [LVC]

Launch vehicle manuals specify maximum accelerations to be expected during launch vehicle ascent flight. An example is shown in Table 29. Important for design are the quasi static loads (QSL). This is the sum of the static and (low frequency) dynamic loads. It is specified in the center of gravity of the spacecraft on the launch vehicle. Longitudinal load is directed along the central vehicle axis, whereas lateral loads are in a direction perpendicular to this axis. Both longitudinal and lateral loading may occur at the same time. A minus sign refers to compression and a plus sign to tension. Largest load is usually the one of interest for S/C structural design.

Table 29: Ariane 5 flight limit loads [LVC]

Acceleration (g) / Critical flight events	Longitudinal		Lateral
	Static	Dynamic	Static + Dynamic
Lift-off	- 1.8	± 1.5	± 2
Maximum dynamic pressure	- 2.7	± 0.5	± 2
SRB end of flight	- 4.55	± 1.45	± 1
Main core thrust tail-off	- 0.2	± 1.4	± 0.25
Max. tension case: SRB jettisoning	+ 2.5**		± 0.9

Different events are given in the table. For a good design all conditions shall be assessed as to make sure that none of these events leads to a catastrophic failure.

Next to launch loads also other loads should be taken into account, like handling loads, transportation loads, shock loads, internal pressure loads (tanks) and thermal loads. Of these, transportation loads are associated with the transportation of S/C over water or land and through air. Typical load factors can be obtained from NASA handbook (NASA-HDBK-7005). Especially for rail and road transportation the vertical loads may be quite excessive. Shock loads may result from thrusters that are being activated on board or propellant sloshing or an unbalanced reaction wheel or from stage separation. Hereafter, we will mainly focus on withstanding launch loads. Internal pressure loads though will be dealt with in the section dealing with launch vehicle design.

For the design of solar panels, radar panels and large antenna dishes or beams usually launch loads are not determining as during launch these panels are usually in a retracted (or stowed) position and nicely secured against the body of the S/C. Once deployed, however, one should reckon with load conditions that occur in space. The higher the loads experienced in space, the heavier the structures must be. For planar elements, the loads are typically limited to maximum 0.2g as to limit the mass of these items. Of course, a lower value may be selected, but this then may also limit the thrust level of thrusters used for propulsive purposes. Some further study may be needed to determine the best limit value for the design of planar elements.

With respect to the dynamic loads/vibrations that are induced by shocks, sound waves reflecting back from the launch environment to the payload, rocket engine operation, etc. some special precautions are needed to prevent dynamic coupling between the low frequency dynamics of the launch vehicle and the payload modes. For this, it is required that the payload natural fundamental frequency must be higher than the low frequency dynamics (excitation frequency) as generated by the launch vehicle. This in general leads to a requirement on the stiffness (stiffness requirement) of the S/C carried as payload on board of the launch vehicle.

Table 30: Spacecraft stiffness requirements for specific launch vehicles [LVC]

Launch Vehicle	In lateral axis	In longitudinal axis (thrust direction)
Dnepr	≥ 15 Hz	20 Hz - 45 Hz
Delta II 7320	> 12 Hz	> 35 Hz
H-IIA	> 10 Hz	> 30 Hz
PSLV	> 18 Hz	> 40Hz
Soyuz BAI/KOU (spacecraft + adapter)	> 12 Hz	> 27 Hz
Soyuz BAI/KOU (spacecraft)	> 15 Hz	> 35 Hz
Zenit 2 SLB	> 6Hz	> 20Hz

For planar elements, like solar panels, radiator panels, and antennae, a typical requirement for the first mode natural frequency is [Fortescue]:

$$f_{nat} \geq 0.2\ Hz \qquad [50]$$

Use of safety factors
An important issue when designing a spacecraft structure is that it is able to function safely, i.e. without failure. In structures design, this is accomplished by using strength and stiffness values that can be guaranteed in say 99.7% of all cases and by introducing a factor of safety which aims to reduce uncertainties in the design.

Factor of safety (FoS) or safety factor[20] is a term describing the structural capacity of a system beyond the applied loads or actual loads. They are used to increase the capability of the structure such that the

[20] 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.

risk of a failing structure is reduced to an acceptable level. Hence, these factors provide for margin in the design. Figure 50 shows how factors of safety can be applied in design to reduce uncertainty in the design.

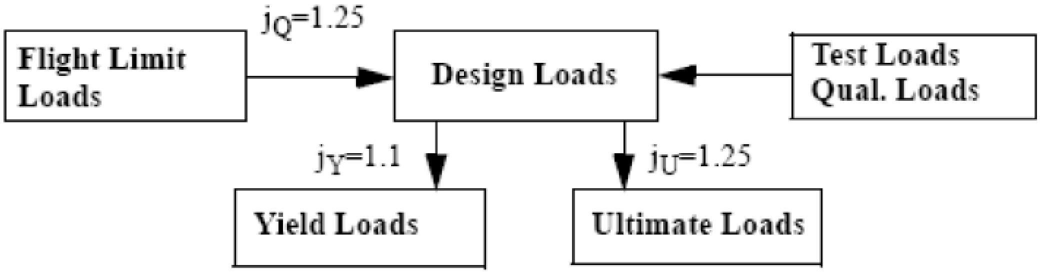


Figure 50: Factors of Safety (FoS) for metallic structures in case of verification by testing

Here the following factors and accompanying definitions are used:

- 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 (DL) is derived by the multiplication of the (Flight) Limit Load by the design safety factor.
- Yield load = Design load x j_Y
- Ultimate load = Design load x j_U

Flight limit load for a given design condition is the maximum occurring load with, for example, a probability of 97.7% (2σ). The stress that is calculated with applied flight limit loads is called the limit stress. Design load is the limit load multiplied by the design (safety) factor to avoid risks during the design and the test phase. –Design limit load is also known as qualification load. Design (safety) factor allows for incorporating mass growth and uncertainty in the models used. Use of very simple analysis methods requires higher design factor than for more advanced, but more complex analysis methods. Hence design factor is variable. Initially in the early phases of design it can easily be 1.25, whereas in later phases of the design it may reduce to 1 (or only slightly above). The design factor may also vary depending on how the structure is verified (by analysis only or by test).

FoS values 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 verification philosophies have been collected in amongst others [ECSS-30]. Table 30 (see next page) is a typical such example valid for unmanned space missions. It shows safety factor in range 1.1 to 5.0, depending on the type of structure used. Notice that in the table also a specific factor of safety is included for buckling.

Next to FoS also other terms are used to indicate margins in the design, be it that they may have a slightly different meaning. See for instance [Fortescue] 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 loads.

Summary

In this section the design of the structures and mechanisms subsystem of a spacecraft has been discussed. The following steps are considered as main steps in the design of such a subsystem:

1. Define the main functions for the structures and mechanism subsystem; some specific functions have been included explaining the need for structures and mechanisms
2. Establish requirements for the structures and mechanism subsystem.
3. Set up different options for the structures subsystem. For instance one can consider a cylindrical central structure or a box-like structure or a truss structure.
4. Analyse the options generated. In this work it is shown how the load carrying capability and the stiffness of the structure can be influenced by adapting the structural design in terms of lay-out (adapting the area moment of inertia or the cross-sectional area of a beam) and the materials used.

The analysis presented is quite simple, but there is nothing against implementing more detailed methods when the design progresses.

5. Compare the design options. This steps has not been discusses specifically. However, the Table 28 shows a nice example of two structural designs being compared. Having multiple options allows assessing all options in terms of cost, mass and maybe also on a number of other issues. Of course all options must allow the structures and mechanisms subsystem to function as intended.
6. Make choice. Once all options have been analysed the best option can be chosen. Sometimes a combination of options may give the best solution
7. Evaluate and if necessary iterate to reduce the uncertainty in the design.

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 51: FoS unmanned spacecraft [ECSS-30]

Problems

1. Give an account of the main functions of the structures and mechanism subsystem (STRUCS).

2. Explain in what degree STRUCS (the structures system) determines vehicle mass, cost, power and reliability.

3. General:
   a) Explain why we should use safety factors in the design process.
   b) Explain the importance of the stiffness of the mechanical mount between the spacecraft and the launch vehicle.
   c) What is the definition of the Margin of Safety?

4. You have been assigned the task of designing the structure of a spacecraft that is to be launched using Ariane 5G as the launch vehicle.
   a) Determine for this launch vehicle the fundamental frequencies in the longitudinal direction
   b) Determine the dimensioning loads the spacecraft should be able to withstand.

   Hint: Use ESA Launch Vehicle Catalogue and perform a search for the requested data.

   Answers: a) The fundamental frequency in the longitudinal axis must be ≥ 31 Hz; b) 6.0 g in longitudinal direction and 2.0 g in lateral direction. Note that we have combined the static and dynamic loads.

5. The primary structure of a spacecraft consists of a thin-walled cylinder of 5 m length and 1 m diameter. This cylinder carries on top a payload of mass 250 kg. Given that:
   - Payload acts as a point load at the free end of the cylinder and is only load to be considered
   - Young's modulus of the cylinder material E = 70 GPa
   - Maximum axial acceleration experienced by the S/C is 6 g (launch load)
   - Mass of the cylindrical structure itself is negligible

   Calculate for this cylinder the required minimum wall thickness (t) to ensure that the fundamental frequency in axial direction is equal or larger than 35 Hz.

   Answer: t ~0.3 mm

6. The primary structure of a spacecraft is a cylinder of 5 m length, 1 m diameter and a wall thickness of 0.5 mm. This cylinder is carrying a payload of mass 250 kg on top. Given:
   - Payload acts as a point load at the free end of the cylinder and is only load to be considered
   - Young's modulus of the cylinder material 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.

   Calculate the maximum stress in this cylindrical structure for tensile strength (compression or tension) assuming the beam behaves as a cantilever beam with a point load at its free end.

   Answer: 69.2 MPa

7. A spacecraft is carrying a small payload at the end of a cantilever boom. Given are:
   - Payload mass: 8 kg
   - Payload mass acts as point load.
   - Boom length: 10 m
   - Boom outer diameter: 0.1 m (boom is a thin cylindrical wall structure)
   - Thickness of boom wall: 0.35 mm
   - Boom mass is negligible.
   - Boom material is aluminium with ultimate strength of 310 MPa and stiffness 69 GPa

   During several instants during the spacecraft's life thrusters are activated thereby producing a vehicle acceleration of 2 m/s^2 in a direction lateral to the boom. Determine for the boom the ratio between maximum ultimate stress and critical stress for buckling using simple beam theory.

   Answer: Ultimate stress – to –critical stress ratio for buckling is 0.263.

8. Sizing for strength of beam like structures is done mainly focussing on beams with cylindrical cross-section and only considering point loads at the end of the beam. It is interesting to ask yourself the question what would change in case the beam has a square or rectangular cross-section or a distributed load or several point loads at different distances from the root are acting on the beam.

   Answer: No answer is given here. This question is for you to realize that other courses may provide you with the material needed to answer such questions.

Further problems for exercising upon are available via the course web pages (look for Mobius exercises) or can be found in the tutorial material.

4.2 Thermal Control

The purpose of the thermal control system (TCS or THERMAL) 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 (°C)
  - Most equipment designed to function at Earth temperatures
  - 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 31: Typical spacecraft component temperatures [Fortescue]

S/C component	**T_{min} (°C)**	**T_{max} (°C)**	**ΔT (°C)**
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 32: Typical allowed temperature gradients and temperature variations for some components (courtesy ESA)

- limited Temperature Gradients
  - ΔT < 5°C across optical instrument (1.5 m)
  - ΔT/Δx < 2°C/m for structural element
  - ΔT < 5°C between MMH and NTO tanks
- Stable Temperatures
  - ΔT/Δt < 5 K/h for typical electronic unit
  - ΔT/Δt < 0.1 K/mn for CCD camera
  - ΔT/Δt < 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

For more detailed data, see appendix D.

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):

$$q = \frac{Q}{A} \qquad [51]$$

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 [52]. Here Q (rate of heat transfer) and W denote the sum of all the heat and work transfer interactions experienced by the closed system.

$$Q = \frac{dE}{dt} + W \qquad [52]$$

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:

$$\dot{Q}_{emitted} = \varepsilon \cdot \sigma \cdot A \cdot T^4 = q_{emitted} \cdot A \qquad [53]$$

- 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 [54], 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.

$$\dot{Q} = \frac{k \cdot A}{l} \cdot \Delta T = \frac{k \cdot A}{l} \cdot \left(T_2 - T_1\right) \qquad [54]$$

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 [55], 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.

$$\dot{Q} = h_c \cdot \Delta T \cdot A \qquad [55]$$

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 52 for a spacecraft nearby Earth.

From Figure 52 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.

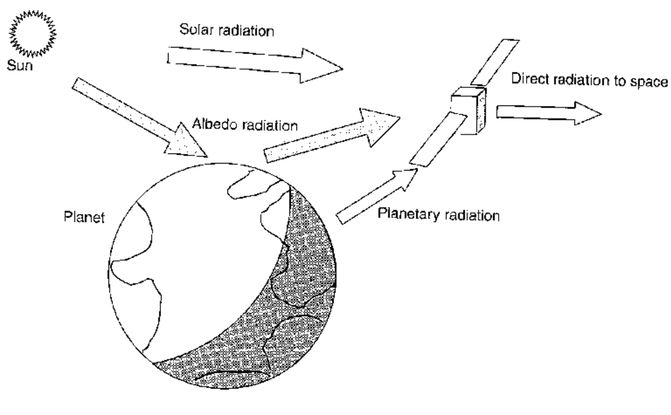


Figure 52: Typical spacecraft thermal environment [Fortescue]

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 about 5780 K, where the effective radiating temperature of a celestial body is determined using relation [53] under the assumption that ε = 1. It follows:

$$q_{emitted} = \sigma \cdot T^4 \qquad [56]$$

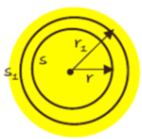
And hence:

$$T_e = \sqrt[4]{\frac{q_{emitted}}{\sigma}} \qquad [57]$$

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[21]).

CONSERVATION OF ENERGY



$$q(r_1) \cdot 4\pi r_1^2 = q(r) \cdot 4\pi r^2$$

$$q(r) = \frac{\text{const}}{r^2}$$

If we can regard sphere s_1 as fixed

$$q \propto 1/r^2$$

Hence, it follows for the solar intensity:

$$J_s = \frac{P}{\left(4 \cdot \pi \cdot d^2\right)} \qquad [58]$$

Solar intensity at a 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 53. It clearly shows that the further away from the Sun, the lower the solar intensity received (and the colder space will be).

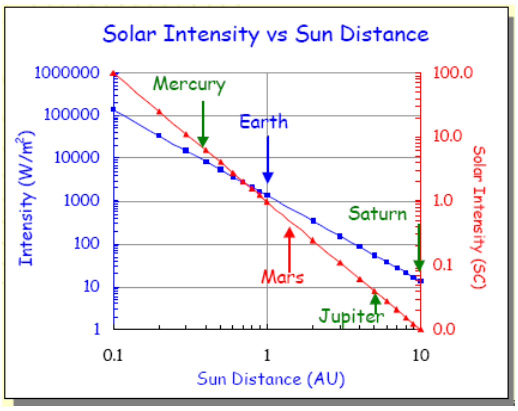


Figure 53: Variation in solar intensity with (average) Sun distance

21 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.[58]. 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 on the previous page.

<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:

$$J_a = a \cdot J_s \cdot F \qquad [59]$$

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 [Fortescue], 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.

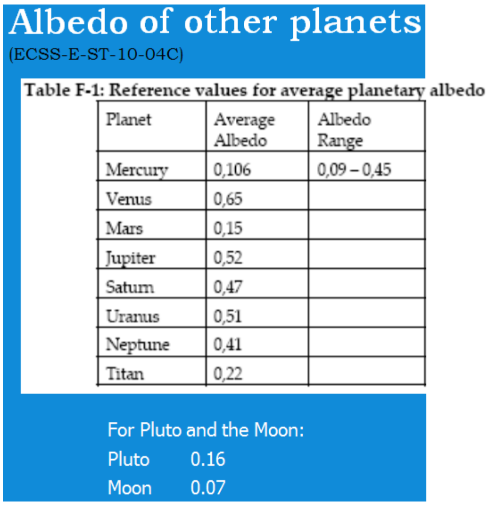

Albedo of other planets
(ECSS-E-ST-10-04C)

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 54: Reference values for average Albedo of various 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:

$$\text{In shadow of planet}: F = 0$$
$$\text{Sun lit side of planet}: F = \left(\frac{R_{planet}}{R_{orbit}}\right)^2 \qquad [60]$$

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 [Fortescue].

<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:

$$J_{IR} = \sigma \cdot {T_{IR}}^4 \qquad [61]$$

The effective radiating temperature of some planets is given in Table 33. For Earth with an effective temperature of 255 K, planet flux (at the planet surface) is equivalent to 240 W/m^2.

Table 33: 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 radiating planet. This makes that in deep space planet IR flux is (almost) negligible.

Heat fluxes as calculated for a real mission

Figure 55 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.

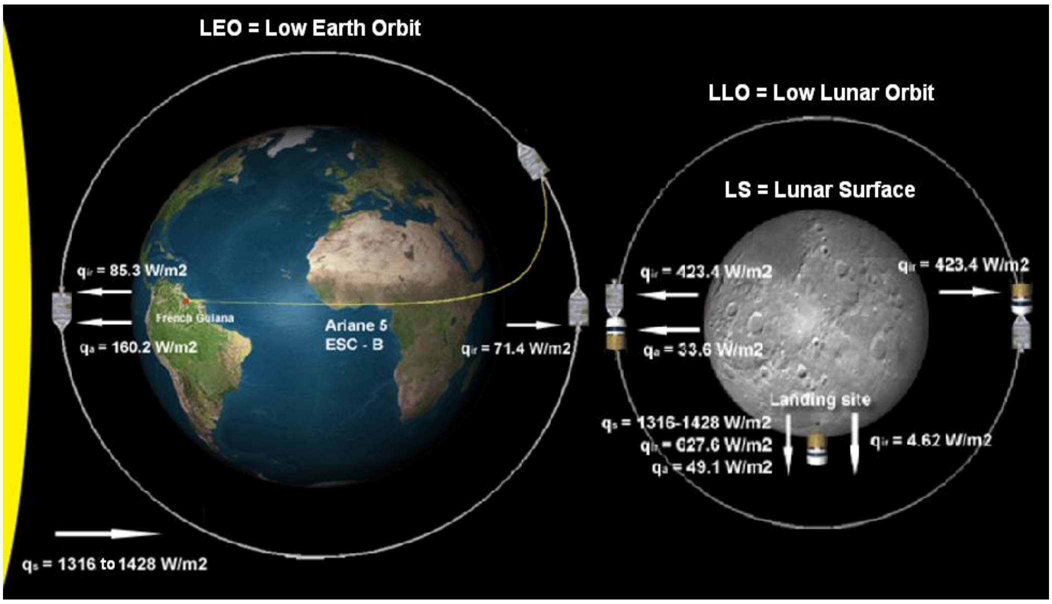


Figure 55: Overview of heat fluxes as experienced for the M3 mission

Students should prepare for calculating 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.[62], 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.

$$\Delta T = \frac{\dot{Q} \cdot \Delta t}{M \cdot C} \qquad [62]$$

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 is 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.[63], 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 $\alpha = 1$.

$$\alpha + \tau + \rho = 1 \qquad [63]$$

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 [Fortescue, 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:

$$\dot{Q}_{in} = \dot{Q}_{out} = \dot{Q}_{absorbed} + \sum P_{dissipated} = \dot{Q}_{emitted} \qquad [64]$$

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

$$\dot{Q}_{absorbed} = \alpha_s \cdot J_s \cdot A_i + \alpha_s \cdot J_a \cdot A_i + \alpha_{IR} \cdot J_{IR} \cdot A_i \qquad [65]$$

$$\dot{Q}_{absorbed} = \alpha_s \cdot J_s \cdot A_i + \alpha_s \cdot J_a \cdot A_i + \varepsilon_{IR} \cdot J_{IR} \cdot A_i \quad \left(\alpha_{IR} = \varepsilon_{IR}\right) \qquad [66]$$

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 [65] 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 [66], 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:

- α for absorption in solar spectrum
- ε for absorption and 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:

$$\dot{Q}_{emitted} = \varepsilon_{IR} \cdot \sigma \cdot A_e \cdot T^4 \qquad [67]$$

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 equals the energy emitted:

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

- *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°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.

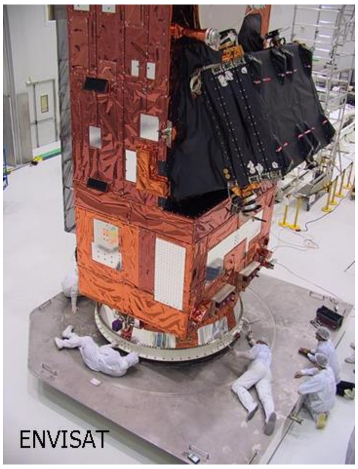
For instance, the equilibrium temperature for the flat plate type calculated in the foregoing example may be lowered by selecting a larger area for radiating heat away. Another option might be to select materials with a different ratio between solar absorptance and IR emissivity, see for instance Table 34.

Table 34: Solar absorptance and hemispherical emissivity of typical space materials

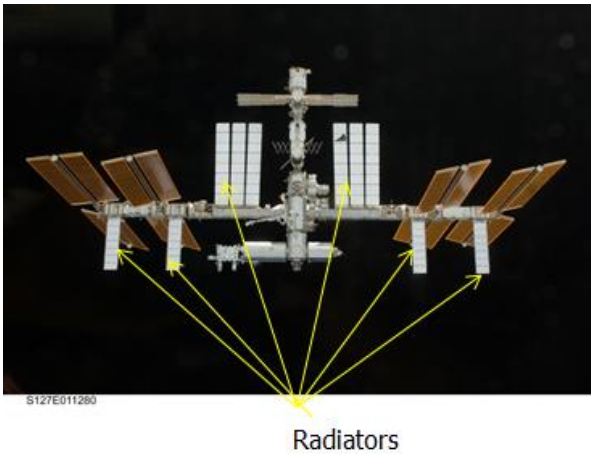
Material	Solar absorptance (BOL)	Hemispherical emissivity
Black paint	0.96	0.75 to 0.88
Aluminized teflon foil 5 mil	0.14	0.78
Sivered teflon foil 5 mil	0.09	0.80
Aluminized kapton foil 2 mil	0.42	0.72
Aluminized kapton foil metal side	0.12	0.05
White paint	0.17 – 0.38	0.82
OSR without glue gaps	0.09	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 is 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 (Kirchhoff's law). Note that all data given in table are indicative only. For instance, various black and white paints exist with properties that can be different depending on the specific nature of the coating.

Figure 56 shows thermal features of 4 different spacecraft. S/C top left is ENVISAT a large environmental satellite in Low Earth Orbit. It is covered with brown/gold shiny colored aluminized Kapton thermal control blankets (multi-layer insulation or shortly MLI) with some parts 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. Figure top right shows the International Space Station with about 10 large deployable radiator areas, which allow radiating excess heat away into space. Note that plane of radiators is perpendicular to the plane of the solar panels. Each one of the six main radiators (there are smaller ones for the electrical subsystem) is designed to evacuate 16 kW by means of an ammonia pumped fluid loop. Each radiator consist of 8 panels of 0.225x0.275 m^2 deployed by a scissor mechanism to 1.875 m full length, has a mass of 1000 kg, and is made of bonded honeycomb panels coated with a white ceramic thermal paint. The angle between the orbit plane of ISS and solar direction varies between +75° and -75°. Figure bottom right shows the Ulysses spacecraft covered with thermal control blankets consisting of 20 layers of Kapton covered with a layer of conductive indium tin oxide with a distinctive gold color. Last figure shows HYLAS S/C with thermal control blankets to prevent heating of the spacecraft by external sources. Panels perpendicular to the solar wing are covered with optical surface reflectors to allow heat being radiated into space.

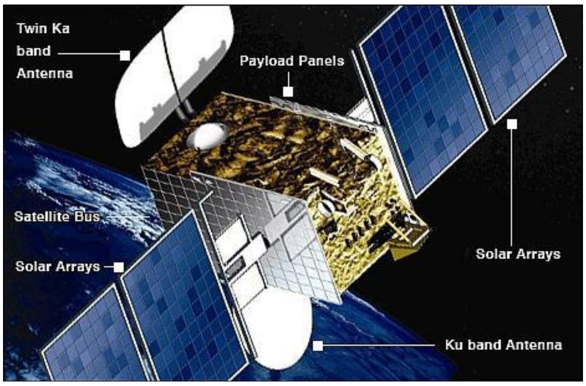


Envisat S/C (courtesy ESA)



ISS with thermal radiators (courtesy NASA)

Ulysses S/C (courtesy ESA)



HYLAS-1 S/C (courtesy NASA)

Figure 56: Thermal design of a satellite

From the foregoing, it shall be clear that a major part of the surface of the spacecraft is involved in thermal control. As components of the thermal control subsystem we mention:

- Paints/coatings that modify the emissivity and/or absorptance of a surface, see Table 34 for typical properties. White paint is usually used to create a radiating surface, while black paint is used to create a warm surface/body. Approximate thermal coating mass is in range 0.24 kg/m^2, depending on type of coating and coating thickness. Coating thickness varies with the coating applied. It generally is in range 5-13 mils (1 mil is one thousandth of an inch).

- Multi-layer insulation (MLI), see Figure 57; Multiple layers of thin foils, see figure, that allow to achieve low emissivity in the range of 0.003-.03. Typical multi-layer insulation consists of 20-30 inner layers of ¼ mil aluminized Mylar, innermost and outermost layers of 1 or 2 mil aluminized Kapton weighing approximately 0.3 kg/m^2. This data has been taken from https://www.dlr.de/iaa.symp/Portaldata/49/Resources/dokumente/archiv4/IAA-B4-0901.pdf. The thin foils used for the inner layers are also referred to as first surface mirrors as they exist of a mirror surface fixed on a substrate below or two mirror surfaces with a substrate in between.

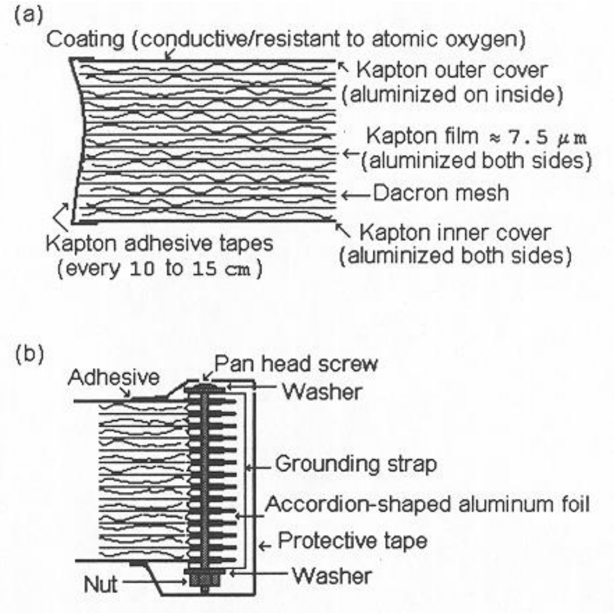


Figure 57: MLI construction: a) typical lay-up and b) electrical grounding (courtesy Dutchspace)

- Second Surface Mirrors or Optical Surface Reflectors (OSR), see Table 34, are generally used to create radiators. Second surface mirrors consist of a substrate on a mirror surface, compare first surface mirrors. The mirror surface reflects most radiation into space (low absorptivity), whereas the substrate ensures a high emittance to emit heat. OSRs are usually heavier than using white paint. OSRs approximate mass is 1 kg/m^2 [Brown].

- Heat pipes: To transport heat from surfaces of high temperature to surfaces with lower temperature. Heat pipe mass is approximately 0.33 kg/m (aluminum pipe, 1.27 cm in diameter) [Brown]. Of course other diameters may also be feasible, which allows adapting the mass of the heat pipe system.

- Radiators: Various types of radiators can be distinguished including passive and active radiators. Passive radiators are radiators that do not require electrical power, and have no mechanisms that can fail. As such they are low cost, lightweight, and have a low failure rate. Passive radiators can be distinguished after radiators directly connected to the heat source and radiators connected via heat pipes. The former option typically uses surfaces covered with OSRs or certain coatings. The latter option essentially uses heat pipes to transport heat from a hot spot to a cold spot where the heat can be radiated out into space. Heat pipes are filled with a fluid (typically ammonia) that at a hot spot heats up and vaporizes. The fluid is transported to a cold spot (the radiator) where it will condense as the radiator will radiate heat out into space. To enlarge the radiating area per pipe, fins may be attached to the pipes. To enlarge the heat flow, the fluid flow may be increased by a pump. In that case the radiator turns into an active cooling system.

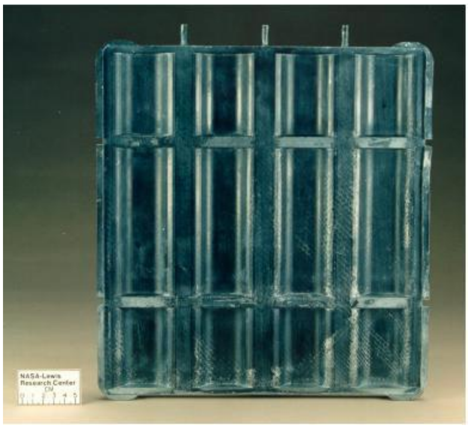


Figure 58: Radiator panel (courtesy NASA)

Active radiators can radiate from 30-40 W/m^2 for body-mounted radiators (radiating on one side only) to 180 W/m^2 for deployable radiators which radiate on both sides. Typical mass for active, deployable radiators is in range 5-15 kg/m^2 not considering tracking. From: Conceptual design and flight simulation of space stations by R. Bertrand.

- Heaters: To provide local heating for instance to prevent propellants from freezing or a drastic reduction in battery capacity. Important type of heater is the patch heater, which consists of an electrical-resistance element sandwiched between two sheets of flexible electrically insulating material. Approximate heater weight is 2 kg/m^2 [Brown]. Typical watt densities are in range 2-6 W/cm^2 [Minco].
- Louvers: "These are mechanical devices that, in effect, regulate the area of a radiator in response to its temperature. The regulation is preset to accommodate a wide range of heating within a relatively small change in temperature. The louvers set used most is the venetian blind, shown in Figure 59. It is a framed array of highly reflective blades, with central shafts that fit tightly into the center of bimetallic spring actuators calibrated to expand or contract to various angular positions at prescribed temperatures. As radiator heating increases, the rise in the temperature within the housing warms up the actuators, which then generate thermal torques that rotate the blades toward their open position leading to an increase in radiation to space. When the radiator temperature decreases, the actuators move the blades toward their closed positions, blocking more of the radiator's view to space and increasing the resistance to radiation. This automatic opening and closing of blades compensates for changes in dissipation and environment heating and keeps the temperature within a narrow band." [Zarchan]

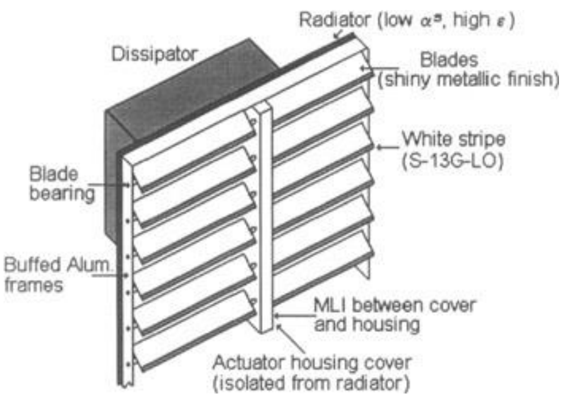


Figure 59: Schematic of a louver system [Zarchan]

Next figure illustrates that a single spacecraft may have various thermal control components.

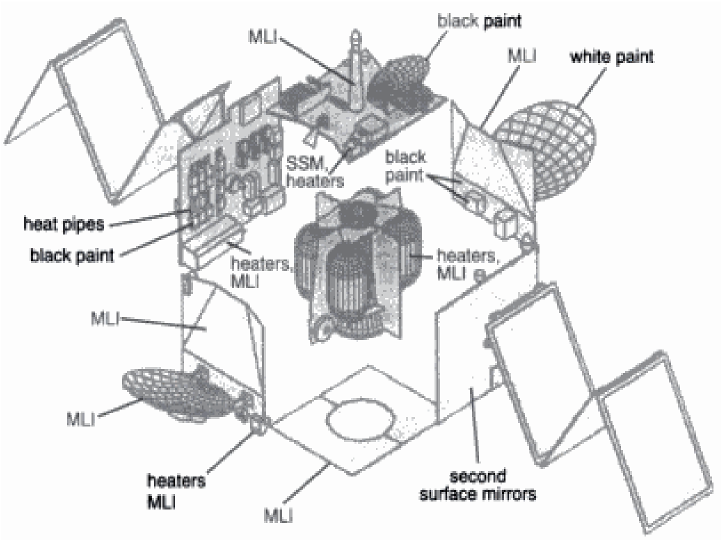


Figure 60: Illustration of S/C thermal control

The heaters in the above design are to keep the propellants from freezing, whereas the second surface mirrors are looking to cold space so as to radiate as much heat as possible. The solar panels should remain as cold as possible as this increases power output. White paint is used on surfaces that need to stay cool, whereas black paint is used on surfaces that need to remain warm. Heat pipes are used to transport heat from a hot spot to a spot (radiator) where excess heat can be radiated into space.

Summary

In the preceding sections, theory has been discussed that allows for analyzing the effect of shape and emissivity and absorptivity on vehicle temperature. Next table illustrates the effect that can be accomplished by changing shape and or selecting a different coating.

Table 35: Effect of shape and coating on equilibrium temperature of body

Temperature °C	White Paint $\alpha_s = 0.15$ $e_H = 0.9$	Black Paint $\alpha_s = 0.9$ $e_H = 0.85$	Aluminium $\alpha_s = 0.15$ $e_H = 0.05$
Flat plate	- 60	+ 60	+ 160
Sphere	- 95	+ 10	+ 90
Cylinder	- 100	- 5	+ 80
Hexagonal prism	- 105	- 10	+ 70
Cube	- 110	- 20	+ 55

Preference is to have a fully passive thermal control system (no heaters and/or active coolers). However, if that is not possible, we need to deliver a design that allows for minimizing the power consumption.

Problems

1. The black body or effective radiating temperature of the Sun is ~5780 K. The Sun has a radius of **6.955E8** m. Determine:

    a. Total power emitted by the Sun.
    b. Solar flux at the surface of the Sun
    c. Solar flux at a distance of 1 AU

    Answers: a) 3.847E26 W, b) 63.3 MW/m^2, c) 1368 W/m^2

2. Consider that the average solar flux received by Earth is 1368 W/m^2. Earth albedo is 0.33 and the heat produced by Earth itself (volcanoes, Earth power plants, etc.) is negligibly small. Determine:
    a. effective radiating (or equilibrium) temperature of Earth.
    b. incident IR flux on a spacecraft in circular orbit at an altitude of 900 km (hint: Earth radius is 6371 km)

    Answer: a) 255 K, b) 184.1 W/m^2

3. A cubic spacecraft of linear dimension 2 m has a temperature of 300 K. Determine total power emitted by the S/C in case the S/C is covered by a material with emissivity 0.6.

    Answer: 6.6 kW (S/C surface area that radiates heat is 24 m^2 (6 sides of 2 x 2 m^2 each)

4. Solar flux received by Earth is 1368 W/m^2. Earth albedo is 0.3. Determine:
    A. Transmission and absorption factor of Earth for solar radiation
    B. Maximum albedo flux incident on spacecraft in a 1000 km altitude circular orbit about Earth considering a view factor F equal to 1.
    C. Same question as under B for an orbital altitude of 10,000 km. Use a more realistic estimate for F (you may assume that the local vertical coincides with the Sun's rays)

    Answers:
    A) Transmission factor = 0 and absorption factor = 0.7
    B) Albedo flux is 1368 W/m^2 x 0.3 x 1 = 410 W/m^2
    C) 62 W/m^2 (F = (6371km/16371km)^2 = 0.15 => 1368 W/m^2 x 0.3 x 0.15)

5. Consider a spherical shaped S/C in Earth orbit. The S/C is covered with a black paint with an absorption factor of 0.96 and an emission factor of 0.75. Calculate for this S/C its equilibrium temperature in full sunlight under the assumption that albedo, IR and internal heat dissipation are negligible.

    Answer: 297 K

6. Consider a spherical spacecraft of diameter 2 m in a 1000 km circular orbit about Earth. This spacecraft has a solar absorptivity of 0.4 and an IR emissivity of 0.2. Summed internal heat dissipation of all instruments is 150 W. Furthermore are given an Earth albedo of 0.3 and a solar flux at 1 AU of 1368 W/m^2. Determine:
    A. Solar flux incident on S/C
    B. IR flux incident on S/C
    C. Maximum albedo flux incident on S/C
    D. Hot case equilibrium temperature of the S/C (S/C in full sunlight, full visibility)
    E. Cold case equilibrium temperature of the S/C (S/C in Earth shadow, no internal heat dissipation)

    Answers: A) 1368 W/m^2, B) 179 W/m^2, C) 410 W/m^2 (F = 1), D) 364 K, E) 168 K

7. Consider a solar wing of a spacecraft. This wing is producing electrical power at a distance of 4.4 AU from the sun. Its frontal area (area covered with solar cells) is 10 m^2.
   - Solar incidence angle is normal to the solar panel
   - Solar flux at 1 AU is 1400 W/m^2
   - alpha/epsilon ratio of the solar cells (illuminated side) is 0.8/0.7
   - alpha/epsilon ratio of the back of the panel is 0.5/0.7
   - Panel is thermally insulated from the spacecraft with the back essentially seeing cold space only
   - Panel produces 108 W electrical power

   Determine for this solar wing the equilibrium temperature under the given conditions.

   Answer: 156 K.

   Note that for an active solar panel (i.e. the panel is generating electric power), some of the power absorbed by the panel is converted into electrical power ($P_{electric}$) and hence the power balance governing the calculation of the wing temperature should be written as:

$$\dot{Q}_{absorbed} + \sum P_{dissipated} = \dot{Q}_{emitted} + P_{electric}$$

8. Consider a space probe in orbit about the planet Mercury. Mercury is close to the Sun with a closest distance of 0.3 AU. Because of this close distance to the Sun, the probe will run hot. To prevent the probe from getting too hot, it is shielded by a thermal shield. This question is about the maximum equilibrium temperature the probe will attain even though when protected by the thermal shield. Given are:

   - Probe has a cylindrical planform
   - Shield has a semi-cylindrical planform of same cylinder height as the probe
   - Probe and shield have identical radius; there exists a tiny gap in between probe and shield as to prevent heat conduction between the two, but this is not of importance for area calculation of probe and shield
   - Only solar radiation is considered (no albedo or Mercury IR)
   - Solar radiation is perpendicular to shield cross-sectional area
   - Probe is fully shielded
   - Shield is coated on side facing the Sun with white paint with α/ε being 0.15/0.9
   - Shield is coated on back side with coating that provides α/ε being 0.2/0.8

   You are to determine the equilibrium temperature of the shield/probe for the following cases:
   a) The **equilibrium temperature** of the shield in case the back of the shield sees only cold space (this is not true, but allows to get an idea of the minimum shield temperature feasible). In this case the presence of the probe itself is neglected.
   b) The equilibrium temperature of the probe itself assuming that the presence of the probe does not affect the shield temperature, all heat radiated at the back of the shield flows to the S/C body and that the cylindrical probe sees only cold space (again, this is a simplification, but allows to find a first temperature for the probe itself. The resulting temperature is considered to be an ideal lower temperature value for the probe).

   Answers: a) 350.2 K (solar intensity at 1 AU is taken equal to 1367 W/m^2), b) 278.2 K

Further problems for exercising upon are available via the course web pages (look for Mobius exercises) or can be found in the tutorial material.

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. As in the early design stages very little data is available one generally uses any available information to come up with a first guestimate. For instance, Table 36 provides an overview of installed power per mission type [ESA bulletin 87].

Table 36: 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

Now if the mission type is known, the above table can be used to come up with a first very global power estimate. Still the uncertainty is quite large as is indicated by the large power range indicated in the table. For instance, for science category spacecraft the mid-range value is 850 W, but apparently some spacecraft will do with 200 W while others require 1500 W.

An alternative approach which allows for a more accurate estimate (and hence reducing power margin) can be followed in case the electrical power usage of the payload is known. This is based on the reasoning that payload power and total power generated are related. This approach has been described in chapter 3 and is based on collecting (average on station) power data for comparable spacecraft, see for instance the data collected in Table 1 and appendix B, and determining the relation that exists between payload power and total generated power. A number of such relations have been collected in Appendix C. These relations typically have a reasonable (but not perfect) estimation accuracy with an RSE in the order of 20%. Because of the uncertainty one should take into account a reasonable margin like for instance 2 RSE. The same approach of collecting data from similar historical spacecraft and determining estimation relationships has been followed in generating a first power budget for the spacecraft. For instance, Figure 61 shows the average power distribution as determined based on a large number of GEO telecommunications satellites.

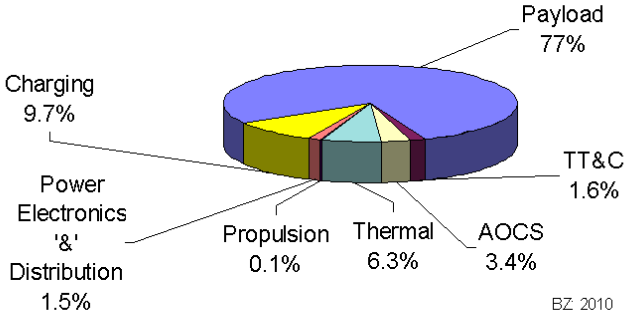


Figure 61: Power distribution of GEO telecommunications satellites (average percentages)

So given some payload power, total power and the power distribution over the various subsystems, expressed either as a percentage or fraction of total power generated also the power required by the subsystems, including the electric power subsystem, can be estimated, see also the earlier section on power budgeting (chapter 3).

A third method can be used once we know more of 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, their working modes, like on/off, idle, etc., and a determination of their power usage for the various modes and the duration over which a certain mode is active. These data are generally made into a power budget, i.e. a list of all electrical apparatus on board (the loads) with per operating mode the electrical properties and duty cycle such that it is clear how long a certain mode is operative per day/orbit/etc. Typical such power budgets (although still not very detailed) are shown in Table 21 and Table 37.

Table 37: 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 above gives power values for two different working modes, distributed over the observation instruments and the various spacecraft subsystems. The maneuvering mode typically is on for 5-10% of the orbital period. A margin of 10% has been included for each mode 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 in the tables is typical for an early design when most design details are still lacking.

Not included in the above 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 and peak) 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 example then should show that we must make a distinction between average power and peak power to be delivered by the system.

Note that the budget examples shown in Table 21 and Table 37 are simple budgets showing little detail. Later in the design budgets will become more detailed and more accurate. This will be left for you to explore for yourself.

Once we know the power level, the duration over which this power should be delivered and hence the total energy needed, 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 makes up a large part of the spacecraft cost. For small satellites, the EPS makes up roughly one third of the total vehicle cost.
- 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 38.

Table 38: 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 external 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. Next figure, taken from [Fortescue, figure 10.1] illustrates how for space missions power level and mission time is related to the selection of some type of power source.

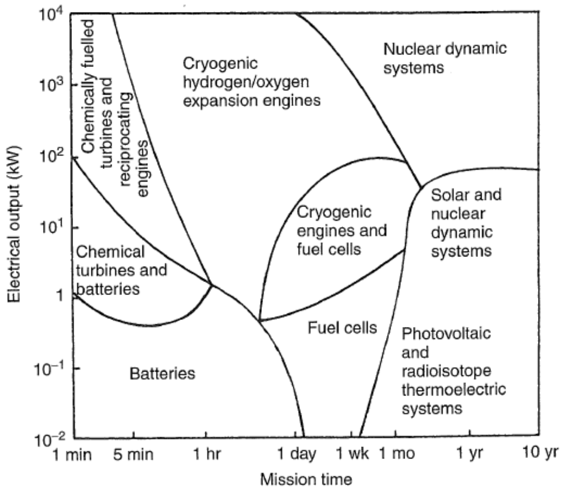


Figure 62: Power output versus mission duration [Fortescue]

In the next few sections, we will discuss the design of the most important EPS in some more detail, including photovoltaic systems, batteries, fuel cells and radioisotope thermoelectric systems. Other types of EPS mentioned in foregoing figure are considered less important for the main stream of spacecraft.

Photovoltaic 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 (PV) effect. "Photo" essentially means light, and "voltaic" refers to voltage, which is a unit of potential electrical energy.

The main constituents of a photovoltaic based power system are:

- Primary power source, i.e. a solar array; solar arrays begin with a single solar energy cell known as a photovoltaic cell. Since a single solar cell only generates a relatively small electrical current, several solar cells are connected together to create a solar panel. When you combine several solar panels, you create a solar array.
- Secondary power source that provides for power during eclipse periods or when the power peaks. Secondary power source is actually an energy storage device that gives out energy when needed and can be recharged at a later stage when energy from the primary source is abundantly available. Typical secondary power source is a battery, but also some fuel cells can be used. To distinguish rechargeable batteries from single-use (non-rechargeable) batteries the former is generally referred to as secondary batteries and the latter as primary batteries.
- 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 using batteries as the secondary power source is given in Figure 63.

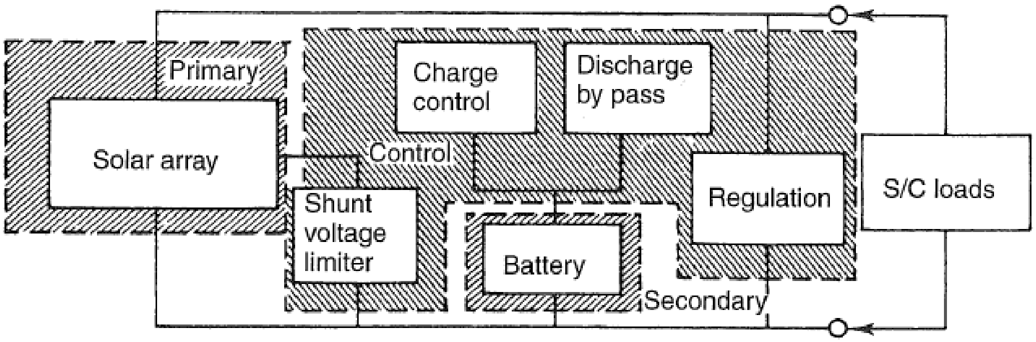


Figure 63: Schematic of photo-voltaic based EPS

When starting the design of a PV system, it should be clear that we need to know about how much power is needed and when. It should also be clear that we need to distinguish between two main operating modes for the PV system; a first mode wherein the solar array is receiving sun light (sunlit period) and provides power not only to the loads, i.e. the equipment requiring electrical power), but also to the secondary power source for recharging and a second mode wherein the solar array is not operative (when the array is shadowed by for instance Earth, see Figure 64.) and the required energy is drawn from the secondary power source. Sometimes a third (peak power) mode is distinguished, wherein the battery supports the primary power source to deliver peak power for those cases where the primary energy source itself cannot deliver sufficient power. All these aspects should be taken into account by the designer when designing an EPS system.

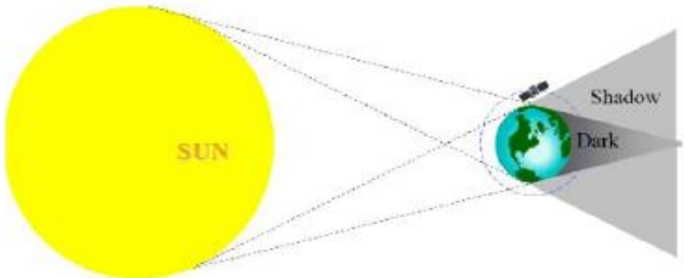


Figure 64: S/C experiencing solar eclipse during its rotation about some planet

Next relation illustrates how the power (P) that should be delivered by the solar array (SA) is determined using an energy balance. The relation applies to a spacecraft in orbit about some planet with the spacecraft experiencing sunlit and eclipse periods:

$$P_{SA} \cdot t_d = \frac{P_d \cdot t_d}{\eta_d} + \frac{P_e \cdot t_e}{\eta_e} \qquad [68]$$

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 (control) system during day time and night time, respectively. Note that:

- Second term on right hand side (RHS) gives the total energy that should be provided for by the battery even though the energy is delivered in the sunlit period.
- A distinction is made in the relationship between day time and night time power consumption. This is because during night time one might operate only a limited set of equipment or at reduced power to conserve energy.
- In the above relationship the day time and night time power are considered constant. Still, you may also look at them as average power over the day and night time period, respectively.

- Day- and night time path efficiency depend on the efficiency of the control and distribution system. Day time efficiency is usually higher than night time efficiency as in the latter case also the efficiency of the storage system (battery) and its management systems need to be taken into account. Some details are presented in Figure 65. For first design purposes, daytime efficiency is typically taken equal to 80% and night time efficiency equal to 60%. Still, if component information becomes available a more detailed estimation can be made, using the following relations for day time and night time efficiency:

  Day time efficiency is the efficiency that results for all the energy flowing from the solar panels via the control system directly to the loads. It essentially is the product of the power conditioning (control) efficiency and the efficiency of the distribution system, i.e. the power lines and the switches:

$$\eta_d = \eta_{power\ conditioning} \cdot \eta_{distribution} \qquad [69]$$

  Night time efficiency is the efficiency that results for the energy flowing from the panel via the controls to the battery and via the controls again to the loads. It includes amongst other the battery efficiency (ratio between energy that flows from the battery to the energy that is inputted to the battery and the efficiency of the battery charge and discharge electronics that ensure proper (dis)charging of the battery:

$$\eta_e = \eta_{power\ conditioning} \cdot \eta_{charge\ el.} \cdot \eta_{battery} \cdot \eta_{discharge\ el.} \cdot \eta_{distribution} \qquad [70]$$

- The duration of the sunlit and eclipse periods depends on the orbit and usually follows from orbit analysis. For design purposes one usually tries to identify the most critical orbit (with shortest sunlit and longest eclipse period). This is usually the case when the satellite is orbiting in a plane perpendicular to the plane in which the planet is orbiting about the Sun parallel to the line connecting planet to Sun. For Earth orbiting satellites, appendix H gives typical maximum eclipse times over the orbit for various orbital altitudes.

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:

$562.5\ \text{W} \cdot (6.7 - 1.25)\ \text{h} + 750\ \text{W} \cdot 1.25\ \text{h} = 4003.1\ \text{Wh}$ = *(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:

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

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

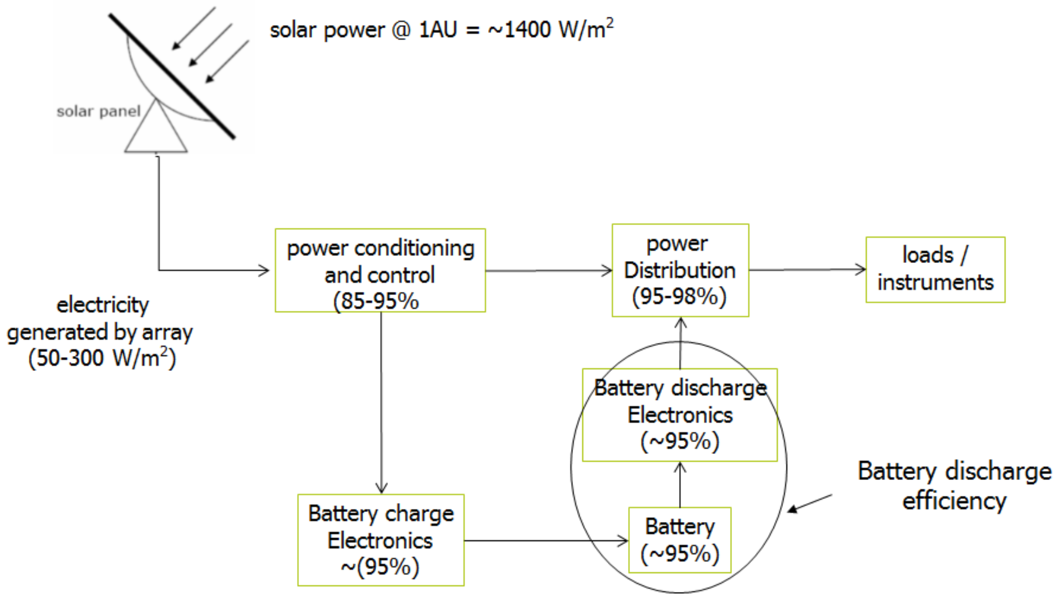


Figure 65: Example of PV system component efficiencies

Once we know the array power that we need to design for, we can start sizing the various elements that make up the PV system, but first we will discuss some typical design characteristics of PV systems. We will do this based on the data collected in Table 39.

Table 39: 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 39: 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. Note that power density is referred to in table as solar array efficiency.
- 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.

A first simple mass estimation of the PV system can be made assuming that the mass of a PV system increases about linearly with the power produced. It follows:

$$M_{EPS} = M_{PV} = \frac{P_{total}}{(P_{sp})_{system}} \qquad [71]$$

Here $(P_{sp})_{system}$ is EPS specific power usually expressed in W_e/kg. From Table 39 it follows EPS specific power is in range 2-40 W_e/kg. The broad range of values signifies there is not one value that encapsulates all designs. Rather is indicates that there are many designs out there that can lead to quite different results. As a side note it is mentioned that some variation in the specific power values given may be because the solar intensity onto the array varies with distance to the Sun, see chapter 3 and later in this chapter. Above relation is a very simple relation that does not tell anything on how the main elements of the EPS contribute to the system mass. Next table provides some mass details for some specific LEO spacecraft (data taken from an ESA presentation).

Table 40: Mass breakdown of PV power subsystem of specific spacecraft orbiting about Earth (ESA)

System	Telecom 2	Jason 1	DEMETER		Average	STD
PMD*	21.6%	31.0%	38.2%		30.3%	8.3%
Storage	44.6%	35.7%	23.5%		34.6%	10.6%
Power generation	33.8%	33.3%	38.2%		35.1%	2.7%

* PMD mass is taken equal to sum of power system + distribution

Table also gives average mass percentages for the three main elements making up a PV system. To generate the above table, it is assumed that the mass of the EPS is given as:

$$M_{EPS} = M_{PV} = M_{SA} + M_{BAT} + M_{PMD} \quad [72]$$

Data in table agrees nicely with the EPS mass breakdown for PV systems for Earth satellites as shown in Table 40. Still, the data also show that some spread does occur. This spread is signified by the standard deviation (STD). This spread may be due to for instance orbital altitude or the orientation of the orbital plane, etc. For instance, some satellites with long eclipses require a large and heavy battery system whereas other can almost do without a battery system.

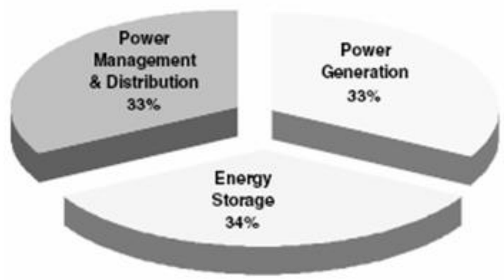


Figure 66: Typical photo-voltaic power subsystem mass breakdown (Earth satellites)

In the next few paragraphs a more detailed sizing of the EPS is discussed.

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:

$$M_{SA} = \frac{P_{SA}}{(P_{sp})_{SA}} \quad [73]$$

$$A_{SA} = \frac{P_{SA}}{(P_{\delta})_{SA}} \quad [74]$$

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), which both depend on the array type, see Table 41.

Table 41: 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

Notice that the above relations have already been used in chapter 3 to generate a first estimate of solar array size and mass. Note that array per power is inverse of the earlier introduced array power density. That here the values for power density are much larger than the one used in chapter 3 is because in chapter 3, low efficiency silicon was considered as baseline. Also the value used in chapter 3 holds for End Of Life (EOL), whereas values given here are for Begin Of Life (BOL). The difference is that during their life, solar arrays tend to degrade thereby producing less and less power. Typical values for life degradation in GEO are:

$$\begin{aligned}&\text{Si array: } 4\% \text{ per year}\\ &\text{GaAs array: } 1-15\% \text{ per year}\\ &\text{Multi-junction GaAs: } 0.5\% \text{ per year}\end{aligned} \qquad [75]$$

Actual values may be quite different and depend on the radiation environment, but above values allow for a conservative estimate. Once the degradation is known, the power at BOL can be determined based on known power at EOL and degradation rate. It follows:

$$\frac{dP}{dt} = -degradation\ factor\ P$$

After integration we obtain:

$$P(t) = P_{BOL} \cdot e^{-degradation\ factor \cdot t} \qquad [76]$$

Here t is time and degradation factor is the percentage degradation expressed as a fraction.

Since the degradation factor usually is small, we may also write:

$$P(t) = \sim P_{BOL} \cdot (1 - degradation\ factor)^t \qquad [77]$$

It must be clear from the foregoing that the power output of an array reduces in time. It should also be clear though that the power delivery at End Of Life (EOL) should still be sufficient to power the spacecraft. Hence in the beginning too much power may be produced and needs to be backed off.

Another effect that needs to be discussed is 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, on the power output of the array. Suppose that the solar array has a constant incidence angle, it follows:

$$(P_{SA})_\theta = (P_{SA})_{\theta=0} \cdot cos\theta \qquad [78]$$

So power reduces if the solar radiation is not perpendicular to the solar array. This is the reason why most spacecraft are equipped with a mechanism that allows rotating the array independently from the spacecraft, thereby ensuring the solar incidence angle is always close to maximum.

Finally, we consider again (see chapter 3) the change in intensity of the solar radiation arriving at the spacecraft with distance to the Sun. For this we use:

$$P_{sp,d} = \frac{P_{sp}}{d^2}\ [\mathrm{W_e/kg}] \qquad [79]$$

$$P_{\delta,d} = \frac{P_\delta}{d^2}\ [\mathrm{W/m^2}] \qquad [80]$$

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 less general form.

Specific power values like power density can be obtained from for instance the Table 41. It should be clear from the table that panels equipped with Si (silicon) cells have lower specific power then panels equipped with GaAs (Gallium Arsenide) cells. These differences are attributed to different cell efficiencies, but also in the different build-ups of solar cells. In this work though, we accept the data as is and we use them to compare different array designs.

Battery sizing

Batteries are used on PV equipped spacecraft to provide for the necessary power during eclipse conditions and or for peak power. Hereafter, as an example we will deal with the design for eclipse conditions. However, the method is quite similar for peak power conditions.

To allow for sizing of a rechargeable (secondary) battery (or batteries), 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 energy to be delivered by the battery is taken equal to the energy needed during eclipse, see earlier. However, when sizing the battery we should not only take into account the discharge efficiency, but also we need to consider that we may not be able to use the full capacity of the battery as the battery is used many times. In general, the total energy (usually expressed in Watt-hour, Wh) to be delivered by the battery follows from:

$$E_{BAT} = \frac{P_{BAT} \cdot t_{discharge}}{DOD \cdot \eta_{BAT}} \qquad [81]$$

Here η_{BAT} is energy transfer efficiency battery to loads and *DOD* is battery Depth of Discharge. Energy transfer efficiency typically is defined as:

$$\eta_{BAT} = \eta_{battery} \cdot \eta_{discharge\ electronics} \qquad [82]$$

Here $\eta_{battery}$ is the efficiency of the battery itself and $\eta_{discharge\ electronics}$ is the efficiency that control the discharge of the battery. It usually is in the range 90%. The battery *DOD* in relation [81] describes the degree to which a battery is emptied relative to its total capacity. It largely depends on the number of charge-discharge cycles a battery undergoes and the battery type. Some typical data are provided in Table 42. More detailed data can be obtained from [SMAD]. 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 42: Sample spacecraft battery configurations [Britton]

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:

$$M_{SBAT} = \frac{E_{BAT}}{(E_{sp})_{BAT}} \qquad [83]$$

$$V_{BAT} = \frac{E_{BAT}}{(E_{\delta})_{BAT}} \qquad [84]$$

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 43.

Table 43: Characteristics of space-grade secondary batteries [Surampudi]

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

Some further data on secondary batteries can be obtained from amongst others [Fortescue].

Sizing of Power management/control and distribution system

As a first approximation to determine the mass of a PMD system for PV systems, we can use a simple rule based on the result earlier shown in Table 40 and Figure 66.

$$M_{PMD} = a\, M_{EPS} \qquad [85]$$

Here M_{EPS} stands for the total mass of the EPS and a is a multiplicative factor indicting the fraction of the total EPS mass assigned to the Power Management/Control and Distribution (PMD/PCD) system. The underlying assumption being that the PMD mass is a fixed percentage of the total EPS system mass. For earth satellites we find an average value of the factor ‘a’ of 0.303 with an SSD of 0.083. For other vehicles like a deep space probe, the value of a may be quite different depending on for instance the usage of batteries and the number of battery cycles and hence we need to use similar systems flown in the past to determine an appropriate value of ‘a’ for the type of vehicle considered.

[Brown] provides for a different relationship (based on 3 data points) that relates PMD (or PCD) mass to EOL power generated:

$$M_{PMD} = 0.071\, P_{EPS,EOL} + 0.15\, kg \qquad [86]$$

Here M is in kg and P is in W. This relation has an R-squared value of 0.9937. The relation shows that the higher the power that is to be managed the heavier the PMD will be. Remarkable though is that mass is linked to EOL power and not BOL power or controlled power.

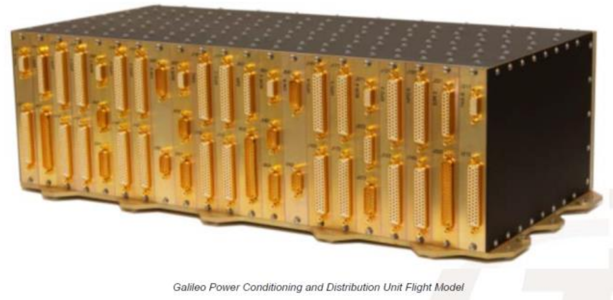
To obtain further details of interest for the dimensioning and sizing of the PMD subsystem of the EPS, it is useful to consider the main elements that make up the PMD subsystem. Main elements include:

- Power Conditioning Unit (PCU) which regulates the array power (array power regulator), the charging of the batteries (battery charge regulator), the discharging of the batteries (battery discharge regulator), see Figure 63. The PCU is a device that is computer controlled and hence also has provisions that allow it to be commanded and to provide status (health) information to the On-Board Computer (see later chapter on Command & Data Handling).
- Power Distribution Unit (PDU), which allows for distributing power to the payloads, heaters, powered propulsion components, like valves, pyrotechnics for separation and deployment, etc.
- Power cabling, which transports the power from the source to the PCU and PCD and to the various loads. Cables are to be shielded to reduce ElectroMagnetic Interference (EMI).

Sometimes the PCU and PDU are combined in a single Power Conditioning and Distribution Unit (PCDU). Figure 67 shows examples of the mentioned PMD components.

Thales Next Generation PCU

Therma small GEO PDU



Galileo S/C Therma PCDU

S/C power cables

Figure 67: PMD components

For each of these components data can be collected and sizing rules determined. As a typical example, some data have been collected in Table 44. From these data, we learn that mass and size of PCU/PDU/PCDU relates strongly to the maximum controlled or distributed power according to:

$$M_i = P_i / P_{sp,i} \qquad [87]$$

$$V_i = \frac{P_i}{P_{\rho,i}} \qquad [88]$$

Here M is mass and V is volume, P is maximum controlled or distributed power, subscript i refers to either PCU, PDU or PCDU. P_{sp} is specific power and P_ρ is power density. It is noticed that controlled or distributed power may be similar to array power, but may also be quite different as in peak power conditions, the amount of power may rise above the power level provided by the array as also the battery may deliver power. Using the tabulated data we find for the average specific power:

$$P_{sp,PCU} = 296\ \frac{W}{kg}\left(SSD = 76\,\frac{W}{kg}\right) \quad [89]$$

$$P_{sp,PDU} = 71\,\frac{W}{kg}\left(SSD = 14\,\frac{W}{kg}\right) \quad [90]$$

$$P_{sp,PCDU} = 145\ \frac{W}{kg}\left(SSD = 123\,\frac{W}{kg}\right) \quad [91]$$

From these data follows that the PDU is relatively heavier than the PCU. The PCDU is somewhere in between as it combines the two in a single device.

In above relations, also the sample standard deviation is given. For PCU this standard deviation is quite high. This may be in part because data has not been interpreted correctly, but may also be related to differences in PCU design. For instance, different types of array regulators may be used and sometimes there is no battery charge or discharge regulator. In the work of [Brown] a similar relation for the mass of the PCU can be found showing an average specific power of 0.0045 kg/kW. This nicely falls in the range as indicated for the PCU specific power in the relation given in the current work. It may be argued that the average specific power as given by Brown holds for larger PCU and hence may suggest that specific mass depends on power level. However, this is considered out of scope and is left for the reader to research for him/herself.

Table 44: PMD component design characteristics [various sources, see internet sources in reference list]

Type	**Provider**	**Name of device**	**Mass [kg]**	**Maximum controlled/distributed power [kW]**	**Dimensions [mm]**
PCU	Terma	PCU	8.3	2.4	267x233x158
PCU	Alcatel	MHPCU	54.5	18	659x620x150
PCU		MHPCU	44	12	563x620x150
PCU	Terma	Mars Express PCU	8.2	1.5	267x238x158
PCU		XMM MRU	24	2.1	540x520x180
PCU	Thales	PCU	<51	21.6	
PCU	Thales	PCU	<55	18	
PCU		Messenger PSE	6.5	1.6	286x196x142
PDU		XMM SVM (service module)	16	1.4	455x270x200
PDU		XMM FPA (payload module)	11	0.84	312x270x200
PDU		XMM Pyro Regulator Unit	5.5		250x250x200
PDU		XMM Mirror thermal control unit	15.8	1.005	402x274x220
PDU	Terma	PDU	13.2	1.000	425x285x160
PDU		New Horizons PDU			
PDU	APL	Messenger PDU	12.5	0.64	234x226x239
PCDU	Terma	Galileo PCDU	18.2	2.6	508x235x156
PCDU	Surrey	PCDU	4.4	0.2	670x305x39
PCDU	Thales	PCDU	4.5	330 W	
PCDU	Thales	ARSAT PCDU	16.6	4.2	332x345x193
PCDU	Thales	ECRS PCDU	27.4	3.7	536x345x193
PCDU	AAC	Nano PCDU		0.045	
PCDU	ClydeS	3G EPS PCDU	0.086	0.034	95x90x15
PCDU	Magellan	PCDU	27	2	370x356x229
PCDU	Magellan	PCDU	12	0.5	370x180x229

* Magellan is Magellan Aerospace ; AAC is AAC Microtec, ClydeS is ClydeSpace

Like for the specific power, we can also determine average power density. It follows the following relations:

$$P_{\rho,PCU} = 192 \frac{W}{liter} (SSD = 88 \frac{W}{liter}) \quad [92]$$

$$P_{\rho,PDU} = 50 \frac{W}{liter} (SSD = 6 \frac{W}{liter}) \quad [93]$$

$$P_{\rho,PCDU} = 92 \frac{W}{liter} (SSD = 64 \frac{W}{liter}) \quad [94]$$

Results show that the PCU is quite dense in relation to the PDU. Again the PCDU is somewhere in the middle.

Another interesting feat that can be taken from the table as shown for the XMM spacecraft is that sometimes a separate PCU or PDU is used for the payloads and the service module.

Not included in the table are data representing the efficiency of the various devices and their own power consumption. The latter sometimes is included in the device efficiency. Without showing data, it is mentioned that for current devices efficiency is in the range 90% and higher. For instance the direct transfer of solar power to the loads is 97.8% efficient for the Galileo PCDU and 95.2% efficient in case the power comes from the battery. The electrical energy lost ends up as thermal energy that needs to be considered for the design of the thermal control system.
Also not shown in the table are data relating to the reliability of these devices. For this usually the failure rate is used expressed in FITs. FIT stands for Failure In Time rate and is the amount of devices that would fail during one billion hours of operation. 1 FIT = 10^{-9} failure/hour. For PCU, PDU and PCDU typical failure rates are in range 250-500 FITs.

Finally we turn to the power cables. These together with the signal data lines (see section on Command and Data Handling) are usually referred to as the cable harness. According to [Brown] the harness mass is in range 3-10% of on-orbit dry mass of the spacecraft. No further information on the cable harness is readily available.

Radio-isotope Thermoelectric Generators (RTGs)
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 68. 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. Hence, they do not need a secondary power source as required by a solar array system.

Apollo fuel cell power-plant (courtesy NASA)

Ulysses (courtesy NASA/ESA)

Figure 68: Alternative spacecraft energy sources

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.

Sizing of RTGs is usually on peak power to be delivered as usually no storage system is included in the system. Another factor that should be taken into account is that because of natural decay the amount of radioactive isotope in the reactor reduces and hence the power delivery reduces with increasing operation time. For power P at time t follows:

$$P = P_o \cdot e^{\left(\frac{-0.693}{\tau_{1/2}} t\right)} \qquad [95]$$

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_o). From [Fortescue] the following data for some isotope fuels is obtained:

Table 45: Characteristic data of some radio-isotope fuels

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.

Note that cost figures in the table are given per Watt of thermal power. In practice, only a few percent of thermal power produced is converted to electric power. The remainder is converted to heat, which is to be radiated away. This is why the Ulysses RTG is painted black and has cooling fins.

For a derivation of the RTG power relation (equation [95]), it is considered that the power produced decreases at a rate proportional to its value (linear decay model):

$$dP/dt = -\alpha P \text{ => Separation of variables => } \frac{dP}{P} = -\alpha dt$$

Integration gives $ln(P/P_o) = -\alpha t$ with P_o is available power at time t = 0. After rearranging we obtain $P = P_o \cdot e^{(-\alpha t)}$. Next, using the information that at half-life the available power has reduced with a factor 0.5, it follows: $P = 1/2\, P_o \cdot e^{(-\alpha \tau_{0.5})}$. This gives $-\alpha\tau_{0.5} = \ln(1/2) = -0.693$ and we obtain: $\alpha = 0.693/\tau_{0.5}$. Substitution of the relation for the half time in the relation for P ($P = P_o \cdot e^{(-\alpha t)}$) results in [95].

Once the BOL power is determined, the mass of an RTG system can be estimated using:

$$M_{RTG} = \frac{P_o}{(P_{sp})_{RTG}} \qquad [96]$$

Here $(P_{sp})_{RTG}$ = RTG specific power (typically in units of W_e/kg) and P_o is initial RTG power (power at Begin Of Life, BOL). Typical values of RTG specific power can be obtained from literature. To this mass, we should then still add the mass of an accompanying power management and distribution system. As a starting point, we may use the relations as discussed earlier for the PMD of photovoltaic systems. However, we may need to look for more specific data as the PCU for a radio-isotope system does not require an array regulator. Also a battery charge and discharge regulator is not always needed.

Fuel cell systems

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 a fuel cell system are peak power level and total energy need. Both should be known at the start of the design for fuel cell systems. As one aims to keep the required power output of a fuel cell steady, this means that excess power must be shunted away in the form of heat. This should be taken into account when designing the thermal system of a spacecraft.

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.

$$M_{fuel\ cell\ system} = \left(M_{fuel\ cell\ system}\right)_{dry} + M_{reactants} + M_{storage} \qquad [97]$$

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 $(P_{sp})_{fc}$ according to:

$$\left(M_{fuel\ cell\ system}\right)_{dry} = \frac{P}{(P_{sp})_{fc}} \qquad [98]$$

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}):

$$M_{reactants} = E \cdot C_{rate} \qquad [99]$$

From [Fortescue] Tables 10.3 and 10.5, it is 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. Typical fuel cell mass density (based on Space Shuttle fuel cell system) is about 750 kg/m^3. Next to the system itself, we must also to take into account the reactants needed to provide for the required energy. For instance, for a hydrogen-oxygen fuel cell hydrogen-oxygen consumption (mass ratio 1 kg of hydrogen reacts with 8 kg of hydrogen) rate 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). It follows 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 and includes about 186.5 kg of hydrogen and 1493.5 kg of oxygen.

How to determine the mass and volume of the reactant storage system is dealt with in the lecture material on launch vehicle design, but is highly linked to the volume of the cryogenic oxygen and hydrogen. Mass density of these species can be obtained from the propulsion section, see later in this work.

Next to the fuel cell system itself, we also should consider an accompanying power management and distribution system. As a starting point for the mass and size estimation of PMD components, we may use the relations as discussed earlier for the PMD of photovoltaic systems. However, as fuel cell systems may differ from PV systems, it is advised to verify whether the earlier introduced relations for PV system power conditioning and distribution units can also be applied for the PMD of fuel cell systems. For further study (not as part of this work) consider consulting [Burke].

Batteries as the primary source of energy (primary batteries)
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):

$$M_{bat} = \frac{E}{E_{sp}} \;\; ; \;\; V_{bat} = E / E_\delta \qquad [100]$$

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

Table 46: 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

A primary battery system consists next to the primary batteries themselves of also a power conditioning unit and a power distribution unit, be it that the PCU may differ from that of a PV system (no need for an array nor a battery charge regulator. Also the active life of primary batteries is limited, which may affect the design of the PCU and PDU for a primary battery system. Still, for a first sizing, it may be considered to use the relations earlier given in relation to PV systems. Energy transfer efficiency may be taken equal to 95% and higher as only a discharge regulator is needed.

Configuration issues
Typical issues that need to be considered include (see also Figure 69):

- 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. Sometimes a second rotational axis is needed to keep the panels truly perpendicular to the solar radiation. Still such mechanisms are quite expensive and some designers decide to not include such devices at the cost of a slightly larger solar array.
- 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. In case of a spinning satellite though, not all panels are 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.

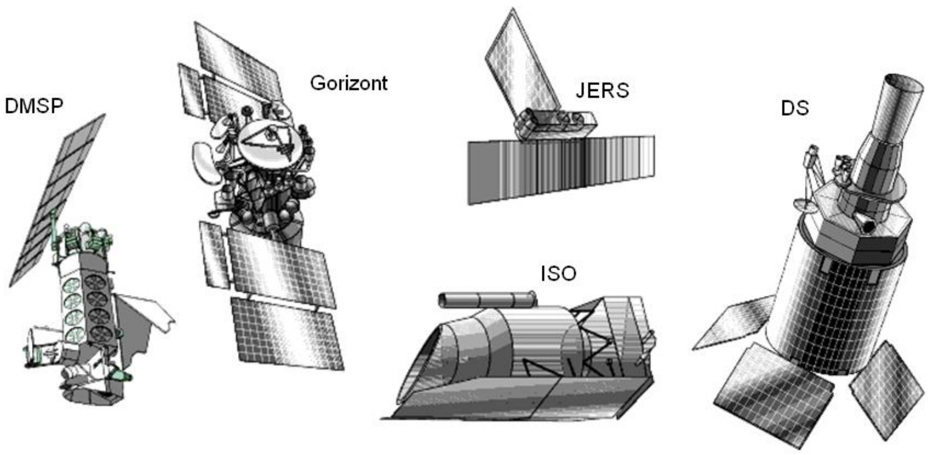


Figure 69: Typical configurations for solar cells

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
1st 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.
2nd 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.
3rd 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.
4th option is to use body-fixed panels on a spinning satellite (compare Meteosat series satellites). We would essentially need 10.9 m^2, but since the panels are on a spinning body, 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.

Problems

1. Give an account of:
   - the main functions of the electrical power generation system (EPS) and discuss the main requirements that should be considered when designing an EPS.
   - the different types of electrical power generation system currently available for use on spacecraft. Indicate for each the relative merits and limitations and identify space classes of space missions for which they are useful.
   - the main parts that make up the overall spacecraft power system for the case of a solar cell powered satellite, thereby discussing the need for each part and its function.

2. A typical external energy source is the Sun with a practically unlimited amount of energy available. However, it is not the amount of energy available from the Sun that is important, but the available power (energy per unit of time) and more specific the power per unit of area (solar flux). Calculate the value of the solar flux at Earth, Mercury, Mars, Jupiter and Pluto given a total power output of the Sun of 3.8 x 10^{26} W and generate a graph showing the solar flux versus the distance measure in Astronomical Units (AU).

3. You are designing the EPS for a spacecraft. The EPS is required to deliver an average 100 W of power for a period of 1 week. Which type of EPS do you consider the most likely candidate for this mission and why?

   Answer: A fuel cell system is the most likely EPS candidate. The reason is that this system has an internal energy source and hence can work independently from the environment, while mass remains within reasonable amounts (lower than for a battery system). An RTG might also be a possibility, but this adds a radiation problem. Also fuel for RTGs is not abundant, so if an alternative is available, it is better to use the alternative.

4. You are designing a deep space probe. This space probe is to carry a payload requiring 50 W of power. Using data from historical spacecraft you have determined the following regression relationship between on board generated power and payload power, both in Watt:

$$P_{generated} = 209.2 \ln(P_{payload}) - 491.8 \text{ W}$$

   This relation has an RSE of 23%. Determine for this vehicle the range that should be taken into account for the on-board generated power given that this range should encompass 95% of all possible outcomes.

   Answer:176 - 477 W (nominal power at 50 W = 326.6 W; RSE = 0.23 x 326.6 W = 75.1 W)

5. A spacecraft in orbit about Mars requires a stable 200 W for operations. You consider using a PV array for power provision. Given are:
   - Orbital period: 3 hour
   - Eclipse period: 58 min
   - Path efficiency during daytime: 80%
   - Path efficiency during eclipse: 60%

   Determine power that needs to be provided for by the solar array

   Answer: 408 W

6. The end-of-mission life capability of a spacecraft power system is 200 watts. Your instrument is expected to use 50 watts, including 25% contingency. You are allotted 75 watts by the satellite provider.
   a) What is the instrument's contingency
   b) What is the power margin in watts and as a percentage (%)?

   Answers: a) 12.5 W, b) Power margin is 75 W-50 W = 25 W; (75 W-50 W)/75W x 100% = 33.33%

7. A cubical S/C in LEO is equipped with a PV array to provide 400 W. Given is a linear dimension of the vehicle of 2 m and a power density of the array of 100 W/m^2.
   A. Consider the array area needed in case we have a 3-axis stabilized craft that allows keeping one side of the craft perpendicular to the incoming solar radiation
   B. Same question, but now in case the craft is spin stabilized with spin axis perpendicular to the incoming solar radiation

   Answers: A) 4 m^2; B) 12.6 m^2

8. A solar array in space is producing 3.5 kW at Begin Of Life (BOL). Over its life this power reduces because of degradation due to radiation. Given is that the power output of the array reduces with 3%/year and that the life of the array is 15.6 year.

   Array power delivered by the solar array of the spacecraft at end of Life (EOL) is …?

Answer: 2.19 kW

9. A battery system is selected to provide 620 W of power for a period of 35 minutes. Due to the cyclic use of the battery, the battery is only capable of using 12% of its capacity. In addition, because of high discharge rates the battery is only 80% efficient. Given that the specific energy of the battery selected is 35 Wh/kg and the specific energy density 100 Wh/l, calculate for the battery system:
   A. Battery storage capacity (in Wh)
   B. Total battery mass, and
   C. Total battery volume

   Answer: A) 3800 Wh, B) 107 kg, C) 381 liter.

10. You have designed an RTG using an isotope fuel with a half-life of 45 days to provide 1000 W at start of mission. End of mission is after 1 year. What is for this RTG the **available power** at end of mission?

    A. 500 W
    B. 123.3 W
    C. 3.62 W
    D. 12.33 W

    Answer: C is correct answer.

11. The following RTG data are taken from [Wikipedia]:

Name & Model	Used On (# of RTGs per User)	Maximum output		Radio-isotope	Max fuel used (kg)	Mass (kg)
		Electrical (W)	Heat (W)			
ASRG*	prototype design (not launched), Discovery Program	~140 (2x70)	~500	^{238}Pu	~1	~34
MMRTG	MSL/Curiosity rover	~110	~2000	^{238}Pu	~4	<45
GPHS-RTG	Cassini (3), New Horizons (1), Galileo (2), Ulysses (1)	300	4400	^{238}Pu	7.8	55.9–57.8[31]
MHW-RTG	LES-8/9, Voyager 1 (3), Voyager 2 (3)	160[31]	2400[32]	^{238}Pu	~4.5	37.7[31]
SNAP-3B	Transit-4A (1)	2.7[31]	52.5	^{238}Pu	?	2.1[31]
SNAP-9A	Transit 5BN1/2 (1)	25[31]	525[32]	^{238}Pu	~1	12.3[31]
SNAP-19	Nimbus-3 (2), Pioneer 10 (4), Pioneer 11 (4)	40.3[31]	525	^{238}Pu	~1	13.6[31]
modified SNAP-19	Viking 1 (2), Viking 2 (2)	42.7[31]	525	^{238}Pu	~1	15.2[31]
SNAP-27	Apollo 12–17 ALSEP (1)	73	1,480	^{238}Pu[33]	3.8	20
Buk (BES-5)**	US-As (1)	3000	100,000	^{235}U	30	~1000
SNAP-10A***	SNAP-10A (1)	600[34]	30,000	Enriched uranium		431

    Average specific power of RTG system is ... ?

    Answer: Average specific power value is 3.04 W/kg with an SSD of 1.26 W/kg.

More problems for exercising upon are available via the course web pages (look for Mobius exercises) or in the tutorial material.

4.4 Propulsion

The propulsion system (PROPS) 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 while on its way 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, like gravity and drag, 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 for the propulsion system concern or are related to:

1. Δv-budget;
2. Maximum and minimum thrust level;
3. Maximum flight duration;
4. Number of thrust periods.

Hereafter these requirements are discussed in some detail.

<u>*Ad 1: Δv* budget</u>

A Δv budget generally lists the required Δv per manoeuvre, see our earlier discussion. Once it is known which element performs what manoeuvre and how, we can turn our attention to the design of the propulsion system for those elements (kick stage, orbiter, lander, ascender, etc.) requiring a propulsion system. In the conceptual design stages this may need to be handled in an iterative way as sometimes the manoeuvres considered are not definite yet. In later stages of the design, the Δv budget may be much more stable and hence also the values of Δv to be delivered by the propulsion system of a specific spacecraft. This is usually determined by orbit analysts together with S/C engineers.

<u>Ad 2: Maximum and minimum thrust level</u>
For some missions, it should be considered that a high thrust-to-weight (*T/W*) ratio will limit gravity loss and or travel duration. However, if thrust is chosen too high, acceleration levels may become too high to sustain. For instance, Figure 2.4 from [Fortescue] provides typical tolerance levels to sustained acceleration levels for astronauts. Typical acceleration levels for manned space launch vehicles are of the order of 3-4g_o, for unmanned space launch vehicles they can be as high as 6-8g_o. For most spacecraft maximum acceleration levels are much more limited than for launch vehicles with a *T/W* ratio in the range of 0.1 and less. This is to limit the vehicle's structural mass. These low values of T/W are possible since in space thrust generally is in a direction perpendicular to gravity instead of opposing gravity as for launch vehicles especially during the initial phases of the ascent.

For some cases it may be more beneficial to select a low thrust propulsion option. In that case, even though mission characteristic velocity increases, still a much higher specific impulse level may be feasible. This in the end may lead to a reduced propulsion system mass leading to either a reduced spacecraft mass or an increased payload mass. This will be discussed later in more detail.

Another issue that must be considered that propulsion can also be used to compensate for disturbing forces. In that case the thrust level needs to be chosen high enough to allow for compensating the disturbing force, but not too high as in that case a relatively heavy system will result. To allow for selecting the proper thrust level, the magnitude (and direction of the disturbing forces need to be known. In the next two sections, we will discuss (aerodynamic) drag and the force resulting from solar radiation[22]. 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 e.g. attitude and or with changes in air density. However, when designing for the worst, this may lead to a relatively heavy system.

a) 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 generally is calculated using:

$$F_a = \frac{1}{2} \cdot \rho \cdot v^2 \cdot S \cdot C_D \qquad [101]$$

- F_a is aerodynamic drag
- ρ is the atmospheric density (depends on the altitude; for values for Earth see appendix H or I. for other planets, see appendix I)
- v is orbital velocity (for typical values of circular velocity versus altitude for Earth, see appendix H)
- S is frontal projected area
- C_D is aerodynamic drag coefficient; for Free Molecular Flow[23] its value ranges from 2 to 4

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

[22] In 1960 the orbit of Echo-1 S/C (a polyester ball of about 30 m diameter) was found to be very irregular. When investigating the cause, it was found to be due to solar radiation. From this came the realisation that light particles reflected by a surface will exert a force on this surface.
[23] **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 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.

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).

b) Solar radiation
The force exerted on the spacecraft by solar radiation can be calculated using:

$$F_s = (1+\rho) \cdot P_s \cdot S \qquad [102]$$

$$P_s = \frac{J_s}{c} \qquad [103]$$

- 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 ($c = 3 \times 10^8$ 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).

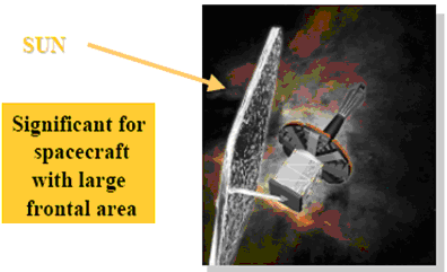


Figure 70: Echo 1 S/C (left, courtesy NASA) and S/C experiencing solar pressure force induced by solar radiation (right)

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.

Ad 3: Maximum flight duration
Maximum flight duration is mainly limited by mission duration. Critical is that total thrust time does not exceed the mission duration.

Ad 4; Number of thrust periods
For most space missions, the propulsion system does not have to be active all the time, but only at certain periods of time. Most interplanetary probes do perform multiple maneuvers for which the propulsion system needs to be active. The number of maneuvers typically follows from orbit analysis. In case the propulsion system is also used for attitude control, many thousands of short burn periods may be needed over the lift of the spacecraft as to ensure for proper attitude control. For instance on the Swarm spacecraft, the thrusters are certified for 1.5 million duty cycles with pulse times as short as 100 microseconds (from http://spaceflight101.com/swarm/swarm-spacecraft-overview/ , retrieved 23 October 2015).

Other requirements
Next to the aforementioned key requirements also requirements with respect to mass, size, cost, etc. may apply. These generally stem from budgets as derived by the spacecraft systems engineer for the complete spacecraft, see earlier chapter on spacecraft design.

Propulsion fundamentals or how does rocket propulsion 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.

The magnitude of the thrust force 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 the (rocket) exhaust velocity:

$$F_T = m \cdot w \qquad [104]$$

Multiplying the left hand side with time essentially gives the impulse delivered to the system. This should equal the change in momentum of the system. Since propellant mass flow rate can be seen as a change in vehicle mass over an infinitely small time step, $m = dM/dt$, it follows:

$$F_T dt = M \cdot dv \quad or \quad M \cdot dv = dM \cdot w \qquad [105]$$

Integration of the relation on the right leads to the rocket equation as earlier defined in Eq.[8]. This relation relates the Δv to be obtained to the effective exhaust velocity and the mass ratio of the vehicle. As such it allows for determining the propellant mass needed. Considering that propellant mass M_p form some maneuver is given by the difference between initial mass M_o and final mass M_f at end of burn:

$$M_p = M_o - M_f \qquad [106]$$

We can derive the following equations for propellant mass for the maneuver at hand:

$$\begin{aligned} M_p &= \left(1 - e^{-\Delta v / w}\right) M_o \\ M_p &= \left(e^{\Delta v / w} - 1\right) M_f \end{aligned} \qquad [107]$$

These equations relate propellant mass to initial and final mass, respectively and stress the importance of a high exhaust velocity for minimizing propellant mass. In case of multiple burns, of course the final mass of the preceding burn becomes the initial mass for the next burn and so on. Using the above relations allows for determining the propellant mass for each of the maneuvers needed.

Once propellant mass is known (from the rocket equation) and a thrust value set (for some exhaust velocity), the burn or operation time of the propulsion system can be determined. Assuming a constant mass flow rate gives:

$$t_b = \frac{M_p}{\dot{m}} \qquad [108]$$

In practice, mass flow rate 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 "Launch Vehicle Design and Sizing"). Depending on the design, it is easily possible to maintain a constant mass flow rate using controls. For some special designs, a decreasing mass flow rate with increasing operation time results.

Example: Exhaust velocity
A rocket is producing a thrust of 30 kN over a 100 s time span. During this time span the mass of the rocket decreases linearly with 900 kg from 1000 kg to 100 kg. 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.

An 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:

$$I_{sp} = \frac{\int F_T \cdot dt}{M_p \cdot g_o} \xrightarrow{\text{for } F_T \text{ = constant in time}} I_{sp} = \frac{F_T \cdot t_b}{M_p \cdot g_o} \qquad [109]$$

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.[108] and Eq.[109]:

$$I_{sp} = \frac{\dot{m} \cdot w \cdot t_b}{\dot{m} \cdot t_b \cdot g_o} = \frac{w}{g_o} \qquad [110]$$

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. Figure 71 gives typical ranges of I_{sp} for a number of different propulsion systems.

Propulsion requires a certain amount of power to be available. How much power is needed can be determined from the jet power P_j:

$$P_j = \tfrac{1}{2} F_T \cdot w = \tfrac{1}{2} \dot{m} \cdot w^2 \qquad [111]$$

Using:

$$P = \frac{P_j}{\eta_T} \qquad [112]$$

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.

Rocket options and their main characteristics
From literature, we learn that various options exist to generate a propulsive force. Two basic methods are distinguished being:

- 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, various non-rocket system concepts exist that can be considered for use on spacecraft. These include solar sailing, tether propulsion, and magnetic sails for in-space propulsion.

Since there are almost no practical applications of non-rocket propulsion systems, we will focus on discussing rocket systems only.

Literature shows that various types of rocket systems exist, including:

- Thermal rocket propulsion, wherein a high pressure gas is accelerated to a high velocity by allowing the gas to expand in a nozzle (generally shaped as a convergent-divergent flow channel; in the convergent the flow accelerates to reach sonic velocity in the nozzle throat after which the flow is further accelerated in the divergent section). Depending on the type of energy source used, we distinguish:
    - Cold gas propulsion, wherein the energy needed stems from a cold gas stored under high pressure.
    - Thermo-chemical propulsion: Chemical energy is used to generate a hot high pressure gas 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): The energy needed for heating the propellant is obtained 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 (a force resulting from the interaction between a magnetic field and an electric current) 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 to accelerate a plasma propellant.

Notice that since ion propulsion, plasma propulsion, arcjet and resistojet all require electric power for their operation they are sometimes referred to as electric propulsion. For more details on the various systems, see later in this section.

Some typical performances of the various systems available today as well as for some futuristic systems (nuclear, laser and antimatter propulsion) are shown in Figure 71. First and foremost, this figure gives typical ranges of I_{sp} for the various rocket systems. The corresponding rocket exhaust velocities are:

- Cold gas propulsion (not shown in figure): 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 30 km/s
- Plasma systems: up to 20 km/s

It shows that chemical systems have low exhaust velocity (or specific impulse) and electric systems (ion and plasma) high exhaust velocity.

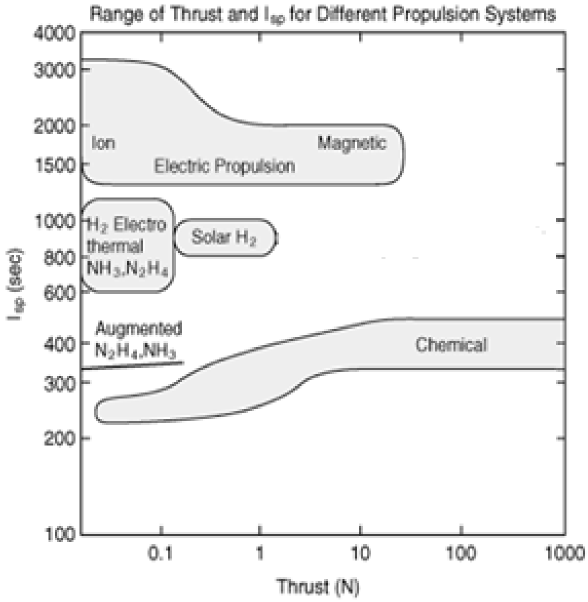


Figure 71: Range of thrust and I_{sp} for different propulsion systems

From Figure 71 we also learn that thermonuclear and chemical systems do cover the complete thrust range from high thrust to low thrust. This makes these systems applicable for both launch vehicle propulsion and spacecraft propulsion. Other types of propulsion are limited to low thrust applications only.

Next table shows typical chemical propellants used for spacecraft propulsion.

Table 47: Characteristic data of S/C chemical propulsion systems

Propellant Type	**Vacuum Isp (steady state)**	**Mixture Ratio (by mass)**	**Thrust level**	**Restart capability**
	[s]	**[-]**	**[N]**	**[cycles]**
Solid	285-303	NA	7,000-200,000	1
N_2 cold gas	57-65	NA	0.001-266	> 500,000
Hydrazine monopropellant	206-235	NA	0.02-572	>750,000
MON/MMH bipropellant (bi)	285-352	1.5-2.1	4-4,000	>50,000
NTO/MMH (bi)	220-317	1.2-2.05	4.5-27,000	>300,000
NTO/Hydrazine (bi)	277-290	1.3-1.65	4.5-100	>300,000

Hydrazine = N_2H_4, NTO is nitrogen tetroxide = N_2O_4. MMH is a hydrazine derivative and MON is Mixed Oxides of Nitrogen - Nitric oxide.

Solid propellants have a vacuum specific impulse in range 285-303 s. They can easily attain high thrust (thereby allowing for reducing gravity loss), but have only a single start capability. Cold gas allows for many restarts and very fine thrust levels, but specific impulse is low. All remaining propellants are liquid under ordinary conditions and self-igniting. This explains why for all these propellants a large number of thrust cycles can be achieved and why these engines are suitable for RCS applications. For bipropellants, the mass mixture ratio indicates how much kg of oxidizer is needed to burn 1 kg of fuel. Note: Oxidizers are mentioned first.

Table 48 gives characteristic data for specific ion thrusters. Thrust and specific impulse values depend on the settings of the thruster. Table shows thrust values in range 0.001 mN to 5000 mN (5 N) and specific impulse values in range 3,000-19,000 s, depending on the propellant used and to a lesser extend the thruster settings (thrust level). It is clear from the table that even to produce a low thrust already high power values are needed. For some engines the total power need already exceeds the power level available for the International Space Station (~100 kW).

Table 48: Characteristics of specific ion thrusters (thrust and specific impulse values hold for vacuum conditions)

Engine	**Propellant**	**Required power**	**Specific impulse**	**Thrust**
		(kW)	(s)	(mN)
NSTAR	Xenon	2.3	3,300 to 1,700	92 max.
NEXT	Xenon	6.9	4,300	236 max
HiPEP	Xenon	20–50	6,000–9,000	460–670
Hall effect	Xenon	25	3,250	950
DS4G	Xenon	250	19,300	2,500 max.
FEEP	Liquid Caesium	6×10^{-5}–0.06	6,000–10,000	0.001–1
VASIMR	Argon	200	3,000–12,000	~5,000
KLIMT	Krypton	0.5		

From the table, we learn that the required (input) power for the NSTAR engine is 2.3 kW. From the given thrust and specific impulse follows a beam or jet power of 1.49 kW (= 0.5 x 92 mN x 3300 s x 9.81 m/s^2); this indicates a thrust efficiency of ~65%. Typical thrust efficiencies reported in literature [SMAD] are in range 10-30% for plasma thrusters, 50-60% for ion thrusters and up to 80-90% for thermal rockets. For chemical systems, thrust efficiency is not important as the energy is contained within the propellants themselves.

Spacecraft propulsion system

The propulsion system of a spacecraft generally consists of a primary and a secondary system.
The primary system provides for the means required to perform large velocity changes/maneuvers. It generally includes of one or more large thrusters, a system to store the propellants and a system feeding the propellants to the thrusters. The primary system can be integrated (integral propulsion system) with the spacecraft, but may also form a separate propulsive stage, i.e. propulsive module or kick stage that can be dropped once the propellant is spend. 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. A disadvantage is that in that case also a structural system is needed to hold the elements of the propulsion system. Most primary systems use high performance (high exhaust velocity) propulsion systems as this allows for limiting propellant mass.
The secondary propulsion system, referred to more commonly as the reaction control system (RCS), provides the propulsive means for small maneuvers, like drag compensation, and/or attitude control. An RCS system typically includes many small thrusters that can provide small amounts of thrust in any desired direction or combination of directions. RCS can also be used to generate torques on the spacecraft to control S/C pointing (by controlling S/C 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 [24] of satellites. Most RCS systems use either hypergolic (self-ignitable) propellants or a monopropellant that allows for many start-stop cycles of the thrusters.
Sometimes the primary and secondary system are integrated to form a single system, meaning they use the same propellants and share the propellant storage system.
Available options are of course chemical (and cold gas) systems and systems that require a separate power source. Both options will be discussed in some detail below.

1. Chemical and cold gas rocket propulsion system elements and their configuration

Chemical and cold gas rocket propulsion systems form the standard systems in use today. Chemical propellants are mostly used in both primary and secondary systems, whereas cold gas systems are mainly used for secondary propulsion system applications. Chemical systems can be further subdivided in solid propellant systems and liquid monopropellant and bipropellant systems. Of these latter category,

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

liquid monopropellant systems use a single propellant that has both fuel and oxidizer qualities, whereas bipropellant systems use a separate fuel and oxidizer. Solid propellants used in spaceflight essentially consist of a separate fuel and oxidizer stored in a solid binder. Figure 72 shows a Star 30B solid propellant kick motor (left figure) as well as a solid propelled Trans Lunar Injection (TLI) stage (in back) with solid rocket motor on foreground ready for integration with the stage. A **kick motor** (KM) generally refers to a solid propellant rocket motor that is integrated on a S/C to perform a (large) maneuver, whereas a kick stage is a separable stage (usually with its own navigation system, power system, and RCS), which in the case of the TLI stage is propelled by a solid rocket motor.

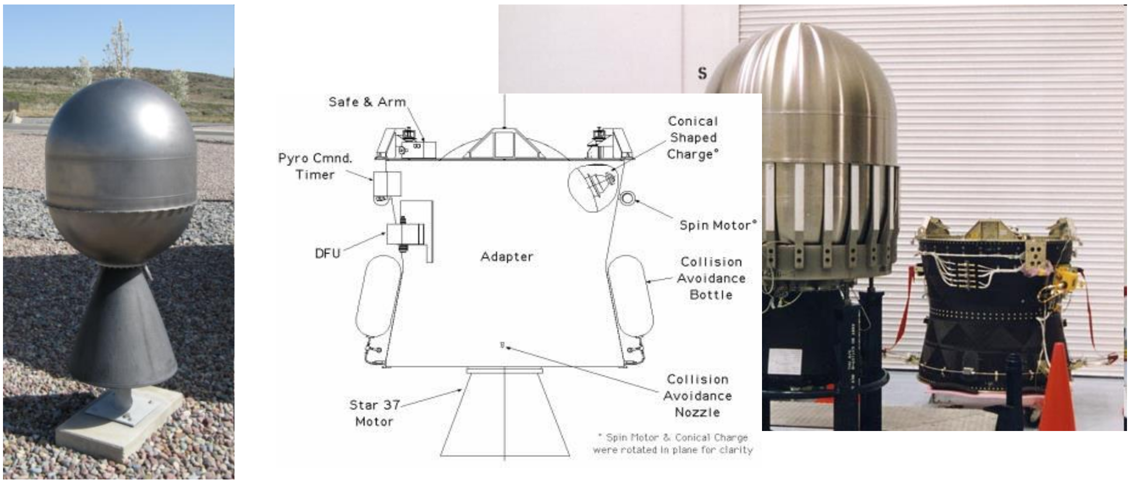


Figure 72: Solid rocket motor (left, courtesy ATK), Trans Lunar Injection stage (right) with solid rocket motor (not installed) on foreground and schematic of Trans Lunar Injection stage (middle) [Andolz]

The TLI Stage was designed to inject the Lunar Prospector Spacecraft into a trans-lunar trajectory from a low earth parking orbit. The following information has been taken from [Andolz]. The stage consists of a STAR 37 FM solid rocket motor, two spin rockets, a graphite epoxy inter-stage structural adapter, spacecraft–TLI stage separation system, a cold gas collision avoidance system (CAS), Command Timer, and a flight termination system. The Star 37FM motor provides enough total impulse to increase Lunar Prospector's velocity by 3143 m/s. The TLI Stage is approximately 170.8 cm long and 128.3 cm in diameter. The total fueled mass of the TLI Stage is 1216.5 kg. This mass value includes 1,010.7 kg of TP-H-3340 solid rocket propellant in the Star 37FM motor (empty mass ~82 kg), 1.1 kg of expendable Arcite 377A propellant in the Spin Motors and 3.6 kg of gaseous N_2 in the CAS bottles pressurized to 24.13 MPa. Stage structure mass is 56 kg.

Next figure shows a typical combined (primary and secondary) liquid propellant propulsion system that makes an integral part of a spacecraft. The system shown is used on large telecommunications satellites destined for GEO and has been developed by Thales and was reported in 2012 to have flow on 29 different spacecraft [Thales].

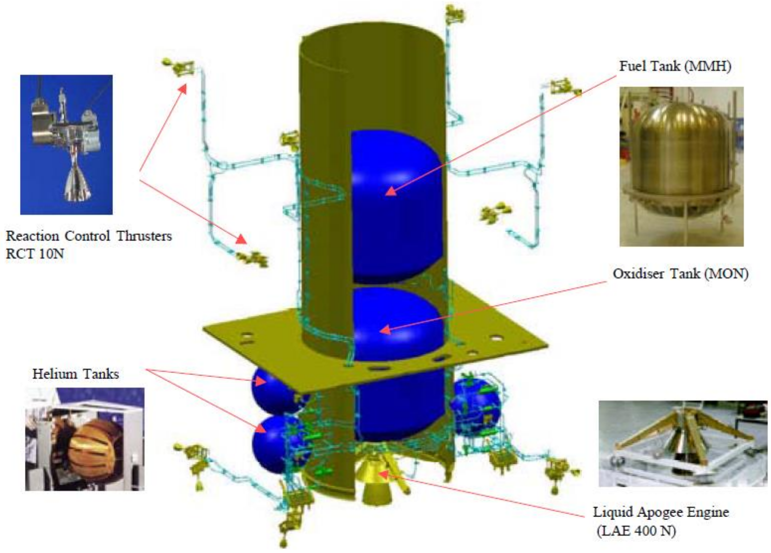


Figure 73: Bi-propellant propulsion system (courtesy Thales)

It incorporates a single 400 N liquid apogee engine (I_{sp} = 318 s) for orbit raising and 16 reaction control thrusters of 10 N each (I_{sp} = 293 s) for attitude control. The two propellant tanks are identical and can be sized with the propellant load carried. Propellant capacity is between 1542 and 3260 kg. Maximum internal pressure is 19.5 bar. Tanks are carried within the cylinder that makes up the main structure of the spacecraft close to the S/C centre axis as to limit MMOI of the vehicle. Figure also shows a set of Helium pressurant tanks that contains a pressurization gas that forces the propellants from the tanks to the thrusters as well as the propellant feed lines. The system has a life time of more than 15 years. Total dry mass of the propulsion system (thrusters, tanks, feed lines) is 192 kg.

Figure 74 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 burning a bipropellant (separate fuel and oxidizer). These powerful engines use mono-methyl hydrazine as fuel and nitrogen tetroxide as oxidizer. Sixteen smaller thrusters use hydrazine monopropellant 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.

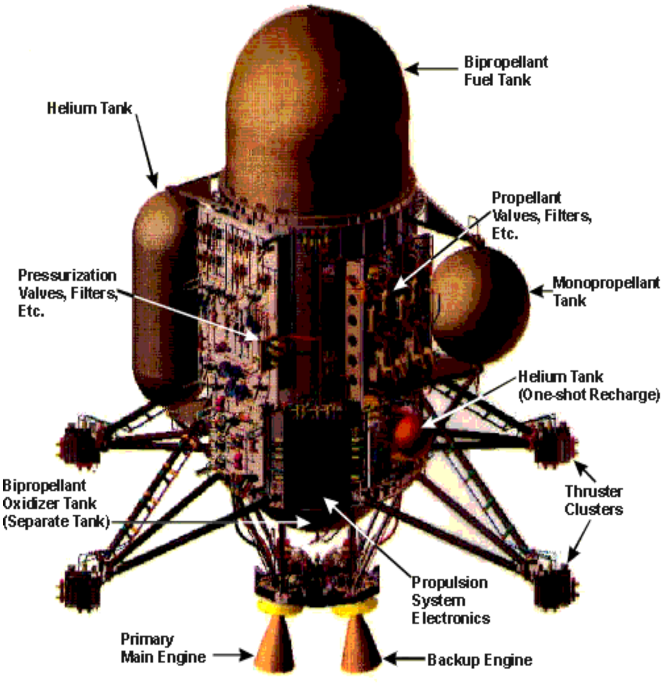
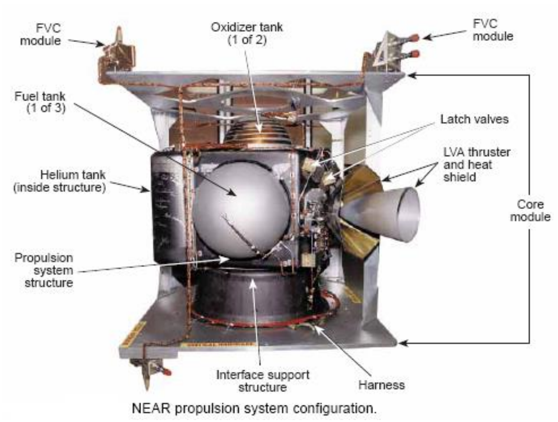


Figure 74: Propulsion system of Cassini spacecraft (Courtesy NASA)

Figure 75 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.



NEAR propulsion system configuration.

FVC is fine velocity control
LVA is large velocity actuator

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

Figure 75: NEAR propulsion system module lay-out and mass characteristics [Mosher]

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 monopropellant 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. A helium tank contains helium pressurant gas that forces the propellants from the tanks to the thrusters.

Next figure shows a monopropellant propulsion module for a small-scale Mars mission. The system uses hydrazine monopropellant. System mass is 39.5 kg of which 22 kg (~55%) is propellant mass. From the given total impulse (thrust times time) and propellant mass, an effective exhaust velocity of 1909 m/s (I_{sp} is ~195s) can be deduced (verify!). Thrust is provided by 4 sets of 3 thrusters each at the spacecraft circumference. This allows for full 3-axis control of the vehicle.

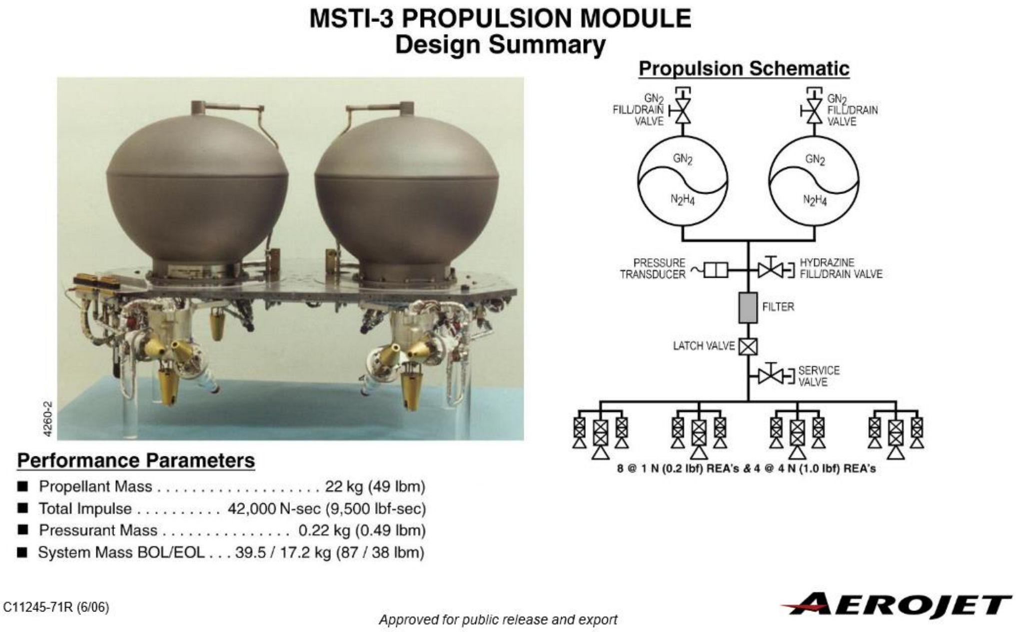


Figure 76: Storable bipropellant propulsion system for small-scale Mars systems (courtesy Aerojet)

Figure 77 shows a typical cold gas propulsion system as used on some spacecraft that require low thrust and low total impulse. The system shown on left in the figure includes 8 thrusters with thrust of 0.6 mN, a cold gas tank containing 14.8 kg of nitrogen gas (here used as propellant), valves, feed lines and a pressure regulator. The energy needed for propulsion in this case is the stored energy in the high pressure gas. At high pressure, the energy stored in the gas can be quite high. Data shows that propellant mass makes up about 60% of total system mass.

In general, it turns out that for most spacecraft applications one tends to use cold gas for low total impulse missions and bipropellant hypergolic systems (and solid systems) for high total impulse missions. Solid systems have the disadvantage that they can be started only once, so in case of two large manoeuvres or more multiple solid motors should be present. Monopropellant systems are mostly used for intermediate total impulse levels. The advantage for all systems (except the solid one) is that they can be used many times. Cold gas and monopropellant systems allow for a relatively simple, reliable and low cost system.

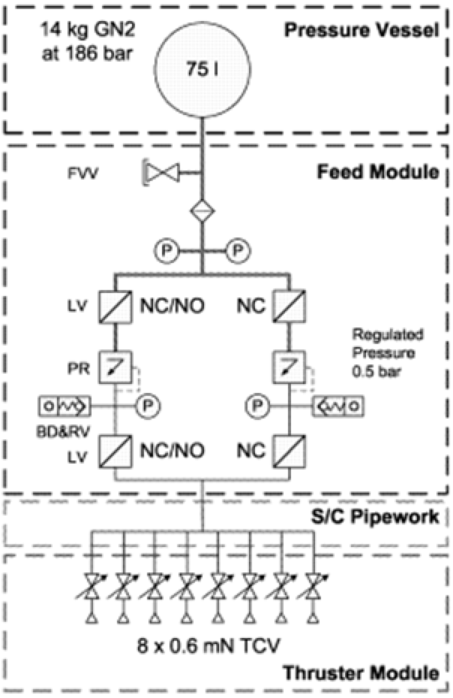


Table 1: Main Characteristics

Feature	GRACE	CryoSat	GOCE
Operational Lifetime [yrs]	5	3.5	2
Propellant Type	GN_2	GN_2	GN_2
Propellant Mass [kg]	33.1	37.4	14.8
Volume [liters] HP:	2 x 52	132	75
LP:	2 x 0.3	2 x 0.4	tbd
Specific Impulse [s] ACT:	65.3	74.3 [1)]	> 40
OCT:	70.2	76.6 [1)]	
Dry Weight [kg]	28.2 [4)]	53.7	tbd

1) measured by ESTEC
4) w/o brackets and thermal H/W

Figure 77: GOCE cold gas system and main characteristics of cold gas systems for three spacecraft [Usbeck]

2. Systems with separate power source

Electric systems (ion, plasma, resistojet, arcjet) as well as some thermal systems (solar-thermal, nuclear thermal) differ from chemical and cold gas systems in that a separate energy source is needed. As an example, Figure 78 shows at left a solar-electric propelled (SEP) module attached to the S/C. The module itself is shown in more detail in figure on right, but with stowed solar array. The module contains an ion propulsion system and two disc-like solar arrays providing approximately 15 kW of power at 1 AU. The design has three ion thrusters (two primary thrusters to perform the mission and one back-up). The module is controlled by the spacecraft processor, thus has limited other subsystems onboard. The SEP module dry mass is estimated at 778 kg, including mass growth. The module has a xenon propellant capacity of 660 kg. This gives a stage dry mass to propellant mass ratio of 778kg/660kg = 1.18

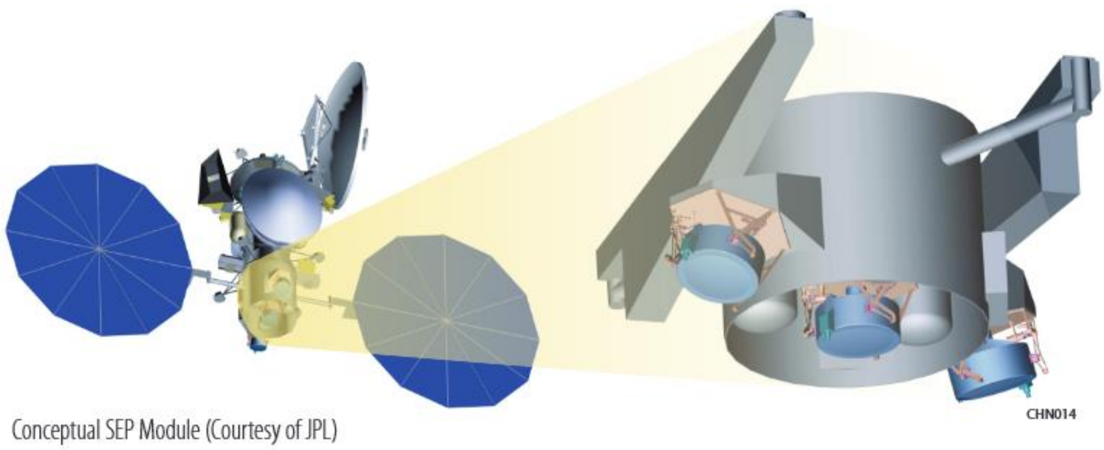
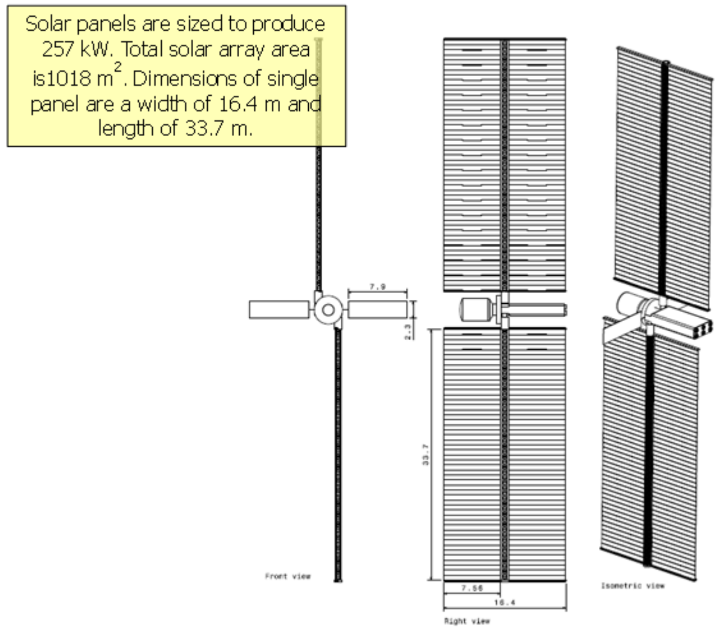


Figure 78: Electric propelled stage

Next figure shows a design of a large solar electric transfer stage capable of transporting 2.5 ton of cargo (payload) to the Moon.



	Mass (kg)
Electric Propulsion System	2,048
Electric Power System	2,034
Solar Array Assembly	1,541
Thermal Control System	1,120
Reaction Control System	255
Command & Data Handling	25
Attitude Determination	35
Communications	40
Primary Structure	2,050
Secondary Structure	513
Mechanisms	158
Docking Mechanism	357
Harness	248
Dry mass	10,425
Dry mass w 20% margin	12,510
Propellant mass	5,070
Wet mass	17,581

Mass breakdown of SEP stage excl. 2.5 ton cargo

LARGE SOLAR ELECTRIC TRANSFER STAGES FOR LUNAR EXPLORATION , by F. Gangami, R. Bos, B. Zandbergen, IAC-13-A 5.1.7

Figure 79: Large solar electric transfer stage [Gangami]

Total dry mass of the stage is roughly 12.5 ton (including margin) and propellant mass is about 5 ton. Together with the payload mass of 2.5 ton, this gives a loaded wet mass of the vehicle (including payload) of 20 ton. The electric propulsion system itself, i.e. the thrusters, tankage, etc. has a mass of about 2.0 ton and the electric power system and the solar array assembly together about 3.5 ton. Noteworthy is also the high mass of the thermal control system of 1.1 ton, which is because of the efficiency of the electric power system in relation with the large amount of power handled by the power system. Total thrust for this stage is given as 9.4 N (from a total of 6 thrusters working together), specific impulse is 2740 s and thrust efficiency is 63%.

From the above data, it follows a jet power of 126.3 kW or a required input power of the electric propulsion system of 200 kW. Total mass flow rate is ~0.35 g/s or about 58.3 mg/s per thruster. Based on the given propellant mass of 5070 kg, we estimate a (total) burn duration of 24,155 hour or about 4026 hour per thruster. Finally, using the rocket equation, it follows for the Δv that the vehicle can deliver:

$\Delta v = 2740 \text{ s} * 9.80665 \text{ m/s}^2 * \ln((17581 + 2500)/(17581+2500-5070)) = 7819 \text{ m/s}.$

From the data you may notice that the propulsion system and the EPS are quite massive. Also note that the thermal system is also quite massive. This is because a lot of waste heat (dissipated power) is generated and needs to be handled by the TCS.

Figure 80 shows a schematic of an electric propulsion system with the electric power source highlighted in the figure. All the other elements are essentially the same as for a (chemical) liquid monopropellant system except that a different (non-chemically reacting) propellant is used and that different thrusters are needed.

Table 49 shows dry mass of a specific electric propulsion system for two different types of electric power sources at two different power levels. From the table it follows that the power supply system makes up the largest part of the electric propulsion system dry mass (almost 80%).

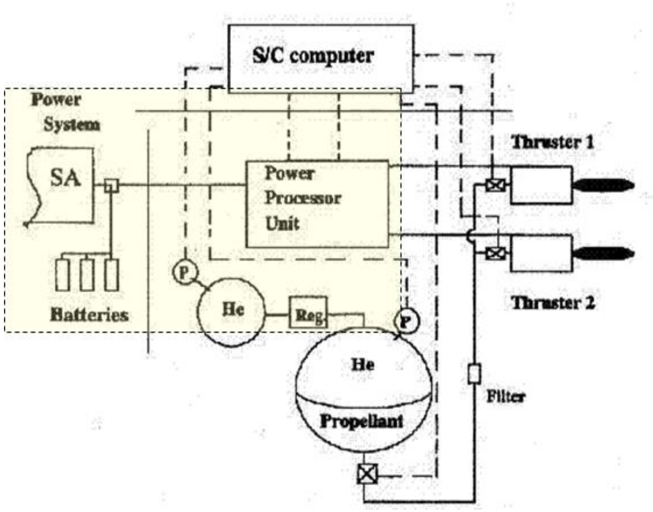


Table 49: Mass data of some specific electric propulsion systems

Specific weight of turbogenerator and thermionic electric propulsion systems

Component	Specific weight, lb/kwe			
	Turbogenerator		Thermionic	
	300 kwe	1500 kwe	300 kwe	1500 kwe
Reactor and conversion system	11.3	10.8	10.1	9.9
Thrust unit, propellant tank, associated plumbing, and controls	1.5	1.0	1.5	1.0
Contingency	1.0	1.0	1.0	1.0
Total	13.8	12.8	12.6	11.9

Figure 80: Schematic of electric propulsion system

Overview of propulsion system elements

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 81:

- one or more thrusters/engines/rocket[25] 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

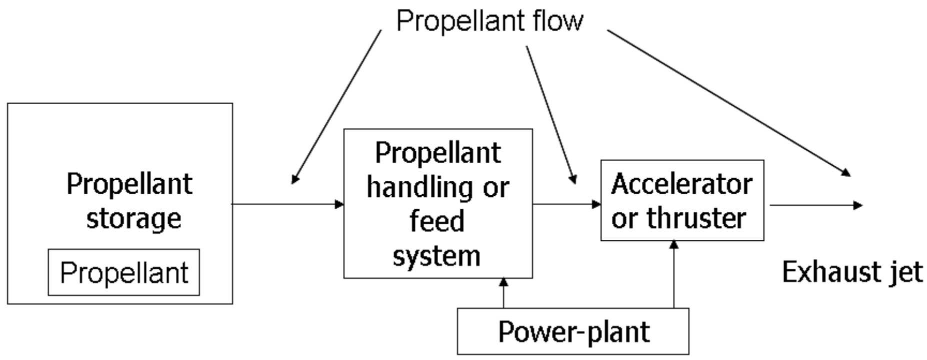


Figure 81: Rocket propulsion system elements

The thrust generation system generally consists of one or more thrusters. 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.

[25] **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.

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.

All 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.

Other important issues related to the design of chemical propulsion systems include:

- Exhaust of thrusters must be pointed away from the spacecraft. This is to prevent damage of the spacecraft by the hot exhaust and or the high velocity particles impinging on delicate S/C surfaces, like solar panels and optical lenses.
- Main propulsion: Work line of thrust shall be pointed through the CoM of the spacecraft as to not produce a disturbance torque.
- 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 (chemical) thruster to the spacecraft
- In some cases a separate upper stage or kick stage (for instance PAM-D) with integrated kick motor is used. The kick motor can also be integrated in the satellite, but this will lead to a higher mass of the spacecraft when on station. Typical propellants used are mostly solids (Star[26] series of motors) that allow for a low cost and simple design thereby allowing for short thrust times (reduced gravity loss).
- Number of thrusters for the RCS ranges from 2-24.
- Electric thrusters can be gimballed to allow for steering. This can help to reduce the number of thrusters needed.

Sizing and dimensioning

Many different types of propulsion system exist. To allow for meaningful designs, the strategy is to select various systems including cold gas, chemical and electric systems using different propellants and to size and dimension these systems after which the various designs can be compared and the "best" design can be selected for further development. Which propellant (combination) to select may depend on flight duration and number of thrust cycles? For instance, in case of single shot chemical systems, solid propellants may do well, however in case of multiple burns, one usually turns to the use of liquid hypergolic bi-propellants. Hereafter, we will discuss some of the necessary steps in some detail.

<u>Propellant mass estimation</u>

The first step in the sizing and dimensioning generally is to estimate propellant mass needed. This may be done per maneuver. For chemical systems, propellant mass generally makes up the largest part of the propulsion system mass as can be seen from Figure 82. This figure presents the mass breakdown of the propulsion system of two spacecraft excluding the structure mass needed to support the system. The NEAR propulsion system has its own 33 kg structure, separate from the spacecraft bus, whereas Clementine I has about 50 kg of structure. Focusing on the propulsion system itself, we learn from the figure that the liquid propellant mass makes up 75-80% of total propulsion system mass.

[26] 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.

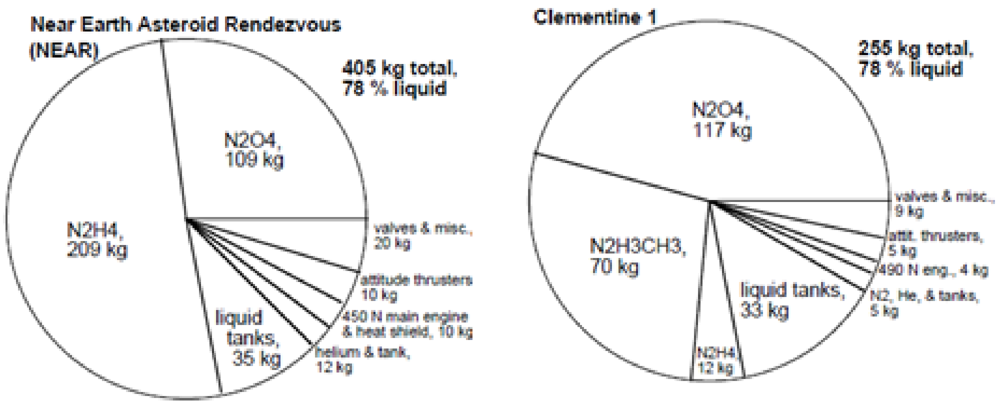


Figure 82: System mass distribution of chemical propulsion system of specific spacecraft [Whitehead]

Propellant load can be estimated using historical data on attainable specific impulse levels from vehicles using comparable propellants, see e.g. the data given earlier in this work.

When calculating propellant load, one should realize that specific impulse depends on the propellants selected 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 [104] *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.**[23]**) 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.

In the foregoing several different propulsion systems have been described and important data has been outlined. To allow for design purposes, the propulsion engineer can use collected data on spacecraft propulsion systems, whether integral or not and whether of the primary type or not to generate estimation relationships for e.g. mass, cost, etc. Below some first MERs are given as obtained using data from existing spacecraft.

Integral system dry mass estimation

Dry mass of *integral liquid chemical propulsion systems* can be estimated based on known propellant mass (M_p) using the following simple scaling rules:

Type	**Estimating relationship**	
Cold gas	$M_{rcs} = 0.99 \cdot M_p + 6.71 \;\; ; \;\; RSE \; = \; 42\;\%$ 5 data points; Propellant mass in range 2-40 kg	[113]
Monopropellant	$M_{rcs} = 0.178 \cdot M_p + 7.69 \;\; ; \;\; RSE \; = \; 8.1\;\%$ 15 data points; Propellant mass in range 30 – 300 kg	[114]
Bipropellant	$M_{rcs} = 0.0348 \cdot M_p + 58.15 \;\; ; \;\; RSE \; = 6.0\%$ 10 data points; Propellant mass in range 700-1800 kg	[115]

All mass values in kg.

Dry mass of *solid propellant kick motors* integrated into a spacecraft can be estimated using:

$$\left(M_{KM}\right)_{Dry}[kg] = 0.071 \cdot \left(M_p[kg]\right) + 18.97 \qquad [116]$$

The above relationship is based on 9 data points in the range 300 to 9500 kg and has an RSE of 16.5%.

For *non-chemical integral propulsion systems*. Like electric propelled spacecraft, no such rules are available yet due to lack of statistical data. As a first approach it is suggested to use the relation for monopropellant chemical systems to estimate the combined mass of thrusters, tanks, piping and feed system. To this we than still have to add the mass of the power-plant (see later in this chapter). However, in case the spacecraft generates sufficient power, we can use the EPS of the spacecraft. In case additional power is needed, this needs to be accounted for by designing a larger EPS.

Propulsive (kick) [27] stage dry mass estimation

Propulsive stages are essentially propulsive modules that are used for large manoeuvres and that can be separated when arriving on station. This gives as advantage that for the actual spacecraft the MMOI and the size are reduced thereby allowing for a more modest RCS system on-board of the spacecraft. Disadvantage is that the stage has its own structure to hold all of the propulsion system elements, see for instance the earlier mentioned TLI stage. Here again a distinction can be made between chemical propelled kick stages and electric or even solar-thermal propelled kick stages. The latter have a separate power-plant to provide for the necessary energy and in that case also the power-plant is part of the stage. A nice trade on different types of propulsive stages is provided in the work of [Martin et al, 2000].

Chemical kick stage dry mass can be estimated using the earlier (Chapter 3) introduced MER:

$$\left(M_{stage}\right)_{Dry} = 10 - 25\% \text{ of } \left(M_p\right)_{stage} \qquad [117]$$

As mentioned in Chapter 3, using a mid-range percentage value of 17.5% gives a mass uncertainty of ±45%. Of course, when performing a design, the reader is free to collect data of existing kick stages

[27] 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 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

and come up with an improved MER. For illustration, the mass data of some such kick stages are provided below (most data, except for TLI data, taken from Wikipedia):

- TLI (Trans-Lunar Injection stage, see earlier in reader) with a total stage mass of 1216.5 kg, propellant mass of 1015.4 kg (= 1010.7 kg + 1.1 kg + 3.6 kg) and a dry mass of 201.1 kg. This gives a dry mass to propellant mass percentage of 19.8%.
- PAM upper stage (US): PAM D has a loaded/empty mass 2,141/232 kg. This gives a stage dry mass to propellant mass percentage of 10.8%.
- IUS (Inertial Upper Stage): Essentially a two stage solid rocket upper stage. Stage 1 has a gross mass of 10400 kg and a propellant load of 9700 kg leading to a stage dry mass to propellant mass percentage of 7.2%. Stage 2 has a gross mass of 3000 kg and a propellant load of 2700 kg, leading to a dry mass to propellant load mass percentage of about 11.1%.
- Centaur upper stage (US): The total dry mass of the Centaur is 2,247 kg and the maximum propellant carrying capacity is 20,830 kg. This gives a kick stage dry mass to propellant mass percentage of 9.7 %.
- Fregat upper stage (Russian): The total dry mass of the Fregat-SB is 1,410 kg and the maximum propellant carrying capacity is 10,150 kg. This gives a kick stage to propellant dry mass percentage of 12.2 %.
- Liquid hydrogen – liquid oxygen kick stage [Martin et al, 2000]: Total inert mass of the stage is 7752 kg and total propellant mass carried is 61152 kg. This gives a kick stage to propellant dry mass percentage of 12.7 %

This data shows that the mid-range value of 17.5% used earlier is quite conservative and that a value in range of 12% and below is feasible. It is the experience of the author that the lower percentages are mostly for the larger kick stages as in that case the fixed mass (the mass that is independent of propellant mass) is relatively low. Of course this can be researched further so that it becomes clear what are the main governing parameters. This is left for the reader for him/herself to explore in some detail in future. As a starting point, a list of chemical kick stages can be obtained from http://en.wikipedia.org/wiki/List_of_upper_stages. The reader may also consider the relation provided to chemical propelled upper stages as provided in the reader "Launch Vehicle Design and Sizing".

For *electric propelled stages*, relatively little data is available. In the present reader though two examples are given that show a dry mass to propellant mass percentage of 200% and 118%, respectively. As a third entry may be used the Bepi-Colombo SEP stage, which has percentage of 148%. Clearly the dry mass to propellant mass percentage of electric propelled vehicles is much higher than for chemical propelled vehicles. This is associated with a.o. the mass of the power-plant that should be accommodated for. Using information from the above three vehicles and two further designs, the following MER can be derived (all masses in kg):

$$\left(M_{stage}\right)_{Dry} = 1.44\left(M_p\right)_{stage} + 241\ kg \qquad [118]$$

This relation is based on 5 data points only giving an R-squared of 0.67 and an RSE of ~54%. Notice the high fixed mass of 241 kg. This mass is usually associated with the avionics etc., or more in general those mass elements that do not scale with propellant mass.

For *solar thermal propelled kick stages* using hydrogen as main propellant, the following relations may be used (all masses in kg):

$$\left(M_{stage}\right)_{Dry} = 0.335\left(M_p\right)_{stage} + 455\ kg \qquad [119]$$

This relation is also based on 3 data points and has an R-squared of almost 1, but the RSE is much better (RSE = ~5%). Again the high fixed mass is related to elements not scaling with propellant mass.

Another approach may be to split up the dry mass estimation of electric and solar thermal propelled stages into two contributions, being 1) the thrusters, tanks and feed system and 2) the power-plant. This is, because for electric propelled systems, the mass of the power-plant (or power source) makes up a large part of the dry system mass (see earlier in this chapter). The first term could then be estimated using same relations as used for chemical systems as given earlier or in case of excessive propellant load, using relations for chemical rocket stages as given in the reader "Launch Vehicle Design and Sizing".

For a first estimate of the power-plant (the second term), we may use:

$$M_W = \alpha_W P_W \qquad \text{(compare [Fortescue et al], eq. 6-26)} \qquad [120]$$

With:

α_W = specific mass of power source (in kg/W); 1/ α_W = specific power.

P_W = power output of power source ($P_W = P_j/\eta$ where P_j follows using Eq.[111])

Here it is assumed that the mass of the power source scales linearly with power output. The proportionality constant is referred to as the specific mass of the power source. Typical values for the latter can be obtained from literature. As a first approach, the following specific power values are advised for use for electric power-plants:

- Photo-voltaic array: 10-40 W_e/kg (compare Eq.[23])
- Photo-voltaic system (incl. batteries): 2-30 W_e/kg (see table in earlier section on EPS design)
- Nuclear-electric: 2.5-100 W_e/kg [SSE]

This way of going into more detail is considered an appropriate way to improve estimates. However to obtain a really accurate estimate, it may be needed to model in detail the EPS as well as also the structure and the thermal control of the craft.

For *nuclear-thermal and solar-thermal propelled kick stages* a similar approach is suggested to determine stage dry mass. To estimate power-plant mass, the following specific power values may be used as a starting point [SSE]:

- 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

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.

Selection of thrust level

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.

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 information on propellant mass density and mass mixture ratio propellant volume can be estimated.

Table 50 shows typical properties of rocket propellants. Some propellants that have a low boiling point are stored at very low temperatures as to allow for the high mass density as indicated in the table.

Table 50: Properties of specific rocket propellants [Braenig]

Compound	Chemical Formula	Molecular Weight	Density	Melting Point	Boiling Point
Liquid Oxygen	O_2	32.00	1.14 g/ml	-218.8°C	-183.0°C
Liquid Fluorine	F_2	38.00	1.50 g/ml	-219.6°C	-188.1°C
Nitrogen Tetroxide	N_2O_4	92.01	1.45 g/ml	-9.3°C	21.15°C
Nitric Acid	HNO_3	63.01	1.55 g/ml	-41.6°C	83°C
Hydrogen Peroxide	H_2O_2	34.02	1.44 g/ml	-0.4°C	150.2°C
Nitrous Oxide	N_2O	44.01	1.22 g/ml	-90.8°C	-88.5°C
Chlorine Pentafluoride	ClF_5	130.45	1.9 g/ml	-103°C	-13.1°C
Ammonium Perchlorate	NH_4ClO_4	117.49	1.95 g/ml	240°C	N/A
Liquid Hydrogen	H_2	2.016	0.071 g/ml	-259.3°C	-252.9°C
Liquid Methane	CH_4	16.04	0.423 g/ml	-182.5°C	-161.6°C
Ethyl Alcohol	C_2H_5OH	46.07	0.789 g/ml	-114.1°C	78.2°C
n-Dodecane (Kerosene)	$C_{12}H_{26}$	170.34	0.749 g/ml	-9.6°C	216.3°C
RP-1	$C_nH_{1.953n}$	≈175	0.820 g/ml	N/A	177-274°C
Hydrazine	N_2H_4	32.05	1.004 g/ml	1.4°C	113.5°C
Methyl Hydrazine	CH_3NHNH_2	46.07	0.866 g/ml	-52.4°C	87.5°C
Dimethyl Hydrazine	$(CH_3)_2NNH_2$	60.10	0.791 g/ml	-58°C	63.9°C
Aluminum	Al	26.98	2.70 g/ml	660.4°C	2467°C
Polybutadiene	$(C_4H_6)_n$	≈3000	≈0.93 g/ml	N/A	N/A

Once propellant volume is determined, tank volume can be determined.

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 include a volume of roughly 10-20% of tank volume.

For non-chemical systems the same rules apply, but we also need to take into account the volume/size of the power source. For electrical power 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.

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
Propulsion system mass 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? To answer this question we make a distinction between systems with a separate energy source and those that don't.

For chemical rocket systems as well as for cold gas systems, it follows that dry mass is strongly related to propellant mass and hence minimization of propellant mass means that also total mass is minimized. However, it might still be possible, depending on the total impulse to be delivered, that a monopropellant system is less heavy than a bipropellant system. This is illustrated in Figure 83, where cold gas is compared to monopropellant and bipropellant systems. Cold gas has lowest specific impulse, whereas bipropellant has highest. Still for low total impulse, cold gas outperforms both monopropellant and bipropellant systems and only at high levels of total impulse the high specific impulse of the bipropellant system leads to a mass saving compared to monopropellant system. This is associated with the mass of the thrusters, valves, etc. that do not scale with propellant mass. In general, these are lightest for cold gas and heaviest for bipropellant systems.

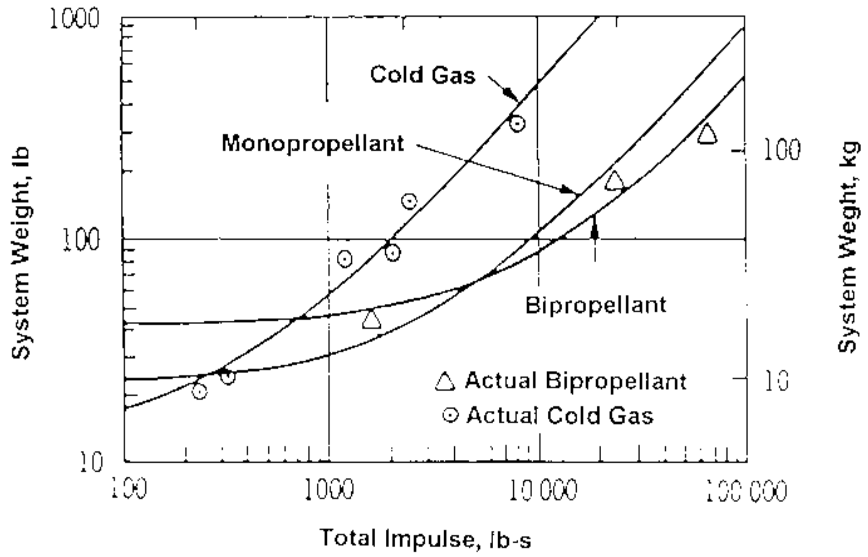


Figure 83: Mass comparison of different propulsion systems in relation to total impulse [Brown]

For non-chemical systems, the mass of the system can be considered to consist of propellant mass, the mass of thrusters and tanks and the mass of the power plant. From Table 49, we learn that the mass of the power-plant is almost 80-90% of the propulsion system dry mass. So for a first approximation, we may consider only the mass of the power-plant. We are interested in how system mass varies with increasing specific impulse. For simplicity, we consider the case that we have a constant thrust and a given propellant. Now when increasing specific impulse (say by regulating the thruster), it follows on one hand that propellant mass decreases. On the other hand beam power goes up. Assuming that thrust efficiency remains constant, input power needs to increase. For this a larger and hence heavier power-plant is needed. These two trends are schematically depicted in next figure, where a linear increase in power-plant mass is assumed with input power (or jet power).

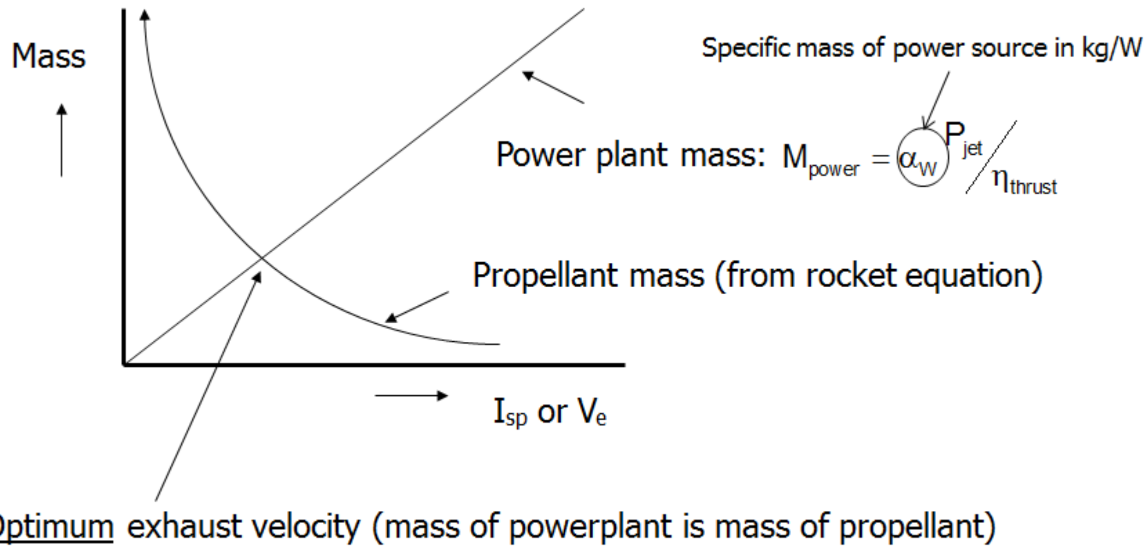


Figure 84: Selecting specific impulse for minimum system mass

Taking the sum of propellant mass and power-plant mass shows that there is a value of specific impulse or exhaust velocity for which propulsion system mass is minimum.

Next to mass and size, also other properties are of importance for design, like cost and reliability. To gain insight in these characteristics it is advised that the reader collects data of existing systems. Methods on how to use historical data to obtain average cost data and/or failure rate data and or to determine cost or reliability estimation relationships have been dealt with earlier for other properties, but can also be used for propulsion systems and their elements.

Problems

Hereafter a small number of problems are given for exercising upon. More problems for exercising upon are available via the course web pages (look for Mobius exercises) and in a separate workbook available from the TU-Delft online print shop. Of this workbook also an electronic copy is available on the course web site.

1. Explain in your own words why we need a space propulsion system and the main functions it fulfils.

2. Identify the forces that disturb the motion of a spacecraft and discuss their magnitude for an Earth orbiting satellite in relation to its orbital altitude.

3. List and explain the basic elements of a rocket propulsion system?

4. List the different rocket types of rocket propulsion used for propelling spacecraft in space and explain their differences.

5. A spacecraft in low Earth orbit experiences a drag force of 25.2 N, which leads to a decay in orbital altitude. To ensure that the spacecraft remains at altitude, you are asked to determine the thrust needed for this spacecraft in case we require that drag makeup is performed (a) continuous and (b) during only 5% of the orbital period.

   Answer: a) 25. 2 N, b) 504.2 N

6. Consider a 2 m diameter spherical shaped spacecraft covered with thermal protection material with a reflectivity of 0.8 at a distance of 0.5 AU from the Sun. Determine for this spacecraft the force due to solar radiation. You may assume the S/C to be in full sunlight.

   Answer: 0.103 mN

7. Consider a vehicle with a constant mass (no mass expulsion) of 1000 kg. You are asked to give the vehicle a velocity change ΔV of 10 km/s (about the velocity change required to escape the solar system). Calculate the total impulse I required, given that velocity increment losses due to gravity, drag, etc. can be considered negligible.

   Answer: 10 MNs

8. A vehicle of initial 1500 kg mass is accelerated through a ΔV of 10 km/s. Final mass of the vehicle is 1000 kg. Given that vehicle mass decreases linearly (e.g. through mass expulsion at constant mass flow rate) from 1500 kg to 1000 kg with increasing velocity, and that losses due to gravity, drag, etc. are negligible, calculate total impulse, I, required.

   Answer: 12.5 MNs

9. Consider a spacecraft of initial mass 2000 kg in low Earth orbit. To reach its destination, the spacecraft has to perform two manoeuvres:
   - Leave Earth orbit: Δv of 1 km/s

- Entry in target orbit: Δv of 1.5 km/s

For each of the two maneuvers a separate propulsion system is used, referred to as system 1 & 2. Of these, system 1 is used to leave Earth orbit and system 2 to perform entry in target orbit.

Determine for this vehicle the propellant mass for each of the two maneuvers given that:

a. System 1 and 2 each produce a rocket jet with a jet (exhaust) velocity of 3000 m/s
b. System 1 delivers a jet velocity of 3000 m/s and system 2 of 2200 m/s

Answers: a) 1131 kg; b) 1276 kg

10. A spacecraft is performing a propulsive manoeuvre resulting in a Δv of 2 km/s. Initial mass of the S/C is 2000 kg. Final mass prior to separation of the payload is 1000 kg. Given is that the S/C has a maximum allowable acceleration load of 0.2 g_o. Determine for the S/C the maximum thrust level allowed in case:

    a. Thrust is constant
    b. Thrust is throttled to limit the acceleration load

    Answers: a) 1961 N; b) 3922 N

11. A 16 kN vacuum thrust rocket engine has a vacuum specific impulse of 850 s and a thrust efficiency of 80%. Engine mass is 2000 kg. Calculate for this engine:
    a. Jet power
    b. Required input power
    c. Specific power
    d. Propellant mass flow

    Answers: a) 66.7 MW; b) 83.4 MW; c) 41.7 kW/kg; d) 1.92 kg/s

12. Consider a spacecraft in Moon orbit. This S/C is equipped with a rocket propulsion system. Given are:
    - Moon gravitational acceleration: 1.6 m/s^2
    - Thrust: 1000 N
    - Specific impulse: 220 s
    - Burn time: 1000 s

    Determine for this propulsion system:
    a) Effective exhaust velocity
    b) Propellant mass expelled

    Answers: a) 2157 m/s; b) 463.5 kg

13. An advanced (non-chemical) rocket propelled system is used to deliver a velocity increment of 4 km/s to a spacecraft with a total initial mass (including the propulsion system) of 2000 kg. Specific impulse of this advanced system is 3000 s, which gives a propellant mass of about 254 kg. The advanced propulsion system requires a separate on-board power source producing the required energy. Given are that the thrust produced by the system is 10 N, the power source has a specific power of 1 kW/kg, and the thruster system has a thrust efficiency of 80% calculate a) the mass of the power source and b) the thrust time.

    Answers: a) 184 kg; b) 207.6 hrs

Further problems for exercising upon are available via the course web pages (look for Mobius exercises) or can be found in the tutorial material.

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.

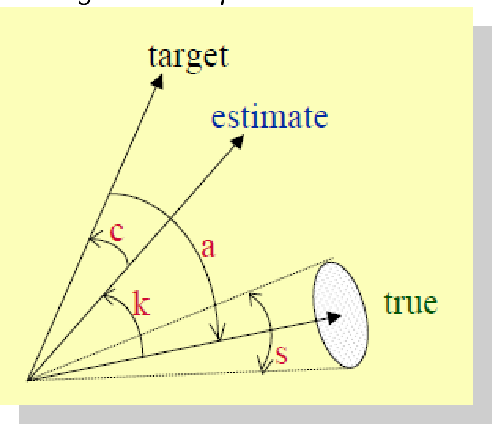
Some definitions

- 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 85: 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.

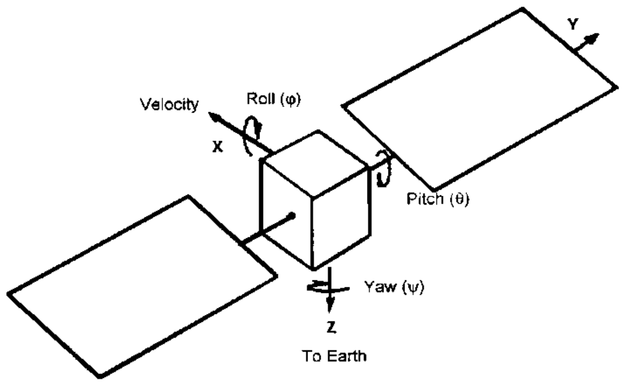


Figure 86: Definition of local reference frame

- 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 51

Table 51: 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 87 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.

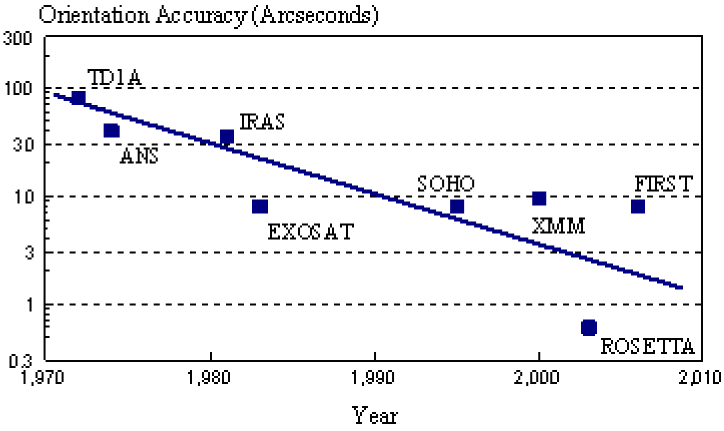


Figure 87: Trend in orientation accuracy for ESA scientific missions (courtesy ESA)

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.

Problem (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.*

Control modes
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.

Key characteristics
Key characteristics of the ADCS system can be obtained from Table 51. The table shows that attitude can be controlled within a range of less than 0.1º to 10º. We distinguish:

- Low accuracy pointing: > 1°
- Medium accuracy pointing: > 0.1 -1°
- High accuracy pointing: < 0.1°

Attitude (or bus) pointing knowledge is usually better than control.

The space challenge
In space there is essentially no friction. If a S/C is rotating in space, there is nothing that brings it to a stand-still. For attitude control, this means that we not only have to start the motion, but also to stop it. It also means that 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. Now in space, we have many small disturbances that can induce an unwanted rotation of the S/C, like:

- Gravity gradient torque
- Magnetic torque
- Solar radiation (pressure) torque
- Aerodynamic torque
- Mass expulsion and internal torques

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 now, we neglect all these details and focus on determining maximum values to learn about how serious these effects are and to allow for a first design of the ADCS, see later. More details may be obtained from [Fortescue, sections 9.4.2 to 9.4.5]. See also [Fortescue, section 9.2.2] to learn about the region where certain torques are dominant. In discussing the disturbances, we distinguish between disturbances external to the spacecraft (external disturbances) and internal to the spacecraft (internal disturbances)

External disturbances

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 88. 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.

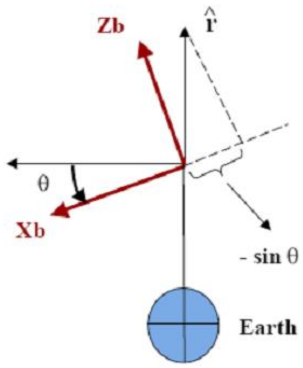


Figure 88: Vehicle attitude in relation with local vertical

For small (sin θ ~ θ in radians) 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:

$$T \cong 3 \cdot n^2 \begin{bmatrix} (I_{zz} - I_{yy}) \cdot \phi \\ (I_{zz} - I_{xx}) \cdot \theta \\ 0 \end{bmatrix} \qquad [121]$$

Here n is mean motion as defined by [relation 4.17, SSE]:

$$n = \sqrt{\mu / a^3} \qquad [122]$$

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-m² 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.*

For large angles, use (notice vector notation):

$$\bar{T} = \frac{3}{2} n^2 \left((I_y - I_z) sin(2\varphi) \right), (I_x - I_z) sin(2\theta), 0 \qquad [123]$$

Aerodynamic torque ("weathervane" effect)
Torque induced by unbalance in aerodynamic pressure on different sides (relative to CoM) of spacecraft.

$$\underline{T} = \underline{r} \times \underline{F}_a \qquad [124]$$

$$F_a = \frac{1}{2} \cdot \rho \cdot V^2 \cdot S \cdot C_D \qquad [125]$$

- 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.

$$\underline{T} = \underline{r} \times \underline{F}_s \quad [126]$$

$$F_s = (1+\rho) \cdot P_s \cdot S \quad [127]$$

- 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
- ρ stands for reflectivity of the sail (see also relation [101]).

Magnetic torque

$$\underline{T} = \underline{M} \times \underline{B} \quad [128]$$

- Vector T is the magnetic torque which is typically 3×10^{-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[28] at about 200 km altitude is 3×10^{-5} TESLA. For other planets, see [Fortescue; Table 2.7]. Sometimes Gauss is used as unit for the magnetic field strength: 1 Gauss is x 10^{-4} TESLA.

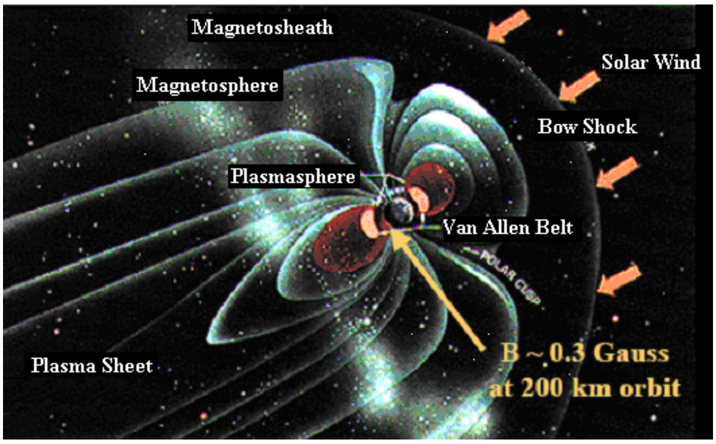


Figure 89: Artist view of Earth's magnetosphere

<u>Internal torques (due to e.g. thrust misalignment and or operating machinery, etc.)</u>

Mass expulsion torque (see also [Fortescue, section 9.4.6]:

$$\underline{T} = \underline{r} \times \underline{F} \quad [129]$$

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

[28] Magnetic field of Earth has been discussed in AE1110-II; see section entitled "Space Environment". For polar satellites (magnetic latitude is 90^o), magnitude of B can be approximated as $2M/r^3$ with M is magnetic moment of Earth 7.96E15 Tesla.m^3 and r is radius form Earth center to S/C center.

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:

$$H = I \cdot \omega \qquad [130]$$

- 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.

$$T = I \cdot \alpha \qquad [131]$$

Here α is angular acceleration.

For constant acceleration maneuver we find:

$$\Delta\theta = \omega \cdot t = \frac{1}{2} \cdot \alpha \cdot t^2 + \omega_o \cdot t \qquad [132]$$

$$\omega = \alpha \cdot t + \omega_o \qquad [133]$$

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.

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 occur due to the vehicle's interaction with the environment). In general, passive control is applied for low accuracy pointing, whereas active control is needed for high accuracy missions. We distinguish the following control categories:

- Coarse control (> 5°)
- Low accuracy pointing: 1 - 5°
- Medium accuracy pointing: > 0.1 -1°
- High accuracy pointing: < 0.1°

Furthermore, control may be exercises about all three S/C axis or less.

<u>Coarse control</u>

- 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.
- Aerodynamic stabilization. The principle is to use a planet's atmosphere to stabilize the vehicle. For aerodynamic stabilization, we need to ensure that the center of pressure as experienced by the spacecraft is behind the center of gravity.

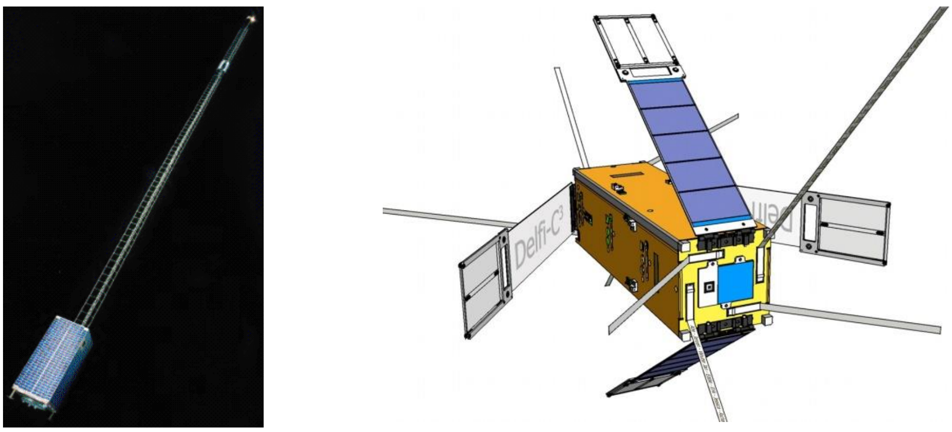


Figure 90: Passively stabilized spacecraft (left: Ørsted ; right: Delfi C^3)

Figure 90 shows the Danish Ørsted gravity gradient stabilized spacecraft (left figure). Ørsted was gravity gradient stabilized using a deployable 8 meter instrument boom. Magnetic torque coils maintained yaw to within 10 degrees. Attitude determination was via a star camera, with sun sensors and magnetometers as backup. Figure on right shows the Dutch Delfi-C^3 spacecraft. Delfi-C^3 uses passive attitude control by applying rods of magnetic material [Hennepe]. Only a very rough control effect was expected as illustrated by the solar panels, which allow obtaining solar power from any direction and the omnidirectional antennas used, see later section on communication.

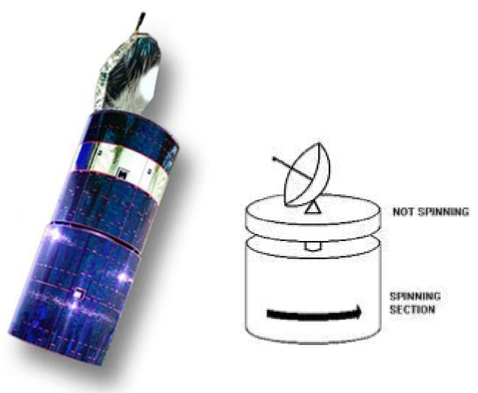
For all three methods described it follows that the lower the orbit the stronger the stabilizing effect. However, the effect allows for control about two axes only.

<u>Low accuracy to high accuracy pointing</u>

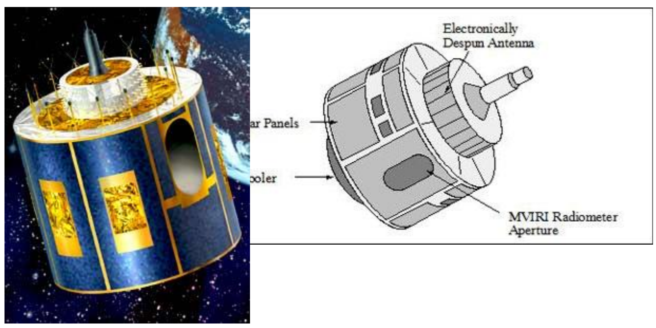
- Spin Stabilization: With spin stabilization, the entire spacecraft rotates around its own vertical axis, spinning like a top (this effect is sometimes referred to as the gyroscopic effect). This keeps the spacecraft's orientation in space under control. The advantage of spin stabilization is that it 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. Criteria for a well-designed spinning spacecraft are that the centre of mass must lie on the user-specified spin-axis, the spin axis is the principal axis that has the largest principal (mass) moment of inertia and the difference with the lateral moments should be substantial [Goldstein et al. 2001].

- 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 “de-spin” maneuvers.

Figure 91 shows two spin stabilized spacecraft. The first spacecraft is the Boeing 376 communications satellite of which 58 were built. It has two telescoping solar panels and antennas that fold for compactness during launch. Vehicle (depending on the version) has a launch mass of about 1445 kg, diameter is 2.13 m and total height is 6.7 (7.97 m) m. The spacecraft is of a dual spin design where the lower section spins to provide stability and the upper section (antenna) is de-spun to allow for antenna pointing. Spin rate is 50 rpm. To improve pointing the spacecraft has 4 (22 N) hydrazine thrusters with a total of 136 kg propellant. Antenna pointing is ensured with an accuracy of better than 0.10 degrees. The spacecraft on the right is the European Meteosat (first generation) spacecraft. It is composed of a main cylindrical body, with a drum-shaped section of 1.3 m diameter and two concentric cylinders stacked on top. The main cylindrical body contains most of the S/C subsystems including the payload and rotates at 100 rpm to provide stability. The outer surface of the vehicle is covered with six solar panels (more than 8000 cells) for power generation (200 W average). The surface of the drum-shaped section is covered with an array of radiating dipole antenna elements. Electronics within the drum activate the individual elements in sequence, in reverse order to the S/C spin sense. The drum constitutes an electronically de-spun portion of the S/C, namely the antenna whose function is to ensure that S-band communications are always directed towards Earth. Two Earth horizon sensors and two sun sensors provide attitude information.



Boeing 376 Coms satellite (courtesy Boeing)

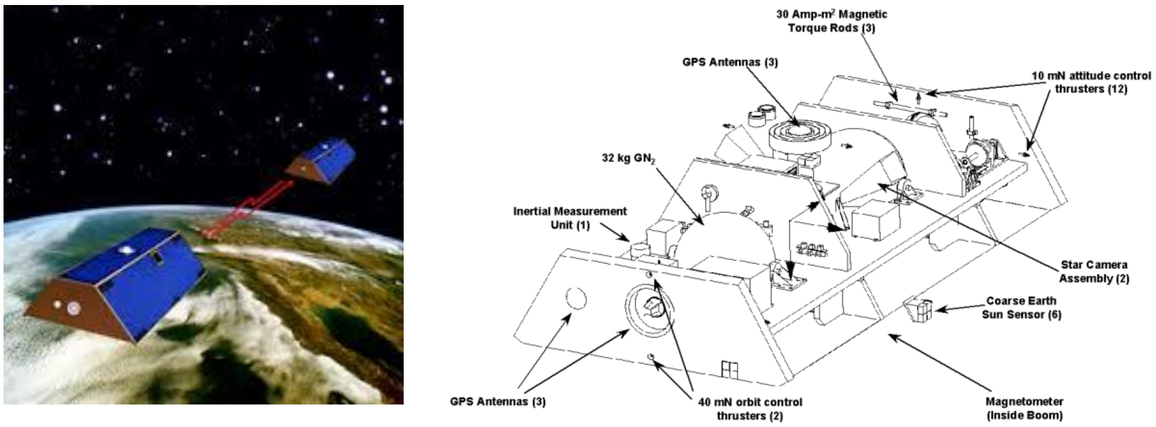


Meteosat first generation (courtesy ESA)

Figure 91: Spin stabilized spacecraft

Next figure shows the GRACE spacecraft (NASA) as an example of a 3-axis stabilized spacecraft. As compared to spin-stabilized spacecraft note the absence of axi-symmetry. It shows:

- a cold gas propulsion system for attitude control (12 x 10 mN) and orbit change (2 x 40 mN) maneuvers using a total of about 32 kg of nitrogen
- a set of three magnetic torque rods for attitude control in support of the cold gas system
- two star cameras to provide the inertial attitude (attitude relative to some inertial frame)
- interfaces to a GPS receiver to provide 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

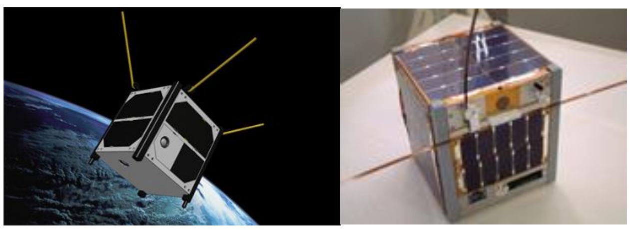


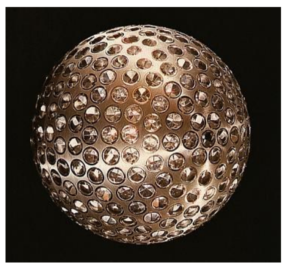
Grace spacecraft (artist view and internal lay-out of ADCS) retrieved from: http://op.gfz-potsdam.de/grace/satellite/satellite.html

Figure 92: A typical 3-axis stabilized spacecraft

It should be clear from the foregoing, that we need not only to be able to control the motion of the vehicle about the three principal vehicle axes, but also determine the attitude of the vehicle with respect to say Earth. For this sensors are needed that allow determining the attitude to some inertial frame.

A special category of spacecraft is formed by those that are designed without having any means of attitude control. However, this is mostly limited to low cost microsatellites and special research vehicles. Some examples are given in Figure 93.





Lageos (courtesy NASA)

Figure 93: S/C with no means of attitude control

How well the attitude is controlled affects amongst others the design of the EPS, and the communications subsystem of the spacecraft, but also other systems may be affected. Reasons are that solar panels need to be pointed to the Sun, communication antennas to for instance Earth ground station or some relay satellite in space and/or cameras to allow viewing the desired object.

System elements

From the foregoing we should have gotten the idea that the ADCS system consists of elements that allow for attitude determination (sensors) and for attitude control (actuators). Next to that also one or more computers are needed that allow for using the measurements to determine attitude and to determine whether the spacecraft is correctly oriented and to determine what actions are needed to control the attitude. Below these three elements are discussed in some detail.

<u>Attitude sensors</u>

Various sensors exist that allow for attitude determination. We mention:

- Sun sensor: An optical instrument 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. Once the direction of the Sun is known, at least the attitude of two –vehicle axes can be determined.
- 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 will be discussed in some detail in a later course. Typical sensor performances and some limitations to their use are given in Table 52 taken from [SMAD]. An important limitation for the sun sensor, earth sensor and star camera is that a single camera only allows for determining the attitude of two vehicle axes. To determine the third axis, a second instrument is needed. This is the reason that the GRACE spacecraft has two star cameras. The reason for GRACE to have also 6 sun sensors (on each S/C surface one) is that it allows for quick attitude determination (but less accurate). The reason why there are 6 is because over the mission, the Sun may be on different sides of the spacecraft. The sensors which are shadowed by the spacecraft body are than not active. Grace also has an IMU with rate gyro that can take over from the star sensors to allow reducing calculation load of the onboard computer. Typically 3 rate gyros are needed (one for each axis).

<u>Attitude control actuators</u>

Typical actuators include:

- Reaction wheels [Fortescue, 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. 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 the reaction wheel is saturated and the stored momentum is maximum. At that time the wheels must be de-saturated. For this usually another system (thrusters or magnetorquers) is needed.

The torque delivered by a reaction wheel can be determined using [131], whereas the total momentum stored in the wheel is given by [130].

Table 52: 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.

- 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:

$$\underline{T}_m = \underline{a}NIA \times \underline{B} = \underline{D} \times \underline{B} \qquad [134]$$

  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[29]. 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:

$$\underline{T}_{thrust} = 2\ \underline{F}_T \cdot \underline{L} \qquad [135]$$

  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.

A comparison of the three actuator discussed is given in Table 53. Some example calculations can be found directly following the table.

[29] 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.

Table 53: Advantages and disadvantages of specific attitude control actuators

Actuator	**Remarks**	**Advantages**	**Disadvantages**
Torque rods	Minimum three needed to allow for control about all three axis	Propellant free	Limited to low Earth orbits where Earth's magnetic field is sufficiently strong. Generation of torques can only be done in direction perpendicular to the magnetic field vector
Reaction wheels	Minimum three needed to allow for control about all three axis	Propellant free. Very high attitude accuracy possible	Need for momentum dumping because wheels get saturated
Thrusters	Minimum 12 needed for pure 3-axis control		Attitude control accuracy depends on minimum impulse of thruster used.

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 2475 deg/s or 413 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 [114]. 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 [136], 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[30] 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 94 next to some elements that are associated with position determination and orbit calculation and orbit prediction (see section on navigation).

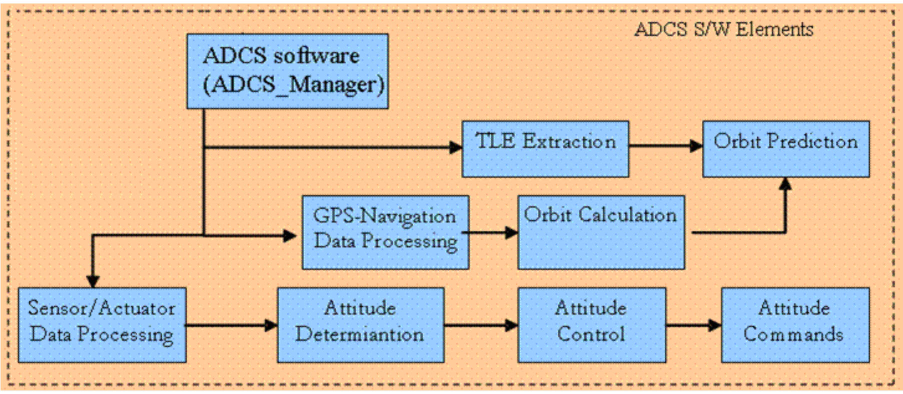


Figure 94: AOCS software (TLE= Two Line Elements)

Drawing the system

Figure 95 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.

[30] An algorithm is a finite sequence of instructions, an explicit, step-by-step procedure for solving a problem.

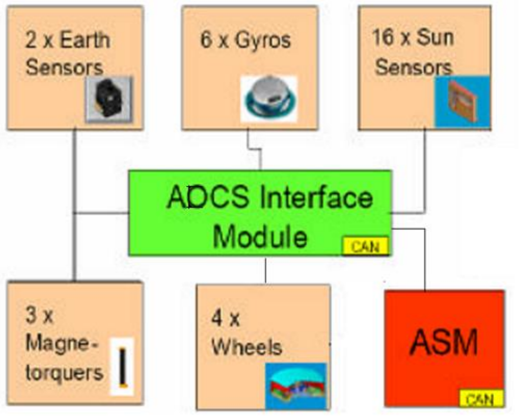


Figure 95: ADCS block diagram (ASM = Attitude Safety Module)

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 at least 2 sensors are needed. More sensors are needed in case the object's (Sun, Earth, stars) position in the sky varies
- 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
- Rate gyros allow for accurate attitude determination, but another more accurate system is needed to allow for periodic resets of attitude information.
- 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.
    - For magnetorquers or reaction wheels this requires at least three devices. For reasons of back-up (in case one fails) usually a fourth is present
    - For thrusters, we need at least 12 thrusters, to allow for pure 3–axis control. For each axis we need 4 thrusters that are placed as far away as possible from the CoG, see Figure 96. This allows for limiting the thrust level and hence propellant consumption. Two thrusters are needed to start and 2 to stop the motion. For reasons of redundancy (back-up) sometimes double the amount of required thrusters is used. In case of non-pure 3-axis control, fewer thrusters can be used.
- 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)

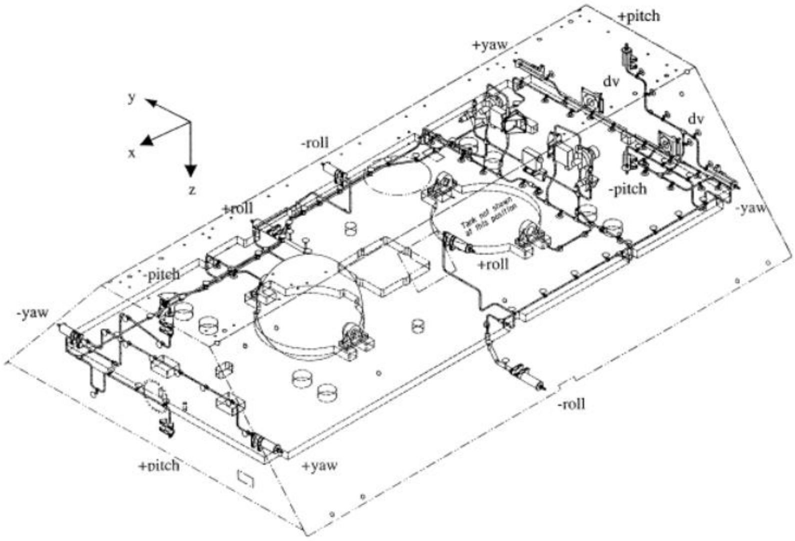


Figure 96: Architectural design of Grace cold gas system [Schelkle]

Dimensioning and sizing

In general, we find that the ADCS represent roughly 5-10% of the mass of the bus, see appendix D. In this section some more detailed data and/or estimation relationships are given that allow for estimating sensor and actuator mass of ADCS systems.

<u>Sensors</u>

Sensor sizing data are given in the next table.

Table 54: 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

<u>Actuators</u>

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	[136]
Torque rods	$\mathrm{M_{tr}} = 0.0167 \cdot D + 0.4876\ ;\ \mathrm{R}^2 = 0.9595$ 10 data points; D in range 1 – 800 Am^2	[137]

With H in Nms and D in Am^2.

For mass of thrusters, see mass estimation of RCS under propulsion. Propellant mass estimation associated with attitude control can be performed once the total impulse generated by the disturbance torques and required for slewing is known.

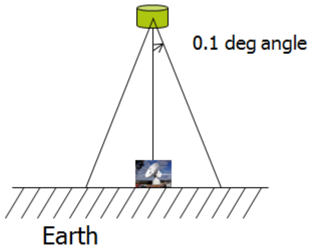
Other characteristics of the ADCS
Reliability, power usage and cost figures of typical ADCS can be obtained from Appendix D. It follows that the ADCS

- accounts for roughly 1/3rd of all serious bus failures
- uses roughly 10% of the total bus power

For more detailed data, see appendix D.

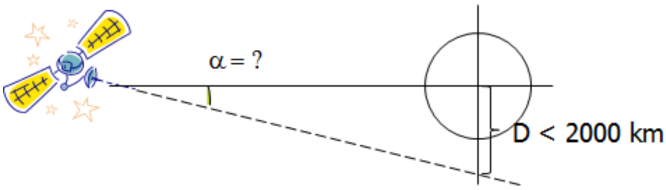
Problems

1. An S/C in GEO uses a small antenna to communicate to ground, see figure. Antenna pointing accuracy is ±360 arc-sec. What is maximum distance between target and point where antenna pointing line (line of sight) intersects with Earth (Earth curvature is negligible)?



Answer: Maximum distance is 63 km.

2. An S/C is in orbit about Jupiter at 6 AU. It uses a large antenna to communicate with Earth. Antenna is pointed by S/C. What pointing accuracy (angle α, see figure) is needed to ensure that line of sight of antenna crosses Earth disc at a point within 2000 km from the target (Earth centre)?



Answer: Pointing accuracy is 0.5 arcsec.

3. Consider an S/C in circular low Earth Orbit at 500 km altitude. For this S/C are given:
I_{xx} = 1000 kg-m^2
I_{yy} = 1000 kg-m^2
I_{zz} = 1100 kg-m^2

What is for this S/C the maximum gravity gradient torque experienced?

Answer: Maximum gravity gradient torque experienced is 2.6 x 10^{-4} Nm.

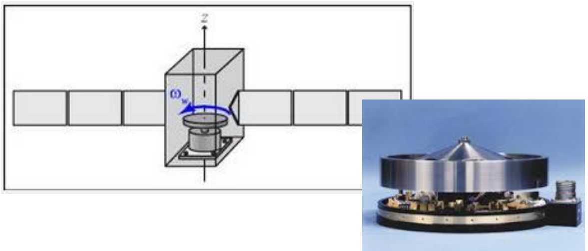
4. Consider an S/C in low Earth orbit. Calculate for this S/C the disturbance torque due to atmospheric forces on the S/C given that the cross-sectional area perpendicular to the velocity vector is equal to the linear dimension squared and that the distance between centre of mass and centre of pressure is less than 2% of the linear dimension. Given are:
   - Maximum mass density at altitude of 10^{-9} kg/m^3.
   - Linear dimension of S/C = 1.7 m
   - Orbital velocity of 5 km/s
   - Drag coefficient: 3

Answer: 3.68 mN

5. For a 800 kg S/C are given:
   - MMOI about x-axis: 700 kg-m^2
   - Disturbance torque about x-axis = 50 mNm (constant)
   - All other disturbances are considered negligible

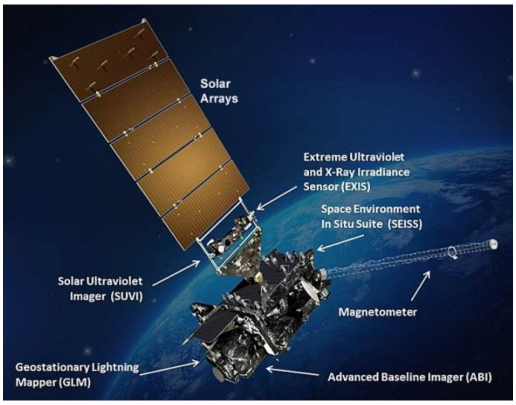
   Determine for this spacecraft:
   a) Angular acceleration ($\ddot{\alpha}$) about x-axis
   b) Rotational rate of S/C (in rad/s and in rpm) after 1 hour if left unattended
   c) Angle over which the vehicle rotates in 1 minute
   d) Change in angular momentum over this 1 minute
   e) (Average) control torque needed in case we have a reaction control system that is active for only 10% of the time

6. A satellite of MMOI of 500 kgm^2 needs to be rotated over an angle (about one of the principal axes of the S/C) of 5 deg. This maneuver shall be accomplished using a reaction wheel, see figure. For this RW are given:
   - Initially at stand-still
   - MMOI: 0.05 kgm^2
   - Manoeuvre duration: 10 sec.
   - Constant Acceleration



   What is maximum rotational velocity of wheel?

7. A spacecraft is powered by a single solar wing, see figure. In sun-light, the array is rotating at a slow rate to allow for solar tracking. In eclipse a fast rewind occurs. During both manoeuvres the vehicle is held stand-still (non-rotating) by a set of reactions wheels.



   Consider the simplified case of a single reaction wheel (RW) with the wheel axis coinciding with the array, rotational axis. Furthermore are given:
   - Wing rotational rate during rewind is 0.3 deg/s;
   - Principal mass moment of inertia (MMOI) of RW about wheel axis is 0.01 kg-m^2;
   - At start of rewind manoeuvre RW is rotating at 1000 rpm;
   - Principal mass moment of inertia of S/C about relevant axis: 1000 kg-m^2;
   - Wing MMOI about rotational axis is 5.4 kg-m^2;
   - Friction effects are negligible.

   Maximum change in rotational velocity of the reaction wheel during the rewind manoeuvre is in range …

   Answer: 162 deg/s

Further problems for exercising upon are available via the course web pages (look for Mobius exercises) or can be found in the tutorial material.

4.6 Command and Data Handling (C&DH) system

Why command and data handling?
The C&DH system essentially makes up the brain and nervous 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[31] 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.

The extent of the C&DH system greatly determines the autonomy of the spacecraft, but also the cost and the mass of the system. For typical subsystem data, see appendix D.

Key requirements
Key requirements for the C& DH system include:

1. Number of devices (payload and other) to be commanded. For a simple spacecraft, this may amount to less than 200 devices to be commanded, whereas for more complex spacecraft, this can easily amount to more than 500 devices.
2. Number of payloads providing data. Most times there is only a single payload, but on some spacecraft there are multiple payloads. Payloads may include cameras, repeaters[32], scientific instruments, etc.
3. Number of devices to be metered for the housekeeping of the satellite, i.e. to ensure the wellbeing and the proper operation of the spacecraft including the payloads. For instance, for a power source one would like to monitor the amount of power delivered to ensure it remains within bounds. The power delivery can be determined in case the voltage of the power source and the current delivered are known. An example telemetry data table is given in Table 55. In total 57 TM points are shown. For each TM point are given a number, an identifier, a numerical value and the units of measurement.

[31] The term ordnance in this context refers to devices containing some explosive materials, like explosive bolts, pyrotechnic valves and igniters.
[32] In telecommunications, a **repeater** is an electronic device that receives a signal and retransmits it at a higher level or higher power, or onto the other side of an obstruction, so that the signal can cover longer distances.

Table 55: 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
```

A highly complex telemetry (TM) system may acquire data from over 1000 different points (TM points). For a complex system this may be anything in the range of 200 – 1000 TM points and for a simple system this is anything below 200 TM points. Next table provides an overview of the number of points telemetered for some specific satellites.

Table 56: Number of TM points for specific spacecraft

Spacecraft	# of TM points
Delfi C^3	114 (estimate)
Delfi n3Xt	135 (current best estimate)
Intelsat 5	520
Eutelsat II	840
SPOT	~500
MSX	~400
ERS	6600
Envisat	13700

4. Payload data rate or how many data are generated by the payload(s). Data generated includes science data, images and text, sound and video messages. 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). Some example data rates may be obtained from Table 5. 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.
5. Telemetry data rate or how many data are to be measured per unit of time and how accurately; It concerns data, like equipment temperature, tank pressures, electric currents and voltages, related to the well-being/health/status/internal environment of the on board equipment. Usually measurement data rate is determined per signal. Summing up the individual data rates gives an overall data rate.

6. Command data rate or how many commands are to be handled per unit of time for each system that is commanded. Simple commands may include such commands as switch on/off, but also more complex ones, wherein say a specific level is to be set. For simple devices, this can include up to 50 commands per device, whereas for complex devices, this amounts to more than 50 commands per device. For instance, the Thermal Ion Dynamics Experiment shown in Table 5 uses 155 different commands to control its operation. As a further example, next table shows 14 commands typical for a telecommunications subsystem.

Table 57: Communications subsystem commands

1	Transmitter Power	ON	2	Transmitter Power	OFF
3	Subcarrier Oscillator	ON	4	Subcarrier Oscillator	OFF
5	Coherent Mode	ON	6	Coherent Mode	OFF
7	Ranging	ON	8	Ranging	OFF
9	Convolutional Encoder	ON	10	Convolutional Encoder	OFF
11	Transmit Ant. Select	LGA	12	Transmit Ant. Select	MGA
13	Receive Ant. Select	LGA - 1&2	14	Receive Ant. Select	LGA - 1 Only

7. Storage time or time that the CDHS is required to store data and/or commands and have it available for retrieval. This may greatly depend on the orbital altitude and the duration of the periods a ground station is in view of the spacecraft

The following steps are advised for generating CDHS requirements:

1. List all devices for which data are to be handled
    - For each device list the type of data (images, voice, music, video, measurements) to be handled
    - Determine number of images, voice/music/video channels and measurement signals to be handled when active and when inactive
    - Determine duty cycle of each device
2. Determine number of telemetry points for each device
    - For each point determine measurement accuracy and frequency
    - Determine duty cycle[33] for each telemetry point
    - Determine required bit rate per point and in total
3. List devices to be commanded
    - For each device determine number of commands
    - Determine how many commands to be processed in time
4. Determine contact times with ground and time in between contacts.

It may be difficult to get all requirements clear at the start of the project. Hence you are advised to work in rounds. After a first round of requirements generation, start designing the system. Once you have a first design (with most likely many things still being uncertain, like some requirements), you can start a second round, etc.

Some fundamentals for determining data rates

In current spacecraft command and data handling systems all data (whether text, images, music, video, or measurement data) and commands are at some point digitized. 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.

[33] Duty cycle of a device tells you how much of the time the device is active.

Command data rate
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. These messages come from:

- a ground station via an uplink
- the on-board computer (OBC), and
- a hardline test interface

As an example of a typical command, we give:
DISP:STAT ON;DATA “Hello, world!”<nl>

This command shall turn a display on and display:
“Hello, world!”.

For illustration, Standard-Commands-for-Programmable-Instruments (SCPI) can be found in the SCPI standard, see http://www.ivifoundation.org/docs/scpi-99.pdf.

Each command essentially consists of a certain number of words/characters. Knowing the amount of words/characters allows for determining the number of bits per command; see also the next entry on “Text”. Command message size (in Bytes or a multiple thereof) can be determined once we know the number of devices to be commanded, the number of commands to be send up per device and the number of bits per command. The actual picture is a bit more complex, but for now the general picture presented above will suffice. Command data rate can be determined from the command message size by taking into account the total time available for transferring the message.

Text
Text can be converted into numbers (ASCII[34]) which in turn can be digitized using the binary number system. In the binary number system, the base is 2 (as opposed to 10 for the decimal number system) and only two bits are used: 0, 1. A binary number can be made up from a sequence of bits. The position x (starting from the right) in the sequence represents an x-1 power of the base. For example, the binary number 10101 represents $(1 \times 2^4) + (0 \times 2^3) + (1 \times 2^2) + (0 \times 2^1) + (1 \times 2^0) = 21$ in the decimal system. With 8 bits (1 byte), 256 different combinations can be made, which essentially allows for 256 different characters to be included. Note that a typical keyboard has 256 different characters. Extending the number of bits gives more possibilities and hence more different characters (including Chinese, Japanese, etc.) can be included or we just switch keyboards.

Measurement signals
Measurement signals are either of an analog 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 (like time). A digital signal is a discontinuous signal that changes from one state to another in discrete steps.

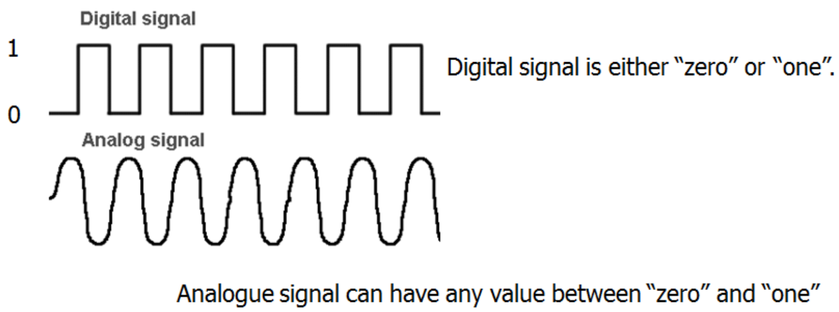


Figure 97: Analog (top) versus digital (bottom) signal

[34] ASCII stands for American Standard Code for Information Interchange. Computers can only understand numbers, so an ASCII code is the numerical representation of a character such as 'a' or '@' or an action of some sort.

Analog signals can be converted from a smooth continuous value to a discrete, digital number (A/D conversion), see Figure 98. For this the signal must be sampled after which each sample value is digitized.

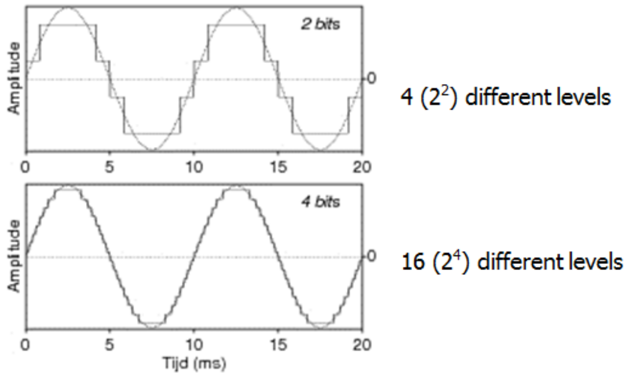


Figure 98: Digitization of analog signal (top digitization using 2 bits and bottom 4 bits)

Top figure shows a coarse digitization using only 2 bits (4 different values), whereas lower figure shows a much finer digitization using 4 bits or 2^4 (16 different values). The number of bits (n_{bits}) per sample determines number of divisions into which a measurement range is divided to approximate an analog input; the higher this number, the lower the quantization error.

An important question is how often the signal is to be sampled? In case the signal is constant in time, the number of samples can be small, but in case of signals with a highly time-varying component (like music) the number of samples depends on the frequencies in the signal. For this, the Nyquist sampling theorem is applicable. This theorem states that:

In order to obtain an accurate representation of a time-varying analog signal, its amplitude must be sampled at a minimum rate that is equal to or greater than twice the highest sinusoidal frequency component that is present in the signal.

Hence to digitize an analog signal we use:

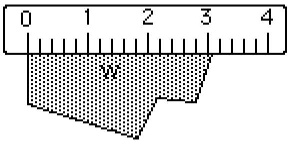
$$DR_{analogue} = f_c \cdot N_{Nyq} \cdot n_{bits} \quad ; \quad f_s = f_c \cdot N_{Nyq} \qquad [138]$$

Here DR is data rate (for instance in bps, kbps or Mbps), f_c is highest frequency in analog 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 and n_{bits} is number of bits per sample, typically ranging from 2 up to about 16, but more is possible.

Example 1: 1000 analogue signals are measured. Highest frequency to be measured in each of the signals is f_c =10 Hz with 8 bits digitization. This gives a data rate of (10 x 2.2) samples/s x 8 bits = 176 bps per signal or 176 kbps for all signals together.

Example 2: 1000 TM points measured at a frequency (f_s) of 10 Hz (10 samples taken every second) with 8 bits digitization gives a data rate of 80 kbps

The number of bits per sample determines how accurate the magnitude of the original signal is represented. In general, the higher the number of bits per sample, the more accurate the representation of the original signal (amplitude) is. Or said otherwise, it is the number of bits per sample that determines the error made in the digitization of the information, i.e. the quantization error. To explain the 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 cm, (0,1) as 2.5 cm and (1,1) as 3.5 cm, 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):

$$e_q(\text{in }\%) = \frac{1}{2^{n_{bits}+1}} \cdot 100 \qquad [139]$$

Images/video (a sequence of images)
Images are digitized by displaying each image in the form of a two-dimensional matrix of individual picture elements - known as pixels. For simple black and white images a single binary digit is needed to represent each pixel, a 0 for a white pixel and a 1 for a black pixel. For color images a whole spectrum of colors is produced by using different proportions of red (R), green (G), and blue (B). Depending on the number of bits per color the different proportions can be defined. To digitize images, the following relation is used:

$$DR_{image} = N_{images} \cdot N_{pixel} \cdot n_{bits} \qquad [140]$$

Here DR is again data rate, N_{images} is the number of images digitized per second (for video this may be 50 images per second or more), N_{pixel} is number of picture elements per image, and n_{bits} is number of bits per sample or per pixel, ranging from 2 (black and white) to 24 for three colors at 256 different levels (true color) up to 48 bit for three colors (deep color). Typical data rates for video are given in Table 58. Values range from 20-1500 Mbps, depending on the quality of the signal and the size of the image and the spatial resolution required (number of pixels in width and height direction of the image). In case of multiple video channels, the overall data rate increases linearly with the number of channels (assuming equal quality).

Table 58: 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.

For imaging (not video) cameras, the number of pixels of an image depends on the size of a scene (say width and height or length) and the spatial resolution required. The latter depends on the smallest element still to be seen on the image. For instance, consider we are viewing an area of 185 km x 185 km directly beneath the spacecraft and we specify a spatial resolution of 30 m, see Figure 99 (figure on left), it means we need roughly 6177 (= 185000m/30m) pixels in along track direction x 6177 pixels in cross track direction or roughly 36 million pixels. Consider the camera from our example is installed onboard of a satellite that is revolving about Earth. This would mean a next image is needed once the satellite has travelled 185 km over Earth. The time to generate a single image then depends on the ground speed of the satellite and the length of the scene according to:

$$N_{images} = \frac{length\ of\ scene\ on\ ground}{v_{ground}} \quad [141]$$

Where the ground speed depends on the orbital velocity of the spacecraft, the altitude (h) above the planet surface and the radius (R) of the planet:

$$v_{ground} = v_{orbit}\frac{R}{R+h} \quad [142]$$

Example: Image data rate estimation
Suppose we have a camera on board of our spacecraft that is taking true color images from Earth. Given is that each image consists of 20,000 pixels (100 x 200). It follows that each image consists of 20,000 pixels x 24 bits (true color) = 480000 bits.
Furthermore, 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. Next, 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 480 kb = 338.4 kbps.

The data rate calculated in the above example is a maximum data rate. If open spaces are allowed between two subsequent images, the data rate reduces. Still for a designer, the maximum data rate is determining.

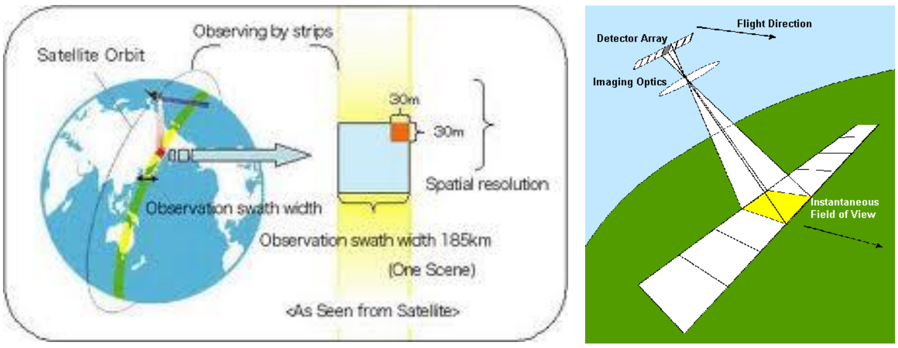


Figure 99: Observation of planet from satellite orbit

For observation cameras, the size of a scene is not known beforehand, but depends on the instrument design and the distance between the camera and the scene to be observed. Important instrument parameters are the instrument Field of View and the pixel Field of View in both cross and along track direction. Both are angular parameters, see Figure 6. The *FoV* in along track direction determines the length (l) of the scene, whereas the *FoV* in cross track direction determines the swath width (w) of the instrument. The pixel *FoV* determines the spatial resolution, i.e. the smallest element that can still be viewed. For an instrument that is looking in nadir direction (down to Earth), it follows for the width and length of the scene:

$$w = 2h \cdot tan\left({FoV_{cross\ track}}/{2}\right) \quad [143]$$

$$l = 2h \cdot tan\left({FoV_{along\ track}}/{2}\right) \quad [144]$$

Here *h* is the orbital altitude of the spacecraft, i.e. the distance to the object.

For the spatial resolution of a square pixel, we obtain for the spatial resolution directly below the spacecraft:

$$Spatial\ resolution = 2h \cdot tan\left({IFoV}/{2}\right) \quad [145]$$

Most current observation cameras do not have a square (or rectangular) detector area, but a line scanning imager, see Figure 99, figure on right. For such a system, every time a line is scanned on Earth. How many lines need to be scanned every second to allow no gaps depends on the *FoV* in along track direction, i.e. the thickness of the line measured on ground also referred to as the ground sampling distance (GSD), according to:

$$N_{lines} = \frac{GSD}{v_{ground}} \quad [146]$$

Notice that in case of a line scanner and square pixels, the *IFoV* is identical to the *FoV* in along track direction.

In case all picture elements are square (which is not necessarily the case), it follows that the GSD is equal to the spatial resolution of the picture. Next dividing swath width by GSD gives the number of ground elements viewed, i.e. the number of detector elements on the line detector array (each element on ground has its own detector or pixel element):

$$N_{pixel/line} = \frac{w}{GSD} \quad [147]$$

As an example, Table 59 provides typical properties of the Ball High Resolution Space Camera (HRSC), taken from Appendix G, which is designed to fly on a platform (satellite) at an altitude of 400 – 900 km. The camera allows for taking panchromatic (black and white) pictures as well as multispectral (color) pictures. For the latter 4 different colors (bands of colors) are viewed. Data provided includes spatial resolution for both black & white and color images in terms of pixel *FoV*. This is 1.37 micro radians when viewing in black & white and 5.47 mrad when viewing in color (multispectral). This gives a GSD of ~0.5 m at 400 km altitude and ~2 m at 900 km altitude for black and white images and ~2 m and ~5 m for color images. Each pixel is represented using an 11 bits quantization. From the pixel *FoV* and the cross track *FoV*, the number of detector elements can be determined. Dividing the *FoV* cross track by the *IFoV* gives the number of pixels in cross track direction. When viewing in black & white, this gives ~27000 pixels and in case of multispectral viewing, we have ~6750 pixels.

Table 59: Ball High Resolution Space Camera resolution and FoV characteristics

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)

* Note mrad in table stands for micro-radians and not milli-radians.

Example:
Considering that the HRSC instrument is at 400 km altitude. This would mean that it has a GSD of 0.5 m for black & white pictures. Using the given Field of View (FoV) cross track of 2.12 deg, it immediately follows a swath width of 14 km at an altitude of 400 km over Earth. For a satellite in circular orbit at 400 km altitude, follows an orbital velocity of 7.669 km/s. Using the relation for the ground speed as given in the foregoing, we obtain a ground speed of ~7.2 km/s. Given a GSD of 0.5 m follows that we have 0.5 m/7200 m/s = 7E-5 seconds to read 1 line. Per second, this means we need to read 14400 lines of ~27000 pixels. With a 11 bits quantization, this gives a bit rate of 14400 x 27000 x 11 = 4.3 Gbps.

The same reasoning can be followed for the multispectral images but in that case, we have ~6700 detector elements per color. Per second we need to read 3600 lines of 6700 elements per color band. At 11 bits quantization, this comes down to 4 bands x 3600 lines/band x 6700 elements/band x 11 bits = 1.1 Gbps.

Above relations allow for an approximate estimation of maximum bit rate. In practice, bit rates can be smaller, especially when taking into account image compression techniques, see later. An extensive treatment though is considered beyond the scope of this course.

Music/voice
Music and voice signals can be digitized in the same way as highly time-variant measurement signals. We just need to determine the highest frequency in the signal that is still relevant for properly understanding the signal and the number of bits per sample. For instance, for CD quality music, the number of bits used generally is 16 bits/sample with a sampling rate of 44 kHz (2.2 times 20 KHz, with the latter being the highest frequency still relevant for good music). For voice lines, the frequency of sampling may be much less (down to about 2.2 x 3.4 kHz), see also Table 58.

Example: Bit rate estimation
FM radio quality covers a frequency range of 15 kHz, see table. Using the Nyquist-Shannon criterion, and given 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.

Data compression
For data rate estimation, it is also important to consider data compression. Data compression reduces the number of bits to the most essential ones. From Wikipedia: In computer science and information theory, data compression involves encoding information using fewer bits than the original representation. Compression can be either lossy or lossless. Lossless compression reduces bits by identifying and eliminating statistical redundancy. No information is lost in lossless compression. Lossy compression reduces bits by identifying unnecessary information and removing it. The process of reducing the size of a data file is popularly referred to as data compression, although its formal name is source coding (coding done at the source of the data before it is stored or transmitted). As an example of compression we mention the use of MP3 to store CD quality music or the storage of photos as jpegs. This on one hand may affect the quality of the signal, but above all, it will greatly reduce the storage space required.

$$DR_{compressed} = DR_{uncompressed}/CF \qquad [148]$$

Here CF is the compression factor. Typical values of the compression factor for both lossless and lossy compression can be taken from literature, but values of up to about 50 seem well possible.

Key characteristics
Key characteristics of the C&DH system are given in Table 60.

Table 60: Key characteristics CDHS [Sarsfield]

Mission	Spacecraft	Central Processor (Mips)	Mass Memory (MB)	Harness Pinouts (No.)	Lines of Flight Software Code (K lines)
New Millennium	Deep Space 1	20.0	128	5,000	5,000
	Earth Observer 1	25.0	1,200	—	—
SSTI	Lewis	8.0	2,000	—	—
	Clark	8.0	2,000	—	23
Surveyor	Mars Global Surveyor	0.3	750	12,256	18
	Mars Surveyor '98—Lander	20.0	128	3,300	30
	Mars Surveyor '98—Orbiter	20.0	128	2,500	30

Key characteristics include:

- Memory size (in Mega-Byte or Giga-Byte; 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:

$$V_{produced} = DR_{produced} \cdot t_{dump\ fraction} \qquad [149]$$

  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.

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 (kb) 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 h x 3600 sec/h = 80.64 Mb or 80.64/8 = 10.08 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.

- Handling speed: A good measure for handling speed, i.e. how many data can be read per seconds or how many lines of code can be processed, is the number of instructions that can be handled every second, typically expressed in Mega Instructions Per Second (MIPS). Typical radiation hardened (space) processors used include 80386, 80C86, NSCC-1, RAD6000, Mongoose V, and LEON[35]. These processors have processing speed (also referred to as throughput or handling speed) in the range from less than 1 MIPS up to 100 MIPS.

[35] An overview of processors used in space can be obtained from http://www.cpushack.com/space-craft-cpu.html

Throughput or processing speed is difficult to estimate. A very approximate estimate can be determined using:

$$\text{Throughput} = N_{\text{LOC}} * \text{I/LOC} \qquad [150]$$

Where:

- N_{LOC}: # (number) of lines of code (LOC, i.e. statements that a programmer writes) processed per unit of time
- I/LOC: # of (machine) instructions carried out per LOC
- # of instructions (I) per LOC depends on programming language.
    - Higher languages, like C^{++} or JAVA, translate to about 5 to 10 machine instructions per LOC (I/LOC = 5-10)
    - Assembly language: 1 instruction per LOC.

Processing speed in general is limited by computer clock speed. A clock speed of 100 MHz roughly indicates a maximum processing speed of 100 MIPS (non-parallel computing). Or in case we need 100 MIPS, we need to select a computer with a clock speed of at least 100 MHz.

> *Example: Consider a C^{++} program for ADCS that has 10,000 LOC with each line representing 10 machine instructions per line. This program is run 10 times per second, which means that every second the data from the attitude sensors are processed and the control outputs determined.*
>
> *This means that the computer has to perform maximum 1,000,000 (=10,000 LOC x 10 instructions/LOC x 10 Hz) instructions per second. This translates in a processing speed or throughput of roughly 1.0 MIPS.*

Another consideration is that the data in the onboard memory must at some point be transmitted to ground. All these data need to be processed by the processor which requires also processing speed. Depending on the ground contact time[36] different data rates follow for reading out the memory:

$$V_{transmit} = t_{transmit} \cdot DR_{readout} \qquad [151]$$

Here $V_{transmit}$ is the memory volume/size to be transmitted, $t_{transmit}$ is the time available for transmission and $DR_{readout}$ is the readout data rate.

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].

> *Example: To transmit 10.08 MB in 6 minutes, requires a read out data rate of 224 kbps and hence a processing speed of roughly 0.25 MIPS.*

- **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/lines, 5000 K lines is roughly 1500 books of 100 pages each). 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.

[36] 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.

Figure 100 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.

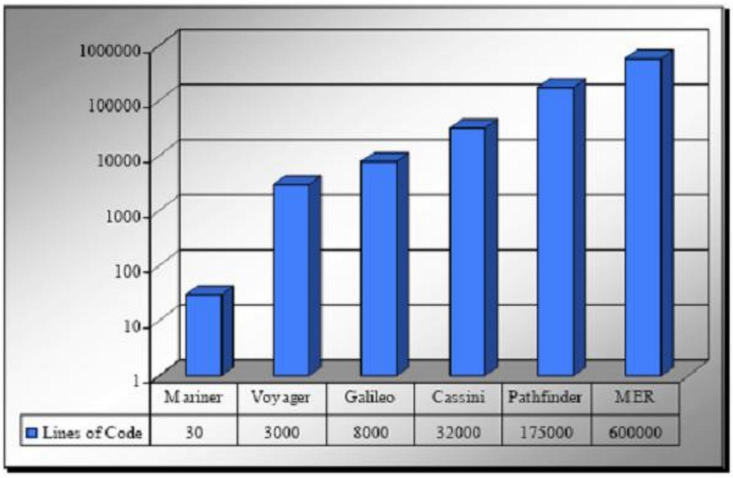


Figure 100: Size of software in S/C missions

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).

- Pinout
  Pinout (sometimes written "**pin-out**") stands for the number of contact pins of an electrical connector of electronic component trough which signals are transferred. Each pin essentially has its own wire to allow connecting the device with some other device or computer. The larger the number of pins, the more wires is needed and hence the mass of the wiring harness increases, see later.

General characteristics of the CDHS
The CDHS of a spacecraft typically accounts for roughly 10% of all serious bus failures, see appendix D. It furthermore represents only a small % of bus mass, see also appendix D, although for small spacecraft, the amount can be considerable (up to 22%). Power consumption is low, but surely not to be overlooked.

System elements
To learn about the important elements that make up the CDHS, it is considered insightful to read about the build-up of the CDHS of a number of spacecraft.
Table 61 provides an overview of the CDHS of some specific spacecraft.

Table 61: Overview of CDHS of specific spacecraft

S/C	Description
Delfi-C^3	The CDHS consists of a fully autonomous On-Board Computer (OBC). It is based on a commercial FM-430 board from Pumpkin Inc. It allows for commanding and collecting telemetry data from the various controlled instruments on board of the spacecraft. The OBC gathers data every second and packs this in an alternating series of housekeeping and payload frames which are transferred to the COMMS, which on its turn directly sends these frames in AX.25 format. There is no data storage on-board, so reception of the data depends on visibility of the satellite by ground stations. [Delfi-C3 web pages]

Near-Shoemaker	The command and data handling subsystem is composed of a redundant light computer, two redundant command and telemetry processors, two solid state recorders, a power switching unit, and an interface to two redundant 1553 standard data buses for communications with other subsystems. The two command and telemetry processor units consist of a number of printed circuit boards that provide for different functions, including amongst others the processing of commands and data (processor board), the collection of housekeeping data including 62 voltages, 31 temperature signals and a range of other signals, a board that allows for commanding on-board devices. Each unit has a weight of 5 kg and measures 24 x 24 x 17 cm^3. The flight computer consists of two redundant processors . It weighs 5 kg and occupies a volume of 3770 cm^3. Power consumption is 8W. The two solid state recorders have 1.1 Gbits of storage and 0.67 Gbits, respectively. [Stott, 1989].
Pluto-Kuiper Express	The on-board computer is a 1.5 MIPS RISC-based system capable of processing a science data stream of 5 Mbps. The solid-state data storage system is capable of storing 400 Mb in both compressed and uncompressed formats.
Mars Express	The spacecraft is run by two Control and Data management Units with a 10 gigabit solid state mass memory for storage of data and housekeeping information for transmission.
Mars Reconnaissance Orbiter	The spacecraft is run by a spaceflight computer based on 133 MHz PowerPC processor technology. Other components of the CDHS include the flight software and a 160 Gbit large solid state recorder for recording science data and housekeeping data.
Mars Odyssey	Command and data handling is through a RAD6000 computer with 128 Mbytes RAM and 3 Mbytes of non-volatile memory. The latter allows the system to maintain data even without power. The subsystem runs Odyssey's flight software and controls the spacecraft through interface electronics. Interface electronics make use of computer cards to communicate with external peripherals. For redundancy purposes, there are two identical strings of these computer and interface electronics, so that if one fails the spacecraft can switch to the other. Communication with Odyssey's sensors that measure the spacecraft' orientation in space, or "attitude," and its science instruments is done via another interface card. A master input/output card collects signals from around the spacecraft and also sends commands to the electrical power subsystem. The interface to Odyssey's telecommunications subsystems is done through another card called the uplink/downlink card. There are two other boards in the command and data handling subsystem, both internally redundant. The module interface card controls when the spacecraft switches to backup hardware and provides the spacecraft time. A converter card takes power from the electrical power subsystem and converts it into the proper voltages for the rest of the command and data handling subsystem components. The last interface card is a single, non-redundant, one-gigabyte mass memory card that is used to store imaging data. The entire command and data handling subsystem weighs 11.1 kilograms

In summary, the following system elements are distinguished:

1. Processor Module or Processor Board for software (see later) execution + central memory that holds the on board software + accurate clock that maintains spacecraft time and generates all of the internal CDHS clock signals;
2. A board that provides the electrical connections by which the processor communicates with other processors and peripherals including the mass memory, the payloads and the communications system. Compare e.g. how for a PC system or laptop the motherboard connects to for instance a printer, video screen, keyboard, loud speaker, microphone and modem. The mother board generally also holds an accurate clock or timer that maintains spacecraft time.
3. Data acquisition or On-Board Data Handling (OBDH) unit or Housekeeping Board in charge of processing the bus interrogations sent by the data management processor module(s). It allows for acquiring amongst others analogue signals, like equipment voltages and currents, digital signals and temperature transducer outputs and transfers the data to the processor board or module.
4. An encoder unit or board (or frame transfer generator) that allows for encoding the telemetry data (protecting the data against data corruption) and which presents the data to the communications subsystem ready for transmission to Earth.
5. A decoder unit that processes the tele-commands received by the communications subsystem, see next chapter. Accepted tele-commands will be transferred to the processor module/board for further action.

6. Memory or data storage unit (solid state mass memory or solid state recorder, hard drive, floppy disk, tape recorder)
7. The network (cable harness/wires + connectors, optical links, WIFI?)
8. Software. This software ensures the correct operation of the mass memory as well as the correct operation of the various payloads and the spacecraft bus subsystems during all mission phases.

The first 4/5 elements can be combined in a single unit, like for the NEAR spacecraft the command and telemetry processor [Stott, 1998] and for Venus Express the Command and Data Management Unit. The 5th element sometimes is considered as part of the communications system The 3rd element may sometimes be combined with the power switching unit (part of PDU, see electrical power system) in a single Acquisition and Control Unit (DACU).

Computers, command processors, control units, etc.
Figure 101 shows some typical CDHS hardware elements.

Figure 101: Specific examples of CDHS hardware


On-Board Computer (OBC)


Command processor


OBDH box-type (with some cables attached)

The figure shows 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) and solid state recorder (SSR), also referred to as Solid State Mass Memory (SSMM). Solid state recorder is basically a bank of Random Access Memory (RAM) boards, used on-board spacecraft to collect and store data, see Figure 102. It 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.

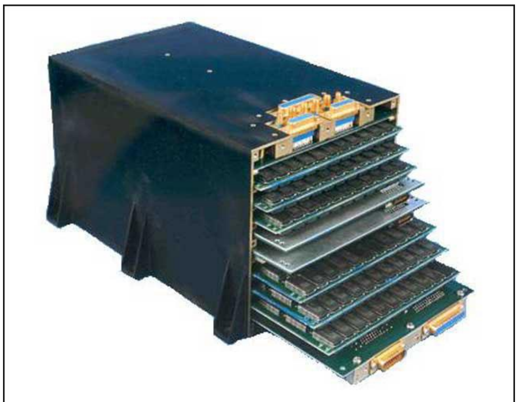


Figure 102: Solid State Recorder (courtesy SEAKR)

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.

Configuration

How the various C&DH elements relate to each other can be seen in Figure 103. 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[37] (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/tele-command) 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:

[37] 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.

- 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.

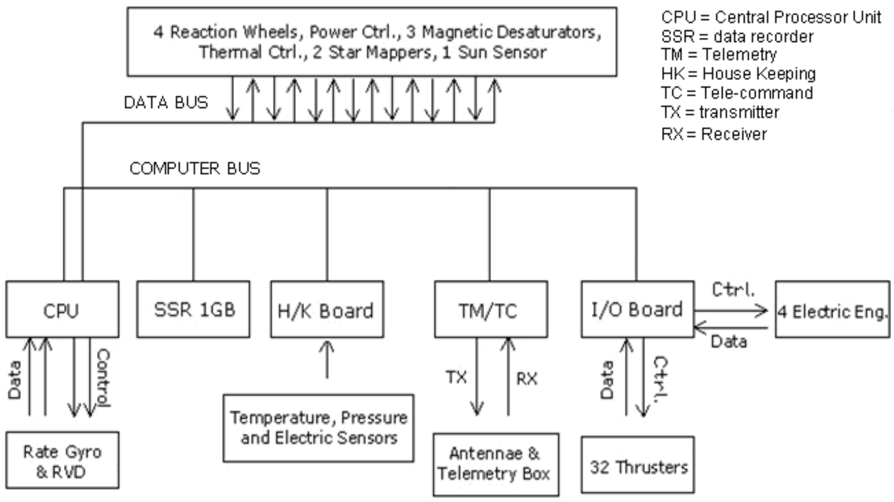


Figure 103: Typical CDHS set up (architecture)

Dimensioning and sizing

For the dimensioning and sizing of the CDHS, general rules can be applied as can be derived from historic data. Some typical data for mass, power and reliability are provided in appendix D. Below some further details are given that allow for estimating mass/size/power of the various units that make up the CDHS. 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 unit (excluding data storage)

For sizing of a data handling/acquisition unit we can use the following relations:

Size (mass) determined by number of TM channels[38]: ~0.5-5 kg/100 channels
Mass density: ~0.5-1 kg/liter
Specific power: ~1-2.5 W/kg [152]

These relations are comparable to the relations used for the dimensioning and sizing of EPS power management devices. Notice though that power is not used as an independent variable, but rather as a dependent variable. As independent variable, the number of data channels is used.

No relations are available for sizing and dimensioning of command processor units. As a first estimate, it can be considered to be of same mass and size as the data handling unit. For the number of channels we should in that case read the number of commanded items.

On board computers + TM encoders and TC decoders

For sizing of on board computers, TM encoder and TC decoders we may use:

Size (mass) determined by MIPS[39] ~0.7 kg/MIPS [153]

[38] With number of data channels, we mean number of input data channels. A 100 channel data acquisition system would allow collecting 100 different signals.

[39] 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.

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 speed of the processor (expressed in MIPS) is used as independent variable. Once the number of MIPS is known, the volume and the power consumption of the device can be determined.

The above relations are very preliminary. Depending on their criticality in the total design of the spacecraft, it is advised to improve the estimations whenever possible.

<u>Data storage devices</u>
Most data storage devices today are solid state recorders. In the past also tape recorders have been used, but these have become obsolete, because they have a relatively high failure rate and are quite heavy in terms of mass. Next table provides characteristics of some specific SSRs.

Table 62: Typical SSR characteristics [Surrey, 2007], [Thales, 2012], [EADS, 2014]

Parameter	**MPC**	**MUSA**	**NEMO 1050**
Data storage capability	8 Gbit	600 Gbit	512 Gbit
Peak input data rate	4 x 8 Mbps	700 Mbps	120 Mbps
Output data rate		2 x 155 Mbps	310 Mbps
Power draw (peak/typical)	9 W /6.5 W typical	131 W/94.5 W	10 W/-
Mass (excluding mounting hardware)	0.8 kg	22.5 kg	6.6 kg
Dimensions	330 x 165 x 38 mm^3	460 x 250 x 253 mm^3	365 x 135 x 235 mm^3
Temperature	-20 to + 50 deg C	-	-25 to + 50 deg C
Reliability	-	-	0.993 (7 yrs in LEO)

For mass estimation of solid state recorders, the following relation can be applied:

$$M_{rec}[kg] = \frac{C_{rec}[GByte]}{.041 \cdot C_{rec}[GByte] + 0.3128} \quad (0.1 \text{ GByte} \leq C_{rec} \leq 180 \text{ GByte}) \quad [154]$$

This relation has an R^2 value of 0.8873.

<u>Signal harness</u>
Signal harness next to power harness forms a substantial part of S/C mass, see discussion on power harness in section on electric power systems. Harness mass estimation as discussed earlier already includes the signal harness. So no further attention to signal harness mass is given here.

Problems

1. Consider the design of a CDHS for an S/C orbiting earth at 600 km. Given are:
    - Data produced:
        - Science data: 0.4 Mbps (On only during day time)
        - HK data:
            - 500 parameters measured at a frequency of 5 Hz using a 12 bits representation
            - 100 parameters measured at a frequency of 10 Hz using a 16 bits representation
    - S/C is in contact with the ground station 3 times a day for a maximum period of 10 minutes/contact with these periods being evenly spaced about the day.

    Calculate:

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.

a) Amount of science data generated per day
b) Amount of HK data per day
c) Data rate required for the down link

2. We are designing the C&DH system of a spacecraft producing 1 Mbps of science data. For housekeeping 500 parameters are measured at a rate of 1 Hz and 12 bits accuracy + another 50 at a rate of 10 Hz and 16 bits accuracy. Given that this S/C on average has 3 times a day contact with the ground station for a period of 10 minutes, you are asked to calculate:
   a) On board storage capacity given that the contact times are evenly distributed over the day
   b) Data transmission speed required when in contact with ground.
   c) Mass of C&DH system (harness + OBC + data handling system) given that average length of signal lines is 4 m and that 4 Mbps is equivalent to 1 MIPS.

   Answers: a) 28.2 Gbit or 3.65 GByte; b) 48.7 Mbps; c) ~80 kg

3. Consider a satellite in GEO collecting a mixture of housekeeping and experimental data. The data consists of:
   - 10 temperature signals. Each of the temperature signals is digitized with a maximum quantization error of 1% and a minimum sampling rate of 10 samples per second.
   - 10 voltage signals. Each of the voltage signals has to be known with a maximum quantization error of 0.1% and a sampling rate of 1 sample per second.
   - One experiment signal. The experimental information has to be known on Earth with a maximum quantization error of 0.01%, the maximum frequency of sampling is 440 Hz.

   What is the **minimum data rate** generated for the combined telemetry signal?

   Answer: Total bit rate is 600 + 90 + 12584 = 13724 bps or 13.274 kbps.

4. The following mass data for a range of solid state recorders are taken from the work of [Brown].

Table 8.10 Solid state data recorders[a]

Mission	Throughput	Memory	Power	Volume	Weight	Launch
SAMPEX	900 kbps	48 Mbit	12 W	0.6 ft^3	8.2 kg	1993
XTE	1 Mbps	1.1 Gbit	38 W	1.5 ft^3	24.9 kg	1995
Clementine	20 Mbps	1.9 Gbit	4 W	2048 cm^3	4.1 kg	1995
MGS	20 Mbps	8 Gbit	8 W	1920 cm^3	6.9 kg	1996
TRMM	2 Mbps	2.2 Gbit	42 W	2.0 ft^3	37.2 kg	1997
Hubble Space Telescope	1 Mbps	12 Gbit	12 W	0.5 ft^3	11.8 kg	1997[b]
Cassini		1.8 Gbit	8.2 W		15.8 kg	1997
LANDSAT-VII	300 n Mbps	378 Gbit	70 W	3.9 ft^3	100.6 kg	1999
EO-1	840 Mbps	48 Gbit	38 W	36 kcm^3	22 kg	1999

[a]Data in part from Webb, Ref. 8. Performance of these devices, like computers, improves rapidly. Today's performance will be better than that shown here.
[b]Added to HST on the second servicing mission.

   Use above tabulated data to develop a relationship for SSDR mass. You are free to select your own independent variable, but make sure you motivate your selection.

   Compare the data given in the table with the data earlier given in the foregoing text for a more modern SSR. What can you conclude?

Further problems for exercising upon are available via the course web pages (look for Mobius exercises) or can be found in the tutorial material.

4.7 Telemetry, tracking and command

The 815 kg craft, Voyager 1, is one of two identical machines that for many years now have been the furthest human-made objects from Earth. It is presently in interstellar space, 12 billion miles (~20 x 10^9 km) from Earth, beyond Pluto and escaping our Solar System at 1 million miles a day. A message takes 17 hours at the speed of light to reach Voyager 1. The responses, from transmitters on the probe running 23 watts of power – have the power of a billionth of a billionth of a watt by the time they reach Earth. An S/C operator explains. “Right now I’m connected to our Canberra station, and these are seven commands, set to radiate one every five minutes starting 30 minutes from now. Late tomorrow I will know if the commands have been received properly.

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 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 operations, 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, The TT&C system transmits 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.

Hereafter, we will discuss in some details the workings of telemetry and tele-command. Tracking is dealt with in a later section. First though we will provide some definitions and introduce some important requirements.

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 devices at a distance.

Requirements
Important TT&C requirements relate to 1) the size of the message (up or down) and 2) the time that we are in contact with ground (Earth ground station). With “up”, we mean here the message sent from ground to the spacecraft and with “down” the message sent from the spacecraft back to ground. The size of a message (up or down) is generally measured in bits or Bytes or a multiple thereof. When taking into account the contact time available, the bit rate can be determined immediately provided that we use all of the contact time to transmit the message. Note that data rates are usually much smaller for the

uplink than for the downlink. This is associated with the need to transport data down, whereas up, there are mostly only commands to send.

Example: Consider a spacecraft with a memory size of 10 GB. Suppose we have to downlink the full memory content in 10 minutes, it follows a bit rate of 10 GB x 8 bits/byte /10 min = 133.3 Mbps.

A third requirement relates to the quality of the transmission. A noisy or a garbled message is generally considered to be of a bad quality. This will be discussed later in more detail.

A fourth important requirement relates to the reliability of the TT&C system. It should be clear that if no data can be received from the spacecraft and/or commands can be sent to the spacecraft, the spacecraft will be severely hampered in its functioning. It is for this reason that the TT&C system generally has a number of back-up options, see later in this chapter.

Other requirements have to do with interfacing of the spacecraft TT&C system with existing ground stations (infrastructure) and the need to abide to international regulations, see later in this work as well as with the mass, size, and cost of the system, etc.

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. To this purpose, data and/or commands are <u>modulated</u>[40] on to an RF signal. Next the signal is amplified, depending on the specific design, and then transmitted over a long distance (tele) to a receiver, see Figure 104.

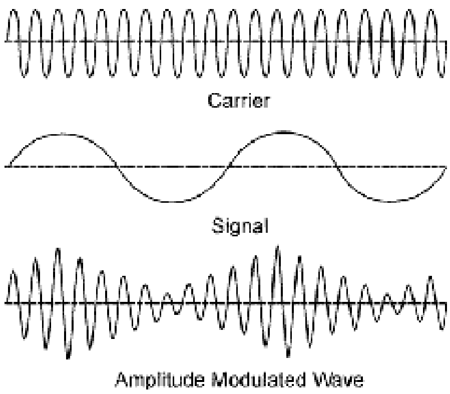


Figure 104: Principle of modulating a low frequency signal onto a high frequency carrier signal

Data and commands form the low frequency <u>information</u> signal. The high frequency RF signal on to which information is modulated is referred to as the <u>carrier</u> signal. Different ways of modulation exist, but this is considered beyond the scope of the present course.

The receiver unit receives the modulated radio waves and converts them back into a signal. Hence the basic elements of any telecommunications system are an oscillator (a crystal) that generates the carrier signal, a modulator which modulates the information signal onto the carrier signal, a transmitter which transmits the signal, a receiver receiving the signal followed by a demodulator which demodulates the signal to obtain the information signal. In addition antennas are be used to provide direction to the signal. See later for more details.

[40] In telecommunications, **modulation** is the process of varying one or more properties of a high-frequency periodic waveform, called the carrier signal, with a modulating signal which typically contains information to be transmitted. This is done in a similar fashion to a musician modulating a tone (a periodic waveform) from a musical instrument by varying its volume, timing and pitch. Different ways of modulating exist, which will be discussed in a later course.

Key characteristics of TT&C systems

Table 63 provides some characteristic data of specific TT&C systems. The table shows downlink data rates in the range 2 kbps – 105 Mbps. This range depends on the type of payload(s) carried, but also on advances made with respect to the use of different frequency bands that allow for higher data rates and/or aim to reduce interferences with terrestrial systems; see later for a more detailed discussion,

Table 63: 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

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 use X-band. Some near-Earth missions could also use X-band.
- See [FSS, table 12.2] and/or Table 10, which provides the technical profile of a typical ESA S- or X-band ground station. To keep mission costs low, frequencies used on board of spacecraft are generally chosen in accordance with frequencies in use with existing ground stations.

Table 63 also shows transmitter (output) power. 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 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.

Other key characteristics of the TT&C subsystem are its mass, size, power consumption, cost, and reliability. These will be discussed later in more detail.

Some fundamentals

The carrier signal used in radio-telecommunications is an electromagnetic wave of some frequency. Radio-frequency transmissions typically are in the frequency range from a few KHz to approximately 40 GHz. As frequency is inversely proportional to wavelength, see equation below, it can easily be found that radio-telecommunications span a wavelength range from a few cm up to a few meter.

Relation between frequency and wavelength is given by:

$$f = {}^{v}\!/_{\lambda} \qquad [155]$$

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.

As the range of frequencies of RF communication is quite large, it 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 64.

Table 64: Microwave frequency 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

Without going into detail, it is mentioned that the design of any communications equipment greatly depends on the carrier frequency selected. More details are given later.

Signals are transmitted using an analog 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 58. 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:

$$B = DR / CF / \text{spectrum utilization} \qquad [156]$$

Where:

- Spectrum utilization (or spectral efficiency) is a measure for number of bits transmitted per unit of frequency (roughly between 0.2-2 bps 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 an 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:

$$t = {}^{d}\!/_{c} \qquad [157]$$

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.

A further 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 high 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 can be found in Table 64.

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.

Starting point for the discussion is the transmitter power (*P*), also referred to as RF output 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 (*G*) and increasing the radiated power in the beam direction. There is also some loss (negative gain; L_l) 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:

$$EIRP = P\, L_l\, G_t \qquad [158]$$

G_t is gain factor of transmitting antenna as to distinguish it from the gain of the receiving antenna, see later.

The amount of power received on ground (or at the spacecraft) reduces with the distance between the ground station and the spacecraft. This is represented by the power flux density (W_f), i.e. the power per unit area at distance (*r*). Consider transmitter is point source with antenna transmitting the signal in all directions with equal strength, we write:

$$W_f(antenna\ gain = 1) = {}^{P}/_{4\pi r^2} \qquad [159]$$

When adding a directional antenna at the transmitter, the radiated power becomes more focused and the power flux increases. It follows:

$$W_f = {}^{P \cdot G_t}/_{4\pi r^2} = {}^{EIRP}/_{4\pi r^2} \qquad [160]$$

Here G_t is gain factor of transmitting antenna. It is a measure of the ability of an antenna to concentrate RF power in a desired direction, when compared to the spherical radiation from an isotropic (or omni-directional) antenna. Antenna gain is 1 when antenna transmits power equally in all directions. It increases when the total power emitted is radiated in a narrow beam. We write:

$$G_t = {}^{Surface\ area\ of\ sphere}\!\big/_{Surface\ area\ of\ antenna\ beam} \qquad [161]$$

For instance, if we would like to focus the total power emitted over a small area (say the Netherlands as seen from Space) only, we need to select an antenna with a high gain factor. Antenna gain can be adjusted by adjusting the size of the antenna. In general, it follows the larger the antenna, the larger the gain. More details on the relation between antenna gain and antenna size will be dealt with in a later course.

The power received (C) by a receiving antenna is determined using:

$$C = W_f\, A_{ant}\, \eta_{ant} \qquad [162]$$

A_{ant} = frontal area of receiving antenna
η_{ant} = efficiency of receiving antenna, typically in range 0.5-0.7 (50-70%)

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):

$$E_b = C/R \qquad [163]$$

Here R is transmission bit rate, which depends on the total amount data to be transmitted and the total time available for transmission. Critical for the design usually is the shortest time available for transmission and the maximum amount of data to be transmitted over this period. See also the earlier example given entitled: *Effect of ground contact time*.

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:

$$E_b / N_o \geq 10 \qquad [164]$$

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:

$$N = K\, T_s\, B = N_o\, B \qquad [165]$$

Here:
- k is Boltzmann 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 (P), but did not consider input power. Typically there is a difference between the two that is referred to as the transmission efficiency (η) also referred to as transmitter dc-to-RF efficiency. This efficiency is defined as:

$$\eta = {}^{P}/_{P_{in}} \quad [166]$$

It typically is in range 10-50% with an average value of 18%.

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.

mp3-geluid niet de sirene is

7 miljoen

euro betaalt Radio 538 jaarlijks al voor haar FM-frequentie

Available frequencies for commercial SATCOM (satellite communications) services are shown in Figure 105.

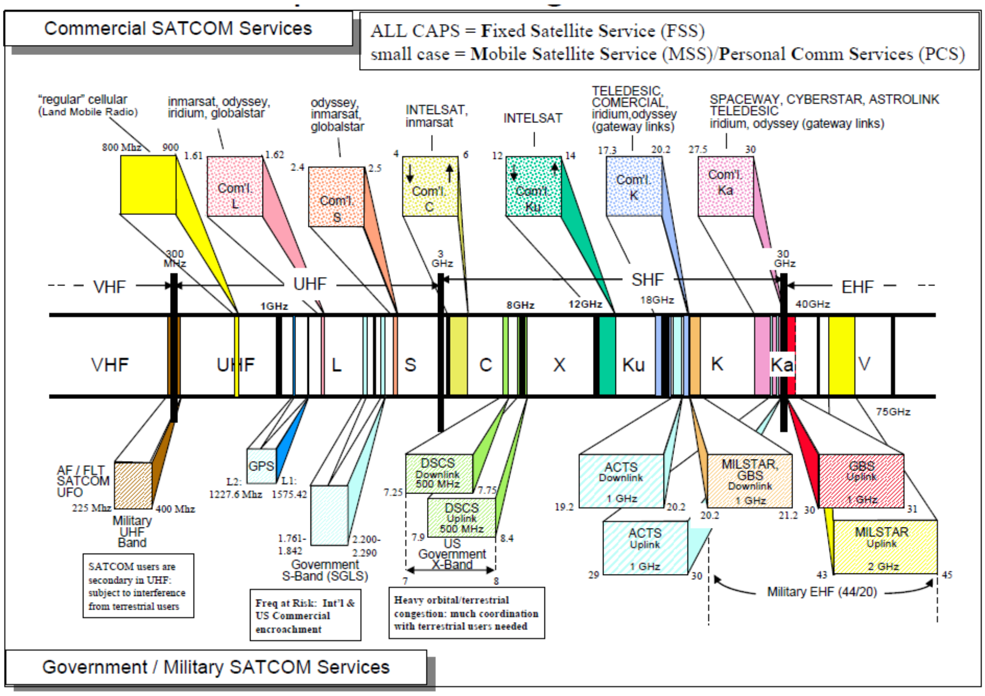


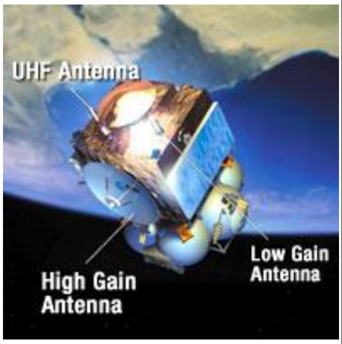
Figure 105: Overview of available frequencies for various SATCOM services (commercial services on top, government /military services at bottom)

The colored portions of the frequency spectrum are for sitcom services. The non-colored parts are also in use, but for non-space (terrestrial) related services, like radio/and video transmissions by Earth stations. A more detailed list (table) can be obtained from [Fortescue]. In case you are serious about developing a new application, it is advised to consult the ITU for available frequencies and bandwidths.

Key system elements

A description of several deep space S/C communications systems is given in Table 65.

Table 65: Overview of deep space S/C communication systems

Near	The NEAR S/C X-band communications system includes a redundant X-band transponder system which connects to the antenna system. RF output power is 5 W which is obtained through an RF amplifier connected to the transmitter side of the transponder. The antenna system is comprised of a 1.5 m (frontal) diameter High-Gain Antenna, a Medium Gain Antenna with beam width 8^o x 40^o, and two Low Gain Antennas (one on front and one on back to ensure omnidirectional reception/transmission). Uplink data rates are 125 bps for normal operations and 7.8 bps for emergency. Downlink data rate varies from 1.1 to 26.5 kbps for normal operations and less than 40 bps for cruise mode and safety mode. Total power consumption (1 transponder active) is 47 W. Total system mass consists of antenna mass of 6.9 kg (6.3 kg for the HGA) + 2 x 9.3 kg per transponder (including amplifier). [Bokulic]
New Horizons	The New Horizons X-band communications system is a fully redundant system except for the antenna system. It uses a very low power receiver (4 W consumption), next to an exciter and a 12 W RF amplifier. The antenna system consists of a 2.1 m High-Gain Antenna (39.1 kg), a 0.3 m Medium Gain Antenna (MGA), and two Low Gain Antennas. The latter are installed on different sides of the S.C to achieve an omni-directional low-gain capability. The HGA beam is only 0.3 degrees wide requiring the spacecraft to be pointed accurately at Earth. The MGA transmits a 4-degree beam for which it will be sufficient to have coarse Earth pointing.
Mars Reconnaissance Orbiter (MRO)	Two way telecommunications will be via X-band at about 8 GHz, primarily through a 3 m diameter steerable high-gain dish antenna. Two low-gain Ka-band antennas, mounted on the high-gain dish, are also available for transmission and reception. Two transponders and three traveling wave tube amplifiers allow maximum data rates of 6 megabits/sec. Total communications subsystem mass is 107.7 kg of which 65 kg is for the 3.0 m HGA with the associated gimbals and drive motors for pointing. [Taylor]
Voyager	X-band and S-band communications is provided through a single 3.66 m high-gain antenna with a low-gain antenna for backup. Uplink is 16 bps, whereas downlink can be set at different values from 40 bps to 115.2 kbps (depending on power availability). Total power usage when transmitting is about 120 W to 200 W depending on high or low power mode and total communications system mass stands at 105.4 kg including 2 transponders of 4.7 kg each, 20 kg of RF power amplifiers and 53.0 kg for the antenna system [Ludwig]
Mars Express	 Mars Express mission operates on two frequency bands, X-band (7.145 – 7.190 GHz for transmission and 8.400 GHz – 8.450 GHz for reception) and S-band (2.3 GHz). S-band is used during the commissioning phase and for back-up in case X-band link is not available. Up- and downlink telecommunications are via two redundant X-/S-band transponders and RF amplifiers (X-band transmission only) connected to the antenna system. X-band RF output power is 55-65 W (only one amplifier can be active at any one time). The S-band transmitters have an RF output of 5 W each. The antenna system consists of a 1.65 m high-gain parabolic antenna and two omnidirectional low-gain S-band antennas. The HGA has a gain factor of about 10,000 and a half cone angle of 0.65^o. The low-gain antennas protrude 40 cm from the S/C and are used at the beginning of the mission (commissioning phase). In addition, Mars Express carries an UHF communication system with two Mars lander relay UHF antennas mounted on the top face for communication with the Beagle 2 Mars lander [AIAA, 2012].

Key elements of a radio-communications system include:

- **Transmitter:** an electronic device, which generates an electromagnetic signal (the carrier) and modulates an information signal onto this signal. After amplification, the modulated signal is transferred to the antenna system for transmission. A transmitter can also be considered as a combination of a low power RF stage (also referred to as **exciter**) connected to an **RF amplifier**. The former (exciter) includes the oscillator and the modulator, whereas the latter increases the power level of the RF signal. Important RF amplifier technology types are solid state power amplifiers and travelling wave tube amplifiers.

- **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 as used in the foregoing table include:

- **Transceiver:** A device that has both a transmitter and a receiver, which are combined and share common circuitry. It usually has lower mass and volume than a separate transmitter and receiver.
- **Transponder:** An interconnected receiver – transmitter (or exciter) unit which forms a single communication channel between receiver and transmitter antennae in a satellite. In communications satellites, it receives signals in a certain frequency range, amplifies them and retransmits them on a separate frequency range. The term transponder comes from that it acts both as a transmitter and a responder. For other spacecraft it usually acts as a transceiver, but to allow for ranging using an RF ranging signal it works as a transponder and hence one rather uses transponder.

Various elements introduced in the preceding are shown in Figure 106.

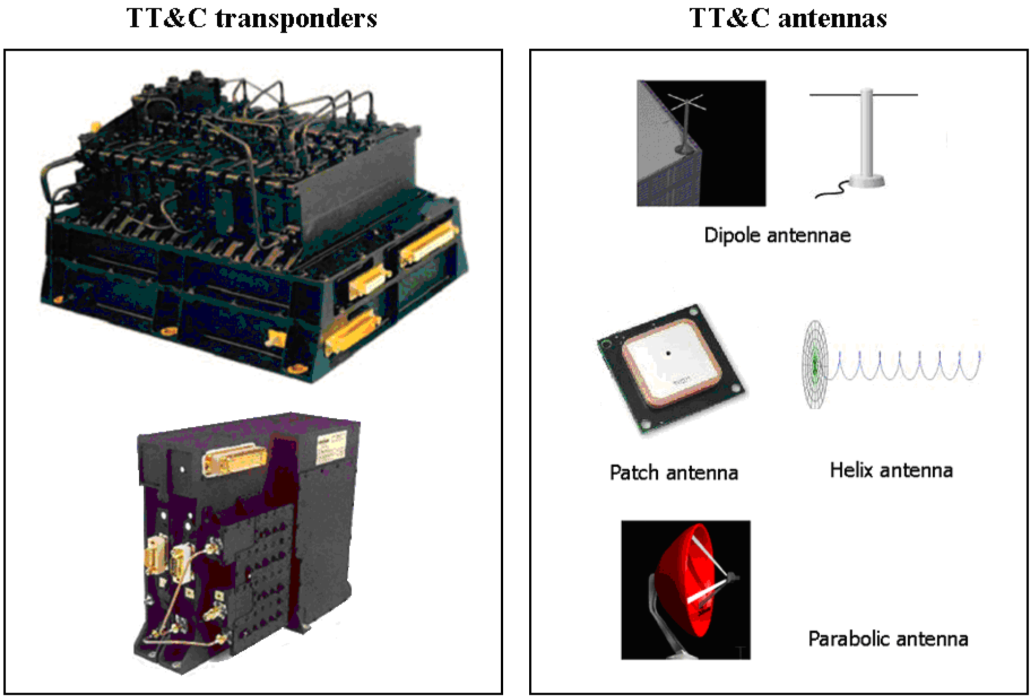


Figure 106: TT&C transponders and antennas

From the figure we also learn that different types of antennas exist, which all allow for a range of antenna gains to be realized. From top to bottom the antennas are more directionally sensitive with increasing gain (i.e. better ability to detect a weak signal). The dipole antenna is actually an omnidirectional antenna with an antenna gain close to 1. The patch antenna as well as the helix antenna can have a beam width of between 30 to 180 degrees, which indicates an antenna gain in range 2 to 8. Parabolic antennas can have a very narrow beam width of less than 1 degree allowing for gain factors of 10,000 and higher depending on their size (the larger the antenna, the higher the gain). The advantages and disadvantages of Low Gain and High Gain antenna types are given in Table 66.

Table 66: 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 • Large and heavy	• Omni-directional, no need to point • Emergency commanding • Ranging • Light weight

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 107 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 as well as a radio frequency network and diplexers. The interface with the rest of the satellite is via the on-board Data Handling (OBDH) subsystem. The command decoders together with the OBDH subsystem are also referred to as the Command and Data Handling Subsystem, see previous chapter. The TT&C subsystem without command decoding is also simply identified as the communications subsystem.

- The uplink carrier with the tele-command (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.
- Coaxial cables and/or wave guides and diplexers ensure that the RF power is properly transferred from transmitter to antenna or from antenna to receiver.
- 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).

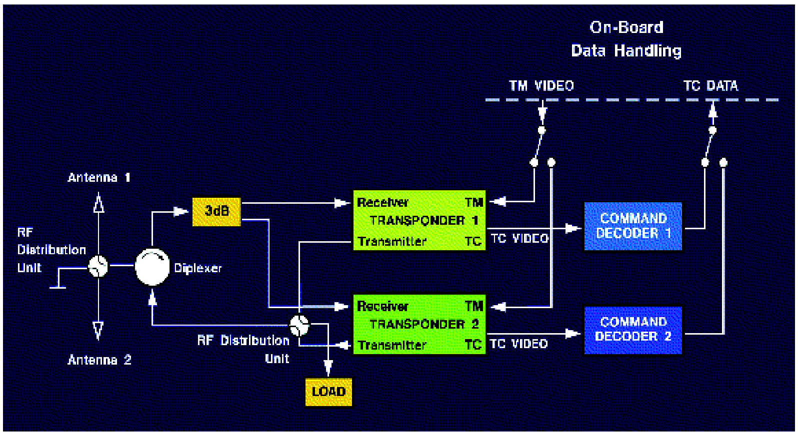


Figure 107: Typical TT&C architecture

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 (isentropic transmission). 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, it may be advantageous to use a high gain antenna on board. Configuration issues include:

- Direct line of sight between transmitting and receiving station
- The antenna needs to be positioned / oriented to adequately cover the receiving/listening area.
- HGA may be mounted on mechanism to allow for steering or whole spacecraft is pointed to ensure proper pointing of the HGA.
- LGAs may be added to allow regaining control of the spacecraft after a failure of the ADCS.
- Transmitting elements of a transmitter, transceiver or transponder usually consumes a lot of power (at low efficiency) and therefore generates a lot of heat. They are therefore usually mounted on the cold side of the satellite.
- Short line length between the various elements to reduce losses in the system
- Typically all components are <u>redundant</u> 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
- Having a separate transmitter for payload data (high data rate) and telemetry/tele-command (low data rate) with the possibility of the TM/TC transmitter acting as a back-up for the former.

Dimensioning and sizing
A first estimate of the TT&C (or communications) subsystem mass, size, power, cost and reliability characteristics can be obtained based on historical data of comparable spacecraft even without knowing what the exact performances for the system should be. Typical such data have been collected for instance in appendix D. Using the data from this appendix, you may find that, depending on the specific spacecraft, the communications subsystem (not the communications payload of a communications satellite) accounts for about 10-12% of all serious bus failures. It also represents a significant part of total bus cost (10-15%) and bus mass (2.5-10%). It also consumes about 10-20% of total bus power.

Clearly this data is not highly accurate (partly as the data apply to different types of spacecraft), so proper margins should be selected. Of course one can also collect additional historical data to try improving the accuracy

Another way to allow for a more detailed estimation of the subsystem characteristic mass, size, power, cost and reliability, etc. is by determining the RF output power and antenna gain (and hence size) that allows for proper reception and transmission of the required data rates. For this the theory presented earlier in this section can be used. In combination with breaking up the system in its main components and dimensioning (or selecting) each of these using historical data for similar components. For instance, we can consider the communications system to consist of one or more transceivers/transponders (with or without additional RF amplifiers), antennas and miscellaneous items. In the remaining text of this paragraph on dimensioning and sizing a first example of such a model is given.

At low RF powers (up to about 10 W RF power), the RF amplifier is usually built in the transponder/transceiver. In that case, transponder/transceiver mass may be estimated using:

$$\mathrm{M}_{trans}[\mathrm{kg}] = \frac{P}{P_{sp}} \frac{[\mathrm{W}]}{\left[\frac{\mathrm{W}}{kg}\right]} \qquad [167]$$

$$\mathrm{V}_{trans}[\mathrm{liter}] = {\mathrm{M}_{trans}\ [kg]} \Big/ {\rho_{trans}\left[\frac{\mathrm{kg}}{l}\right]} \qquad [168]$$

M_{trans} is mass of transceiver/transponder, V_{trans} is transceiver/transponder size (volume), P = RF output power of transceiver/transponder, P_{sp} = specific power and ρ_{trans} is transceiver/transponder mass density. When determining the mass of the transponder/transceiver system, one must take into account redundancy as transceivers and transponders have a relatively high failure rate. For instance, on board of a communications satellite with say 20 transponders, roughly 1/4th of these acts as back-up for failing transponders. How much redundancy is needed depends on the specific design and on how critical the system is for the mission success. Typical values for P_{sp} and ρ_{trans} can be taken from literature. For simple low power transceivers/transponders (RF power typically below about 10 W), we find:

$$P_{sp} = 2.9 \text{ W/kg (SSD} = 2.8 \text{ W/kg)} \qquad [169]$$

$$\rho_{trans} = 0.5 - 1 \text{ kg/liter} \qquad [170]$$

For high RF power transceivers/responders (typically starting at about 1 W RF power) one can consider determining the mass of the RF transceiver/transponder as the sum of the mass of a low power transceiver/transponder[41] + the mass of one or more separate high power amplifiers with typical RF power levels in range 10 W to 300 W. For this two different technologies are available, including Travelling Wave Tube Amplifiers (TWTAs) and Solid State Power Amplifiers (SSPAs). TWTA's in general allow for reaching higher power levels at higher efficiency levels than SSPAs especially at the higher transmission frequencies. SSPAs though allow amongst others for a smaller and lighter design. In case of using TWTAs, the following relation may be used to determine the mass of the RF power amplifier including the electric power conditioning device needed:

$$M_{TWTA}[\text{kg}] = 0.07 \text{ P[W]} + 0.634 \qquad [171]$$

Data for the above relation has been taken from [Bosch]. No relation is available for Solid State RF Power Amplifiers (SSPA) mass. Also no relations are given on the size of RF power amplifiers, whether TWTA or SSPA. This is left for the reader to determine these for him/herself depending on the need.

The antenna system like the transponder/receiver system may consist of multiple antennas. All antennas are sized to obtain a certain gain. This gain is generally determined by the size of the antenna. This will be discussed in more detail in a later course, but in general the larger the size of the antenna, the larger the gain. For mass estimation, the mass is determined based on the assumption that the mass of the antenna is linearly proportional to the size of the antenna, according to:

$$M_{antenna}[\text{kg}] = \rho_{antenna}\left[\frac{\text{kg}}{\text{m}^2}\right] \text{x } S_{antenna}[m^2] \qquad [172]$$

Here $\rho_{antenna}$ is specific antenna mass. Specific antenna mass depends on the type of antenna, its specific design (solid dish or mesh antenna), whether feed assembly and mounts are included, whether it is movable, etc. The antenna surface depends on the type of antenna. For a parabolic antenna, usually the frontal area is used.

For parabolic antennas, we find an average specific antenna mass as given by the next relation:

$$\rho_{antenna} = 5 - 8 \text{ kg}/m^2 \qquad [173]$$

Still values as low as 1.5 kg/m^2 and as high as 11 kg/m^2 are also reported. Of course, you are advised to find out how accurate the above values are by comparing with results found in literature.

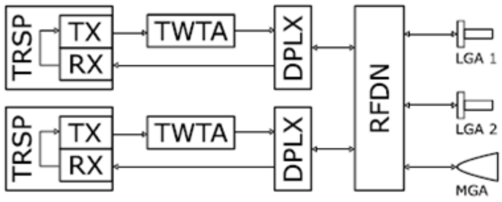
For other types of antennas, specific antenna mass needs to be taken from literature. As an example, we give here the data for a small S-band helix and patch antenna as developed by [Surrey].

	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

[41] Output power of the exciter part of a transponder may be very limited and hence the relation [161] may give an unrealistically low value. Instead use a typical transponder mass of 2 kg as a first estimate.

From this data, we find that for a helix antenna taking the area of the helix antenna as the product of the length of the helix times the diameter of the helix a helix antenna mass density of 11 kg/m^2.

Next to transmitters and receivers or transceivers/transponders and antennas, we also need to take into account some miscellaneous items, like fixings, mounting plates, switches, cables and various other items. As a first attempt to account for cables, switches and other miscellaneous items, it is advised to add an arbitrary 20-30% to the mass of the system. Later in the design, when more details on the components are available, this number can be verified and if needed updated. Same procedure can be followed for power, size and cost estimates. Once all the estimates are in, typical overview tables can be generated, like the one shown in below figure.



Communications subsystem overview

	Mass (kg)	Mass incl. margin (kg)
High power amplifier 1	2.3	2.42
High power amplifier 2	2.3	2.42
Low Gain Antenna 1	1	1.1
Low Gain Antenna 1	1	1.1
Medium Gain Antenna	2	2.2
Transponder 1	3.3	3.47
Transponder 2	3.3	3.47
Radio frequency network + diplexers	5.8	6.88
Medium Gain Antenna pointing device	10	12
Total	31	35

	Power (W) active	Power (W) standby
High power amplifier 1	70	13
High power amplifier 2	70	13
Low Gain Antenna 1	0	0
Low Gain Antenna 1	0	0
Medium Gain Antenna	0	0
Transponder 1	55	0
Transponder 2	55	0
Radio frequency network + diplexers	0	0
Medium Gain Antenna pointing device	20	5
Total	270	32

Figure 108: Example communications subsystem TT&C preliminary design [ESA, July 2018]

Note that the system as given in figure has in total three antennae of which two low gain antennae and one medium gain antenna, two diplexers, two TWTA and 2 transponders (each consisting of a receiver (RX) and a transmitter (TX)). The lines interconnecting the transmitters/receivers with the antennae make up the radio frequency network. For the particular spacecraft considered, the RF network has a mass of about 6 kg or about 20% of the total system mass. From the power budget in the figure, it is clear that two states are distinguished being active and standby. Clearly the power consumption varies with the state. To be able to compute the required power it would be good if it is clear what duration the system is active and the duration it is on standby (transponders are never switched off as this impedes the life).

As a final remark to this section, we note that the numbers derived using the above described methods should not be considered as exact. Large discrepancies with real systems should still be considered, thereby stressing the continuous need of developing more exact estimation methods and/or taking appropriate margins. For now, it is advised to take a margin of at least 30% on total communications system mass.

Example communications subsystem design:

Consider the TT&C system of a S/C in low lunar orbit. For this TT&C system are given:

- *Input power: 10 W*
- *DC to RF (transmission) efficiency: 20%*
- *Transmission bandwidth: 2-2.3 GHz (in S-band)*
- *Downlink data rate: 0.3 Gbps*
- *Transmit antenna gain G_t: 100*
- *Receive antenna diameter on Earth: 10 m*
- *Receive antenna efficiency: 0.5 (50%)*
- *System noise temperature: 135 K*

Determine for this system the following properties:

A. *Effective Isotropic Radiated Power (EIRP)*
B. *Minimum flux W_f received on ground (Earth)*
C. *Power received on ground (C)*
D. *Noise power (N)*
E. *E_b/N_o*
F. *Total mass in case we use for reasons of redundancy 2 transceivers/transponders. This in combination with a single parabolic antenna of frontal diameter 1.1 m (same antenna size as used for Clementine S/C).*

Ad A.) *EIRP is obtained by multiplying the RF output power with the transmit antenna gain. It follows:*

$$\text{RF output power is } 0.2 \times 10 \text{ W} = 2 \text{ W}$$

$$EIRP = 2 \text{ W} \times 100 = 200 \text{ W}$$

Ad B.) *Moon is at a maximum distance of 407000 km of Earth. This gives for the minimum flux received on ground:*

$$W_f = 200 \text{ W}/(4\pi \times (407E6 \text{ m})^2) = 9.6E\text{-}17 \text{ W/m}^2$$

Ad C.) *Power received on ground by receiving antenna of diameter 10 m taking into account an antenna efficiency of 50% is:*

$$C = 9.6E\text{-}17 \text{ W/m}^2 \times (\pi \times (5 \text{ m})^2) \times 0.5 = 3.77E\text{-}15 \text{ W}$$

Ad D.) *With a noise temperature of 135 K and a bandwidth of 0.3 GHz, follows a noise power:*

$$N = 1.38E\text{-}23 \times 135 \times .3E9 = 5.6E\text{-}13 \text{ W}$$

Ad E.) *E_b/N_o = (3.77E-15 W/300 Mbps)/(5.6E-13 W/0.3 GHz) < 1.*
It follows that no reception is possible. We should either increase the size of the receiving antenna, the transmit antenna and/or increase the transmitter power.

Ad F.) Mass estimation

- *Transceiver/transponder mass is 10 W/2.9 W/kg = 3 kg*
- *Size is 3 kg/0.75 kg/liter is 4 liter*
- *Since we use 2 transceivers/transponders this gives a total mass of 6 kg and volume of 8 liter.*
- *Using an antenna area to weight ratio of roughly 8 kg/m^2 it follows an antenna mass of 0.95 m^2 x 8 kg/m^2 = 7.6 kg,*
- *To take into account the RF network we add 30% of the summed mass of transponder and antenna (0.2 * (2 x 3 kg +7.6 kg) = 4.1 kg, which brings total system mass to about 17.7 kg excluding margin. Adding a margin of 30%, gives a total mass of 23.0 kg.*

Problems

1. Give an account of the main TT&C functions and explain the importance of these functions for the spacecraft.

2. Explain the importance of frequency bands, bandwidth and spectrum utilization for the design of a spacecraft.

3. Explain in your own words:
    a. The difference between carrier signal and information signal?
    b. Why only designated frequencies can be used or what is the reason that led to setting up ITU?
    c. What is modulation?
    d. What is meant with the bandwidth of a telecommunications signal and how does this relate to the carrier frequency?
    e. What is meant by a clear line of sight for a communications antenna?
    f. What is meant by spectrum utilization?
    g. Why should the energy per bit to noise density ratio be in excess of 1?

4. Complete the following table.

Frequency	**Wavelength**
150 MHz	
400 MHz	
1.75 GHz	
	0.15 m
	0.05 m
	0.02 m

5. Consider an S/C in orbit about Mars. Given is that Mars is located at an average distance of 1.52 AU from the Sun and Earth is at an average distance of 1 AU from the Sun. What is the maximum time needed for a telecommunications signal to travel the distance between Earth and Mars (*back and forth*)?

    Answer:41.9 minutes

6. Consider a satellite in geo-stationary orbit. Downlink data rate is 6410 bits per second. The on board satellite transmitter has an RF (output) power of 30 Watt.
    a. Given a spectrum utilization of 0.5 bit/Hz, determine the required bandwidth (in kHz);
    b. Calculate the received power flux density W/m^2) at a distance of 36,000 km from the transmitter (transmit antenna gain is 1 and line losses may be neglected);
    c. Same question, but now in case we use a transmit antenna with an antenna gain of 3000;
    d. Calculate the power received on ground (Earth) by an antenna of 1 m diameter and an antenna efficiency of 60%;
    e. In case we have a noise temperature of 300 K, determine the noise spectral density.
    f. Calculate the energy per bit to noise spectral density ratio.

    Answers: a) 12.820 kHz; b) 1.84E-15 W/m^2; c) 5.33E-12 W/m^2 ; d) 2.61E-12 W; e) 4.14E-21 W/Hz; f) ~98000.

7. This problem deals with the design of the communications (COMMS) subsystem of a spacecraft that is required to communicate with a ground station at some given distance from the spacecraft. More specifically, this problem focusses on determining the optimum spacecraft RF transmission power and transmit antenna size for a given transmit frequency, ground station (antenna size and antenna efficiency), bit rate, and feed line loss factor.

A first design of the communications system has been made allowing for an Eb/No ratio of 10. It is based on the use of a transceiver/transponder (hereafter simply referred to as transponder) with an RF output power of 10 W and a parabola type of antenna with a frontal diameter of 1 m. Furthermore are given:

- Mass of single transponder: $M_{transponder} = 0.06*P + 2.1$
- Transmit antenna gain: $G = 150*D^2$
- Antenna mass: $M_{antenna}$ = $=3.94*D^2 + 2.4$

All masses are in kg, P is RF power output in Watt, and D is frontal diameter of dish antenna in meter.

a. Determine for the baseline communications system antenna mass, transponder mass and total communications system mass given that we have a back-up transponder.

To investigate whether the mass of the communications subsystem can be reduced you consider varying the antenna size over the range 0.3-1.1 m (with minimum step size 0.2 m), while keeping the E_b/N_o ratio constant.

b. Determine the antenna diameter (and associated RF output power) for which the COMMS subsystem mass is lowest. Hint: Consider using a graphical solution approach where the mass of the communications subsystem is plotted along the y-axis and the diameter along the x-axis.

When changing the RF output power it follows that for a constant transmitter efficiency, also the electric input power of the COMMS subsystem changes. This will also have an effect on the electric power output of the electric power subsystem (EPS) and hence the mass of the EPS. Given is now that the EPS power output changes with the same amount as for the COMMS subsystem (transfer of power from EPS to COMSS is essentially loss-less), i.e. if electric input power of the COMMS increases with 1 W than also the output power of the EPS increases with 1 W, an electrical to RF power (transmitter) efficiency of 20%, and an EPS system specific power of 10 We/kg.

c. Determine graphically the value of the antenna diameter and associated RF output power for which the total mass of EPS and COMMS is lowest.
d. Discuss any other reasons that should be taken into account when selecting a high or low gain antenna other than the trade made in the foregoing.

Answers: a) 11.74 kg; b) 0.7 m (0.72 m is more exact); c) approximately 1.1 m.

Further problems for exercising upon are available via the course web pages (look for Mobius exercises) or can be found in the tutorial material.

4.8 Navigation (not for examination)

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.

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 act of measuring the direction and magnitude of spacecraft motion or the use of consecutive observed locations of the same target into tracks. Important parameters to consider 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

How navigation works

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 109. 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).

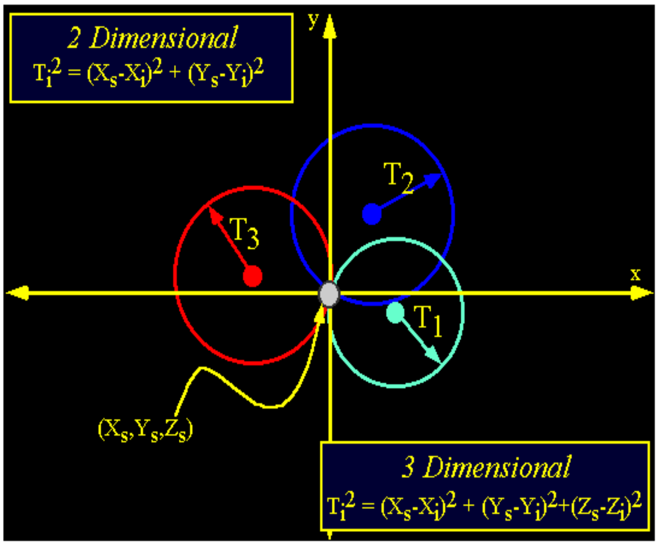


Figure 109: Principle of positioning

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 [Fortescue] 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 relationship [157]. Travel velocity can be determined using:

$$f_w = f_b\sqrt{\frac{c+v}{c-v}} \qquad [174]$$

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 110*** 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 111, whereas Table 67 and Table 68 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 110: Typical components of DORIS SC navigation system

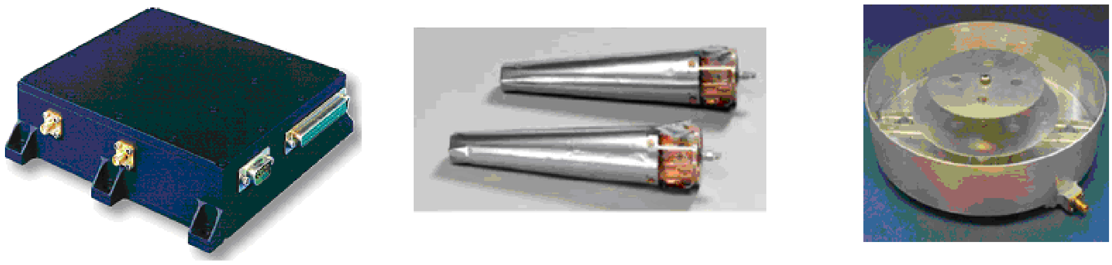


Figure 111: Typical GPS receiver with accompanying antennas

Table 67: 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 68: 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

From the above data, it should be clear that a simple, but accurate, mass estimation relationship for GPS receivers (as for any communications receiver) is not easily found and may require not only more insight in the different technologies in use today (and in the past), but also in the packaging methods applied, like printed circuit board (PCB) only or PCB mounted in a metallic housing (to safeguard against damaging).

4.9 Other subsystems (not for 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 112. 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 112: Astronaut in space with ECLSS integrated in backpack and suit (courtesy NASA)

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 113. A typical minimum overall system reliability goal for the Command Destruct System is 0.999 at a 95 percent confidence level.

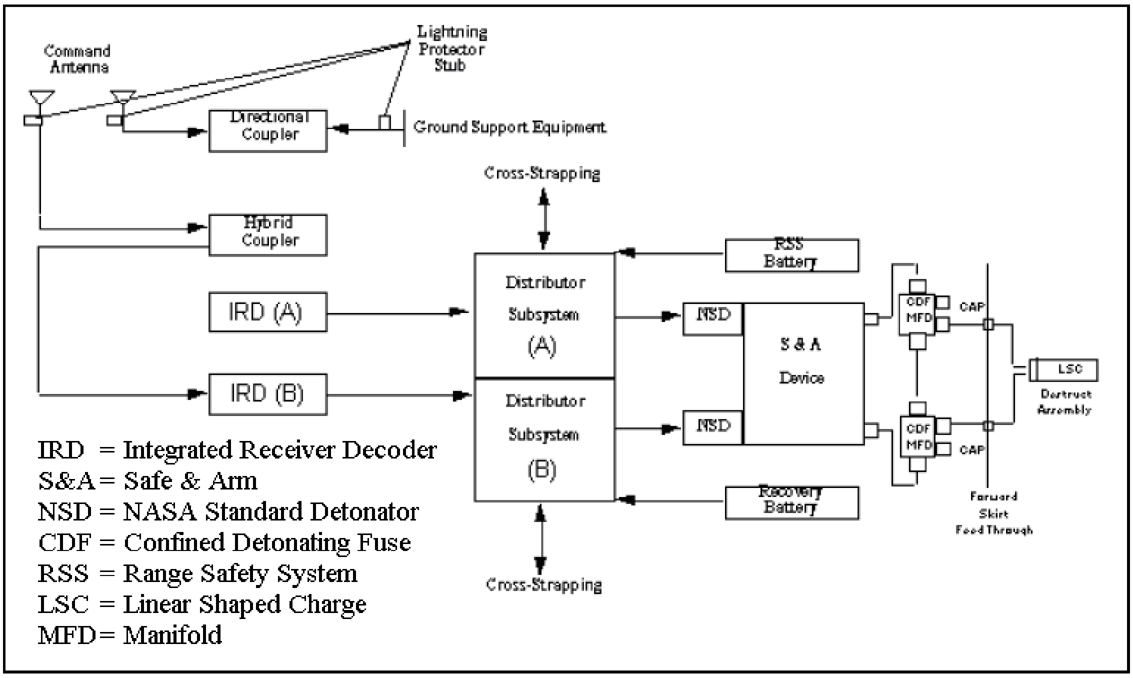
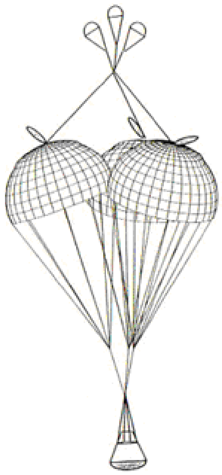


Figure 113: SRB Command Destruct System Functional Diagram (courtesy NASA)

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 114 and Figure 115.



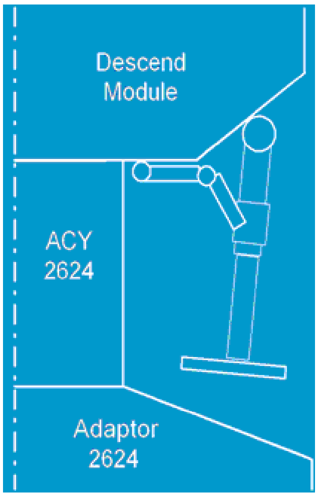


Figure 114: Typical components of a landing system

Figure 115: Vehicle Landing on Mars (courtesy NASA)

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 116 and Figure 117, respectively.

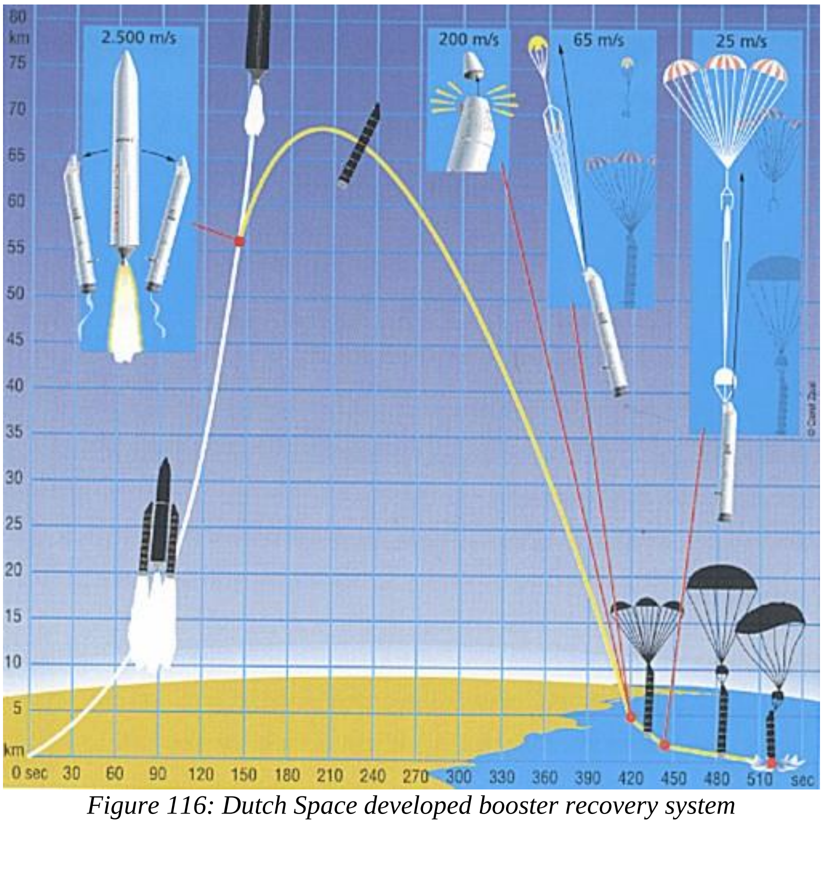


Figure 116: Dutch Space developed booster recovery system

Figure 117: 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 118 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

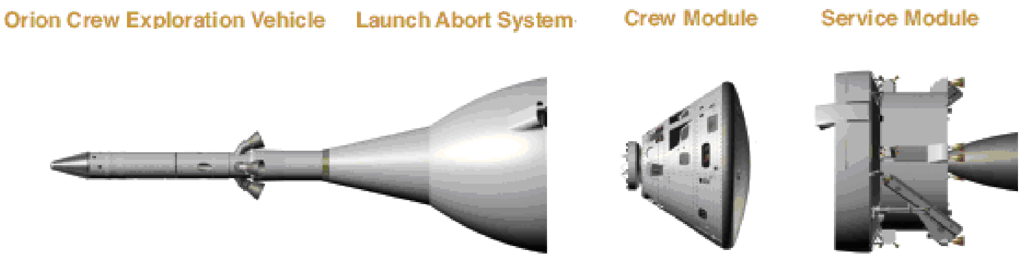


Figure 118: Orion crew exploration vehicle launch abort system (courtesy NASA)

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 parameters; 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 problems (in this work) + Mobius 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	9.2-10.2 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)	2.0-2.6 km/s (2.2 km/s for Apollo ascent stage)
LLO to Moon surface	1.6-2.9 km/s (2.5 km/s for Apollo descent stage)
Mars Surface to low Mars orbit	4.1-5.7 km/s
Low Mars orbit to Mars surface	4.7 km/s (atmospheric drag helps to slow down)

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 Mars transfer orbit (MTO)	3.4 km/s (depends on relative position of Mars)
MTO to Mars orbit insertion:	
Highly elliptical orbit	1.4 km/s
Circular orbit	2.5 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

1. Values in part taken from Space Mission Analysis and Design by Larson et al
2. Additional values taken from Payload and mission description definition in space sciences by Martinez Pillet, V., Aparicio, A. and Sanchez, F. Cambridge University Press (December 26, 2005)

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
Mars Transfer Orbit (MTO) to Mars Orbit Insertion (MOI)	~6.5	
Earth escape to Jupiter sphere of influence (using gravity assist)	~16.2	1724 days

3. Transfer or trip time for constant thrust spiral 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.
4. Δv for LEO to GEO transfer orbit calculated using T.N. Edelbaum's equation: $\Delta v = SQRT(v_1^2 - 2 v_1 v_2 \cos(\pi/2\ \Delta i) + v_2^2)$ where v_1 is circular velocity initial orbit, v_2 is circular velocity final orbit, and Δi is plane change in degrees.
5. Values for LEO to Earth escape taken from Rocket Propulsion and Spaceflight dynamics, by Cornelisse, Schoyer & Wakker, for jet exhaust to initial circular velocity ratio equal to 10.
6. Value for GTO to Lunar Orbit taken from SMART-1 by D. Racca.
7. Value for LEO to Low Lunar Orbit taken from Optimized Low-Thrust Transfer for Space Tugs by Pukniel.
8. Values for LEO to Lunar/Mars orbit taken from NASA-JPL.
9. Value for MTO to MOI from the work of G. Mingotti et al. entitled "Earth–Mars Transfers with Ballistic Escape and Low-Thrust Capture", presented at the 61st International Astronautical Congress, Prague, Czech Republic, 27 September – 1 October 2010.
10. Value for Earth escape to Jupiter from Low-Thrust Trajectories to Jupiter via Gravity Assists from Venus, Earth, and Mars, by Masataka Okutsu, Chit Hong Yam, and James Longuski, AIAA 2006-6745, 2006, published online 2012.

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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. Such data can be used in the design of spacecraft to make first guestimates. It may also help to find differences between different spacecraft, such as those illustrated in below figure.

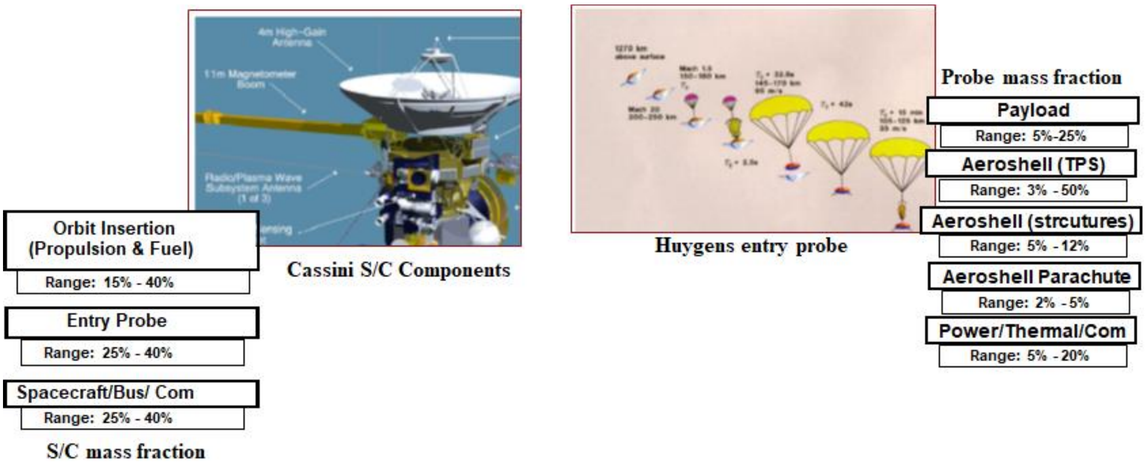


Figure shows Cassini orbiter and Huygens entry probe as well as typical mass distribution for planetary orbiter vehicles like Cassini (carrying an entry probe) and typical entry probes like Huygens. Notice the differences in shape (Huygens is much more aerodynamically shaped) as well as the differences in the mass breakdown (distribution). Note the almost complete absence of propulsion and propellant (fuel) for the probe as the latter is essentially un-propelled.

Spacecraft overall Mass, Cost and Life Data

In the table on the next two pages mass, cost and life data of specific 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 ae1110-II. 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

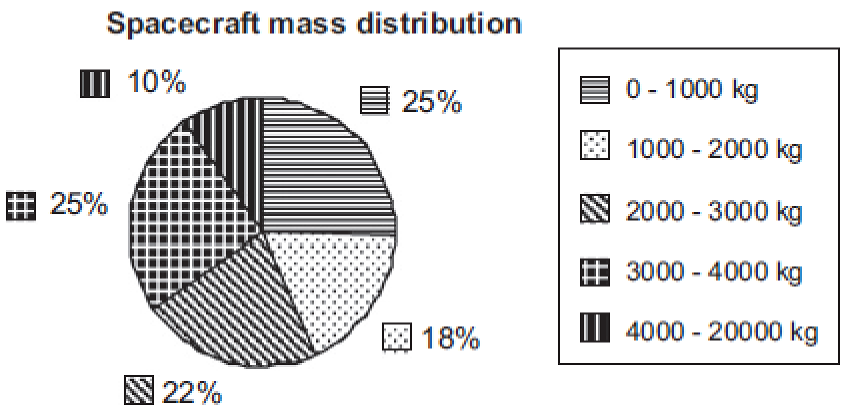
Application	Mission	Satellite	Operator	Bus	Orbit	Life	Launch mass [kg]	Dry mass [kg]	Cost [M$]	Cost FY 00 [M$]	Year	Specific cost [k$/kg]
Communications	FSS	Thaicom	Shinawatra	HS 376 L	GEO	13.5	1080	436	50	59.7	1991	137.0
Communications		Apstar 1A	APT satellite Co	HS 376	GEO	10	1383	557	80	87.1	1995	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

Next table shows dry mass data of some large (total mass in excess of 500 kg) satellites.

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: Dry mass and payload mass data 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 (total mass below 500 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

Typical spacecraft power data for a range of large geostationary telecommunications spacecraft are collected in next table.

Table 1: Total spacecraft power (in Watt EOL) as produced by the power source 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.

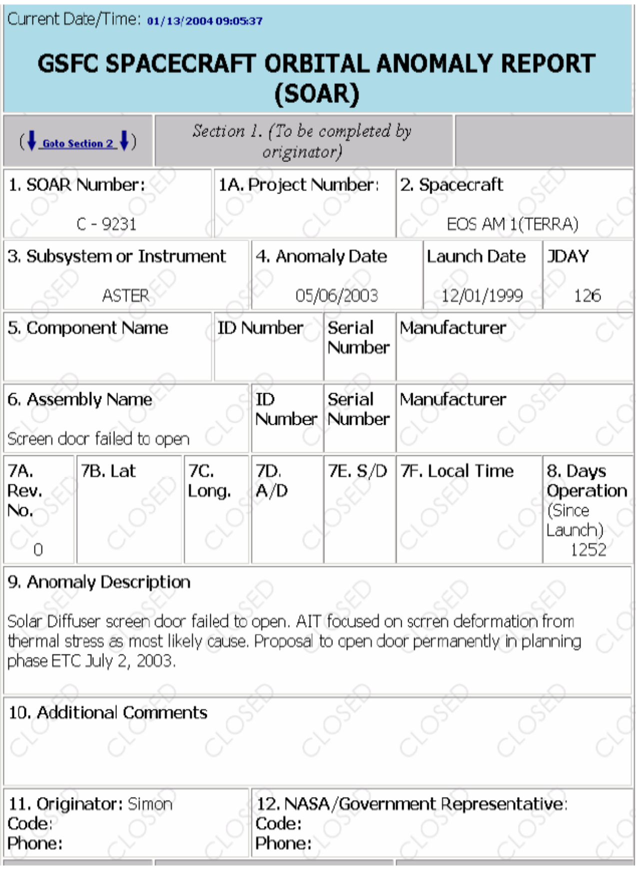
Spacecraft failures and failure data

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 categories that usually lead to an insurance claim, whereas only failures from the final category are counted as real failures. Focusing on the latter category only, we find 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:

$$R = (e)^{-(\lambda \cdot t)} \qquad [1]$$

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

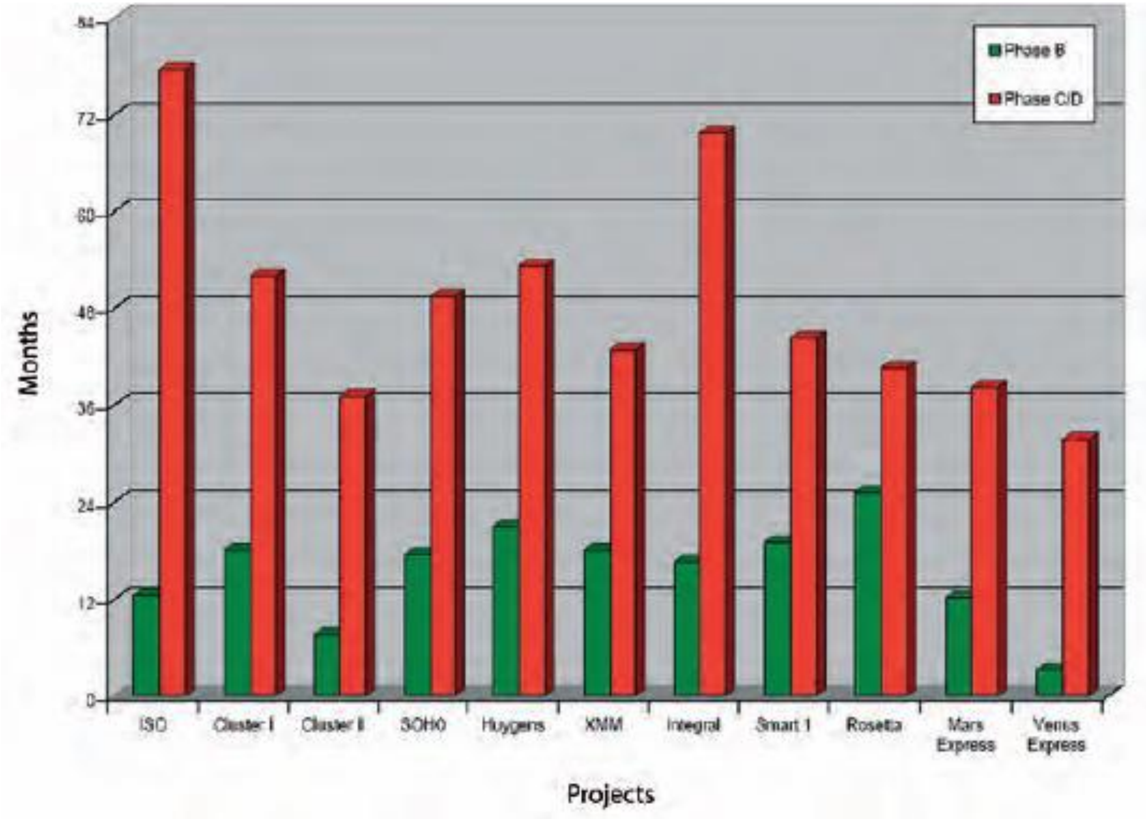


Figure 1: Spacecraft development time (ESA)

From the data given in the above figure, it follows that:

- Phase B (detailed design) for ESA typically takes 13.8 months (~1 year) with an SSD of 5.4 months (~0.5 year).
- Phase C/D (development) for ESA 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 (pre-design) 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 equipment. The reduction in phase C/D duration is much less than for the phase B. This is attributed to the need to adapt all equipment 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.

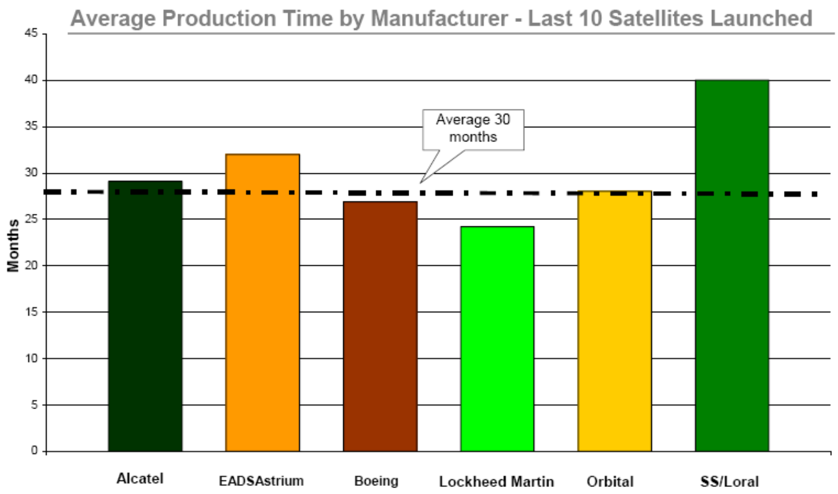


Figure 2: Commercial GEO communications satellite production time [Futron 2004]

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".

References

1. Remez, J.W. and Simon, P.S., Orbital anomalies in Goddard Spacecraft for Fiscal Year 2003, NASA GSFC, August 2005.
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3. Sultan, N. CSAT, Cheaper, faster is better only if safer: Need for "FIRST"®: "Failure Information Reliability Space Tool " To retrieve & analyze orbital failures.
4. Tafazoli, M., A study of on-orbit spacecraft failures, Acta Astronautica 64 (2009) 195–205.
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 characteristics (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.

The data used to generate these relationships has been taken amongst others from the data collection in appendix B or data was collected separately. In the latter case, the vehicles used are explicitly mentioned in footnotes that go with the various relations. Some relationships were taken from other publications. This is indicated with the relation. If available, the number of data points used is included. Note that for most relationships taken from literature the underlying data is not known. This is because it takes a lot of time to collect the data and most people do not like to give away their data for free.

As each relationship (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. If not available, this adds some uncertainty on the applicability of the relation.

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, relative standard error (RSE) and/or the R-squared[2] (R^2) value, again provided that this data is available. An RSE of 50% on spacecraft mass means that 32% of the vehicles have a mass that is outside a range defined by the estimated value (MLE) ± 50%. It also means that only 16% of the vehicles have a mass that is larger than the MLE + 50%. RSE information can be used to determine appropriate margins (or contingencies) on estimated values. In general, an RSE of 50% is considered acceptable (OK) for preliminary studies, but in later stages of the design, the accuracy of the estimations should be improved.

As a general limitation to the vehicles considered in this section, it is mentioned that propellant mass for all spacecraft is in range up to about 50% of loaded vehicle mass unless specified otherwise. It is the author's experience that in case propellant mass becomes too high, it is better to consider including in the design a separate kick stage that provides for the large manoeuvres. This will allow for separating the kick stage before arriving on target, thereby reducing the propellant consumption on station.

WARNING

Be cautious to apply the relationships for vehicles that are not part of the specific category intended as this may lead to large errors in the estimated parameter value. To improve the understanding of the category of vehicles for which the relations are valid, vehicles used to determine the relationship have been included in footnotes that go with the relationships.

Additionally, it is mentioned that kick stages are not addressed in this section as these are considered special vehicles that have propulsion as the primary function. All other functions are dealt with by the S/C carried into orbit, although here also some exceptions can be made.

[2] Statistical measure of how well a regression line approximates real data points; an R-squared of 1.0 (100%) indicates a perfect fit.

Spacecraft vehicle level estimation relationships

Vehicle mass

Vehicle mass can be estimated directly using the relations from *Table 1* (loaded/gross mass) or by adding the results from *Table 2* (dry mass) and *Table 3* (on station propellant mass). When using the relations, make sure that you understand the differences between the various vehicles mentioned and be careful when using the relationships outside their applicable range. Note that entry vehicles are only included in the first table as propellant mass on board of these vehicles is usually low (less than 10% of spacecraft loaded mass) and hence loaded mass and dry mass are about the same.

Table 1: Spacecraft 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
RSE	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
RSE	34.7%
Manned entry vehicles[3]	
$M_{on\ station} = 1.861\ M_{PL} + 1952.1$ kg	[3]
Source	Zandbergen (6 data points)
Payload mass range	20-6000 kg
R^2	0.745
RSE	44.2%
Unmanned entry vehicles[4]	
$M_{Loaded} = 1.404\ M_{PL} + 224.3$ kg	[4]
Source	Zandbergen (10 data points)
Payload mass range	15-300 kg
R^2	0.322
RSE	56.3%
Planetary orbiters not carrying probes[5]	
$M_{Loaded} = 15.909\ M_{PL} + 24.7\ kg$	[5]
Source	Zandbergen (21 data points)
Payload mass range	8-180 kg
R^2	0.6686
RSE	95.5%[6]

[3] Vehicles include Mercury capsule, Gemini capsule, Apollo CM, Orion CM, Soyuz DM and Apollo lunar module.
[4] Vehicles included are a.o. Viking Mars lander, Mars Pathfinder, and Rosetta.
[5] Vehicles include Mangalyaan, NEAR, SMART-1, Clementine, Mars Reconnaissance Orbiter, Mars Global Surveyor orbiter, Juno, Venus Express, Messenger, Pioneer 13, MAVEN, Mars Global Surveyor-98 orbiter (Mars Climate Orbiter), etc. Note that these are all pure orbiters, meaning none of them carries one or more probes that is/are to descend into the atmosphere.
[6] The very high RSE value is mainly because of Mangalyaan. This is attributed to that the spacecraft has a relatively large propellant load of about 70% of the loaded mass. Without Mangalyaan, the RSE for this relationship reduces to 46%. This result shows it is worthwhile to estimate dry mass and propellant mass separately.

Table 1: Continued

Planetary orbiters carrying one or more probes[7]	
$M_{Loaded} = 3.492\ M_{PL} + 1029.6\ \text{kg}$ [6]	
Source	Zandbergen (8 data points)
Payload mass range	70-700 kg
R^2	0.2766
RSE	50.0%
Flyby spacecraft[8]	
$M_{Loaded} = 4.3458\ M_{PL} + 270.4\ kg$ [7]	
Source	Zandbergen (7 data points)
Payload mass range	10-105 kg
R^2	0.5333
RSE	38.5%

Some relations have a low value of R-squared, which indicates that loaded mass is also affected to a large extent by other parameters. We mention in particular that the loaded mass strongly varies with the Δv needed to reach the target planet and/or whether all propellant is carried on board or that part of the propellant is contained in a kick stage. This is left for the reader to explore for him/herself.

Table 2: Dry mass estimation

Earth orbiting spacecraft	
$M_{Dry} = a \cdot M_{PL}$ kg [8]	
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 [9]	
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 [10]	
Source	Zandbergen (19 data points)
Payload mass range	50-950 kg
R^2	0.9436
RSE	14.6%
Communications satellites	
$M_{Dry} = 3.6 \cdot M_{PL}$ kg [11]	
Source	Brown (7 data points)
Payload mass range	100-620 kg
$M_{Dry} = 1.8225 \cdot M_{PL} + 545.1$ kg [12]	
Source	Zandbergen (12 data points)
Payload mass range	100-750 kg
RSE	19.4%

[7] Vehicles include Mars Express, Cassini Huygens, Chandrayaan-1, Exomars Trace Gas Orbiter (TGO), Rosetta, Pioneer 13, Viking Mars Orbiter, and Galileo.

[8] Vehicles include Voyager, Pioneer 10, New Horizons, Mariner 2 & 10, Giotto (excluding kick stage propellant, and Deep Space 1.

Table 2: Continued

Planetary orbiters and flyby spacecraft	
$M_{Dry} = 2.233 \cdot M_{PL} + 396.6$ kg * [13]	
Source	Zandbergen (31 data points)
Payload mass range	8-365 kg
R^2	0.603
RSE	38.2%
$M_{Dry} = 5.92 \cdot M_{PL} + 179.9$ kg + $M_{lander/probe/impactor}$ **[14]	
Source	Zandbergen (7 data points)
Payload mass range	8-365 kg
Lander/probe/impactor mass range	28-585 kg
R^2	0.902
RSE	31.3%
$M_{Dry} = 7.5 \cdot M_{PL}$ kg [15]	
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 [16]	
Source	Zandbergen
SSD	55% of estimated value

Notice the improvement in RSE obtained when estimating dry mass for the planetary spacecraft. This is mainly associated with the large variation in propellant mass, which is now excluded from the estimation.

* Vehicles included are planetary and Moon orbiters as well as fly-by spacecraft. No distinction is made between vehicles carrying RTGs or solar arrays as a means to generate electrical power. Some vehicles are carrying landers, lander probes and/or penetrators/impactors. The latter are included in the payload mass.

** This relation compares to relation [13], but lander and/or impactor/penetrator mass is/are considered separately.

Vehicle power[9]

Table 4: Total power estimation (photovoltaic systems only)

All missions	
$P_t = 1.13 \cdot P_{PL} + 122$ W [17]	
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 [18]	
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 [19]	
Source	SMAD
Mini spacecraft (< 100 W total)	
Operating power is 2 to 3 times the payload power	
Source	SMAD
Communications satellites	
$P_t = 1.1148 \cdot P_{PL} + 384.2$ W [20]	
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 [21]	
Source	Brown (10 data points)
Payload power range	100-1500 W
Meteorology satellites	
$P_t = 1.96 \cdot P_{PL}$ [22]	
Source	Brown (8 data points)
Payload power range	100-450 W
Planetary vehicles/deep space probes	
$P_t = 4.178 \cdot P_{PL} + 281.3$ W [23]	
Source	Zandbergen (17 data points)
Payload power range	36 – 193 W
R^2	0.5569*

Table 5: Total power estimation (RTG-based systems)

RTG powered planetary vehicles/deep space probes	
$P_t = 209.23 \cdot \ln\left(P_{PL}\right) - 491.8$ W [24]	
Source	Zandbergen (6 data points)
Payload power range	18-178 W
R^2	0.9188
$P_t = 332.93 \cdot \ln\left(P_{PL}\right) - 1046.6$ W [25]	
Source	Brown (3 data points)
Payload power range	75-250 W

* The relatively low value of R-squared is considered to be because in literature the distinction between total power generated in Earth orbit and when in orbit about the target is not always clearly made.

[9] Vehicles included are similar to the ones referred to in the previous section.

Vehicle size

One approach to vehicle size estimation is by collecting size information data of similar vehicles. Next figure shows mass density values found for a range of small satellites (loaded mass ≤ 500 kg). Values were obtained by dividing the S/C loaded mass by the volume of the smallest envelope completely encompassing the spacecraft in the un-deployed state. Values have been plotted and a regression relationship has been determined. The result shows that smaller satellites tend to be of higher mass density. This is explained because of lack of space and hence, the S/C are completely filled up, whereas for larger spacecraft the packaging allows for empty spaces.

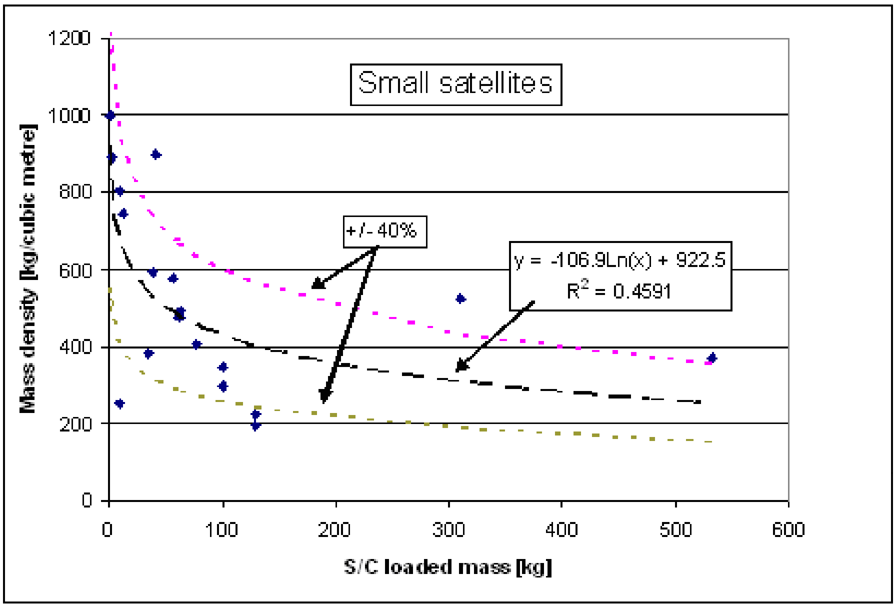


Figure 1: Spacecraft mass density versus spacecraft loaded mass for small satellites [Zandbergen]

The above relationship with some similar volume estimation relationships, but for different spacecraft, has been included in below table. Note that small satellites have a much higher density than the mass density that follows from relation [26] (100 kg/m^3).

Table 6: Size estimation

All Spacecraft	
$V_{S/C} = 0.01 \cdot M_{Loaded}$ m^3 [26]	
Source	SMAD
Mass range	135-3625 kg
Slope range	0.005-0.05
Density range	20-179 kg/m^3
Average density	79 kg/m^3
Small satellites	
$V_{S/C} = \frac{M_{Loaded}}{\rho_{S/C}}$ m^3 [27]	
$\rho_{S/C} = -106.9 \cdot \ln(M_{Loaded}) + 922.5$ kg/m^3 [28]	
Source	Zandbergen
R^2	0.4591
Mass range	1-500 kg (see figure)
RSE	40%

Table 6: continued

Micro satellites	
$V_{S/C} = 0.0019 \cdot M_{Loaded}$ m^3 [29]	
Source	TU-Delft, Aas
Slope range	0.0006-0.005
Density range	194-1584 kg/m^3
Mass range	1-50 kg
Solar powered deep space probes[10]	
$V_{S/C} = 0.0045 \cdot M_{Loaded}$ m^3 [30]	
Source	Zandbergen (19 data points)
Slope range	0.0024-0.0185
Density range	54-409 kg/m^3
Average density	222 kg/m^3
Mass range	286-3625 kg
RTG powered deep space probes[11]	
$V_{S/C} = 0.0269 \cdot M_{Loaded}$ m^3 [31]	
Source	Zandbergen (6 data points)
Slope range	0.0149-0.0555
Density range	18-67 kg/m^3
Average density	37.2 kg/m^3
Mass range	258-5623 kg

[10] Vehicles used to determine the relationship shown include Mangalyaan, NEAR, SMART-1, Clementine, Mars Reconnaissance Orbiter, Mars Global Surveyor orbiter, Juno, Venus Express, Messenger, Pioneer 13, MAVEN, Mars Global Surveyor-98 orbiter (Mars Climate Orbiter), etc.
[11] Vehicles used to determine the relationship shown include Voyager, Ulysses, Cassini, and Galileo.

Vehicle cost

Figure 2, taken from a 1996 NASA study, shows specific spacecraft cost (i.e. spacecraft cost per unit of spacecraft dry mass) of a number of NASA science and planetary 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 1) specific cost in the range 150.000 – 450.000 $/kg and 2) the larger specific cost 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. This is the range of vehicles build under the faster, better, cheaper regime as initiated by NASA administrator Goldwin. 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.

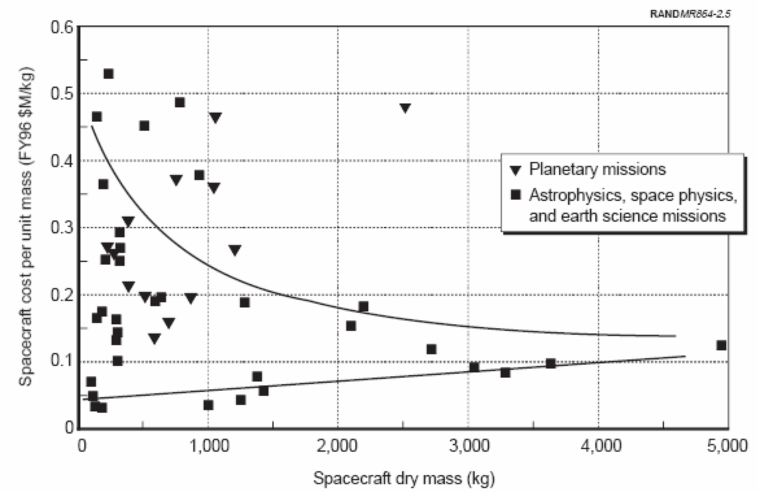


Figure 2: Effect of size on specific cost of NASA science and planetary spacecraft [Sarsfield]

An explanation for the increasing specific cost with spacecraft mass is that the larger the spacecraft become, they tend to become more complex and not just heavier (larger).

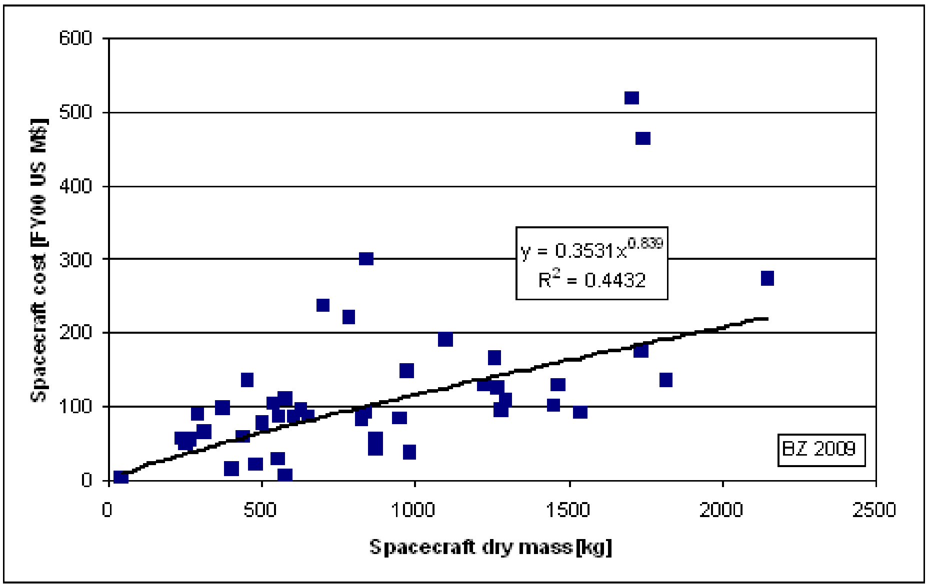


Figure 3: Spacecraft cost versus spacecraft dry mass [Zandbergen]

Cost data in foregoing figure have been taken from appendix B, and applies mainly to Earth orbiting (communications, Earth observation, weather, navigation, military) spacecraft. It shows again a considerable spread in data at identical dry mass. This is in part attributed to that different spacecraft of same mass may be different in complexity (single payload versus multiple payloads), but also because some spacecraft are built in multiple. When build in multiple, the cost per spacecraft will reduce as building it goes faster because of the experience obtained with the earlier build.

The relation shown in foregoing figure is summarized in table 7.

Table 7: Cost estimation

Spacecraft (general)	
1. $C_{S/C} = 0.3531 \cdot \left(M_{SC}\right)_{Dry}^{0.839}$ FY2000 M$ [32]	
Source	See below figure
Mass range	40-2350 kg

To get more accurate cost data, it is better to distinguish different categories as is done in next table. Note that cost relations in this table are providing <u>specific cost</u> and not just cost. To obtain vehicle level cost, you need to multiply the specific cost with the vehicle dry mass.

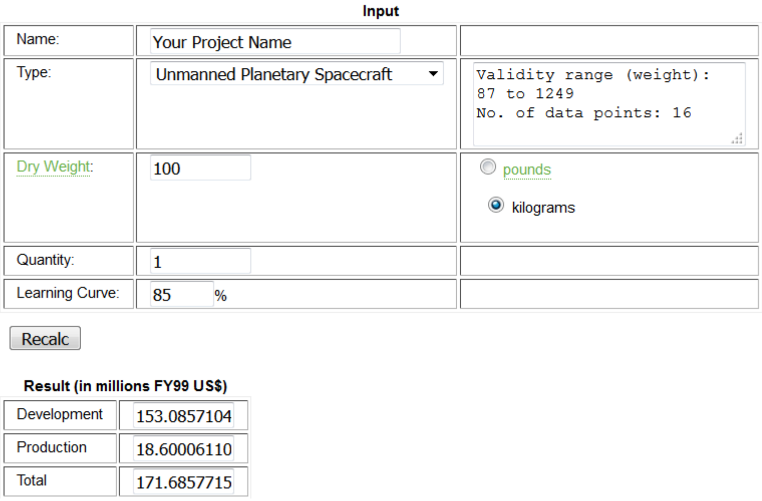
Table 8: Cost estimation based on specific cost (cost per kg) for some spacecraft categories

GEO Communications satellites (Commsats)	
$C_{S/C} = -0.0673 \cdot \left(M_{SC}\right)_{Dry} + 190.1$ FY2000 k$/kg [33]	
Source	Zandbergen
Mass range	400-1800 kg
R^2	0.5912
RSE	23%
Navigation satellites (Navsats)	
$C_{S/C} = 40 - 45$ FY2000 k$/kg [34]	
Source	Zandbergen
Mass range	400-1000 kg
Earth Observation satellites (EOSats; optical imaging)	
$C_{S/C} = 0.0264 \cdot \left(M_{SC}\right)_{Dry} + 192.95$ FY2000 k$/kg [35]	
Source	Zandbergen
Mass range	250-2000 kg
R^2	0.276
RSE	13.8%

See next page for a simple online cost model as available from NASA. This simple cost model can also be used to get an impression of the effect of building multiple of a kind instead of a single one.

NASA spacecraft/vehicle level cost model

A simple online cost model (SVLCM) that provides a useful method for quick turnaround, rough-order-of-magnitude estimating. The model can be used for estimating the total cost, development and production cost of a range of spacecraft. It also allows for cost estimation of launch vehicle stages, rocket engines and scientific instruments.



Costs are presented in FY 99 US$. To convert these cost into year 2015 US$, multiply the result with a factor 1.50. Notice that cost estimations are based on NASA developed vehicles and hence do not necessarily relate to other spacecraft. In 2019, the model could be accessed via the following URL: http://www.globalsecurity.org/military/intro/reference/calc/SVLCM.htm

Notes:

1. In input sheet a typical learning curve factor is given. This is a factor that is important when producing more than 1 spacecraft and it factors in that if we build more than 1 item (vehicle/spacecraft), the average time (and hence cost) to build the item reduces. A cost factor of 85% essentially indicates a cost reduction with a factor 0.85 with every doubling of the quantity of vehicles produced.
2. Ty estimating the cost of the Indian Mangelyaan spacecraft with a dry mass of about 483 kg. It gives a vehicle cost of about US$ 416 million (FY1999). Actual mission cost (spacecraft, launch, etc) was less than US$ 100 million, which shows that US cost figures are not necessarily identical to Indian (and/or European) cost figures.

Vehicle reliability

Table 9: Reliability estimation

General	
$R = (e)^{-(\lambda \cdot t)}$ [36]	
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 10: Reliability estimation details

General	
$R_{SC} = R_{payload} \times R_{bus}$ [37]	
Here $R_{payload}$ and R_{bus} can be estimated using the relation [36] 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

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 11. 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 11: 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 [8] and the second by Formula [9], see Table 2. Formula [8] is taken from [Brown (46 data points)], and Formula [9] 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 11.

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, being [22], [17] and [18], that could be used. The results are given in Table 11. Comparing the values found, we find that the three values are reasonably comparable, thereby lending credibility to our estimates. Question remains whether we should take peak or average power? For now the answer is that it depends on whether you design the S/C with a secondary power source or not. If not, it is better to use the value for peak power.

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	STRUC	Thermal	EPS	TT&C	ADCS	PROP	Harness	C&DH
Orsted	Science	56.3	22.9	28.8	0.9	16.9	10.3	7.3	NA	5.5	7.5
Freya	Science	216.9	33.7	22.5	2.4	18.7	2.6	5.9	8.2	NA	6.0
SAMPEX	Science	160	32.5	23.1	2.5	20.0	3.1	6.3	NA	5.0	7.5
ANS	Science	129.3	33.2	29.4	0.9	8.7	5.4	13.5	NA	NA	6.0
Viking	Science	289.6	16.8	16.1	2.7	6.9	3.1	3.5	5.8	NA	NA
Bird	Science	77.74	30.0	25.9	8.2	14.7	2.4	9.3	NA	NA	0.9
NATO III	Comm.	316.6	22.1	19.3	6.5	34.7	7.5	6.3	2.4	NA	NA
Gurwin II	Techn. test	47.0	14.0	36.0	NA	12.3	4.9	7.3	NA	3.4	22.1
Temisat	Comm.	41.9	25.8	19.8	NA	36.8	NA	1.7	NA	9.1	6.9
ORBCOMM	Comm.	47.5	19.4	15.3	1.9	22.5	NA	6.8	4.0	NA	6.8
PoSAT-1	Comm. / test	50.2	12.2	13.7	NA	32.9	11.2	20.9	NA	3.0	6.2
Hausat-1	Techn. test	1	16.0	27.0	3.0	21.0	14.0	6.0	NA	NA	4.5
Delfi C3	Techn. test	2.9	16.7	26.5	2.3	15.4	12.1	5.2	NA	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
NA: Not Applicable. This may be because the system is not available or because data has been included in some other subsystem

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 C^3 all lack a propulsion system on board.

All information provided on spacecraft mass distribution is for Earth orbiting spacecraft. These are not necessarily applicable to interplanetary spacecraft / deep space probes. Next table hence provides a mass distribution for some interplanetary spacecraft. Data is limited to some early vehicles as data of more recent spacecraft is hard to find.

Table 413: Mass distribution data of some early interplanetary spacecraft

Program	Total Weight of one S/C in lbs	% Weight of Telecommun. & Data	% Weight of Structure	% Weight of Propulsion	% Weight of Guidance and Control	% Weight of Power	% Weight of Experiments
Ranger 6-9	788	11	34	2	8	32	13
Ranger 1-5	612	11	32	2	8	27	20
Surveyor 1-7	787	7	37	29	11	13	3
Mariner R	437	19	30	4	12	24	11
Mariner 64	555	20	25	4	15	28	8

From: SPACECRAFT COST ESTIMATION. by W.P. Finnegan, C.A. Stone, Astro Sciences Center of IIT Research Institute Chicago, Illinois for Lunar and Planetary Programs, Office of Space Science and Applications, NASA Headquarters Washington, D. C.

Spacecraft subsystem power usage data

Table 1 provides data on power usage of the various S/C subsystems for specific Earth orbiting spacecraft. Note that other types of spacecraft may have a different distribution. To find out whether this is the case (or not), you essentially have to search for mass data of these other types of 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. Once data has been collected and processed, these data can be used to determine relationships that allow for budgeting, see a later section.

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 2: 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*. Interesting to note here is that kick stages are seen as a subsystem of the spacecraft.

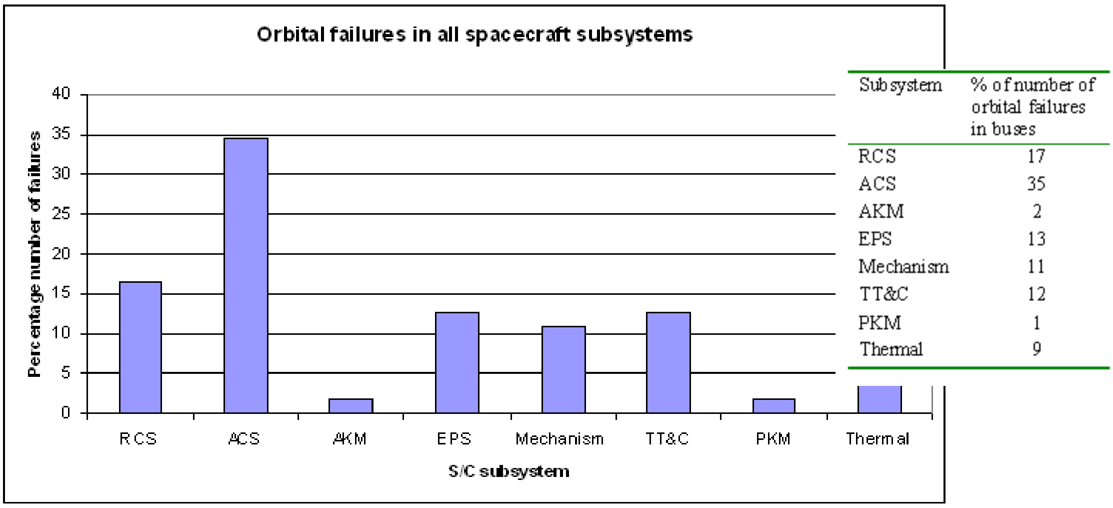


Figure 1: Percentage number of failures distributed over the various spacecraft subsystems

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*.

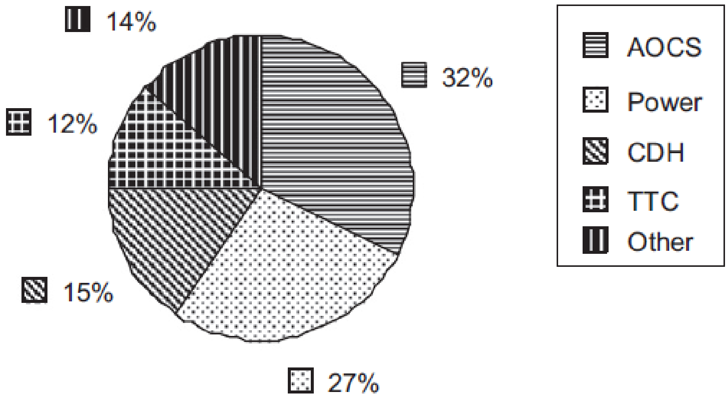


Figure 2: Spacecraft subsystem failures as a percentage of total number of failures [Tafazoli]

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 10*.

Table 3: 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 4*, 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 4: 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. It is mentioned that data should be applied with some caution as only average values are given. Depending on the (type of) spacecraft, the developing organization, and engineering experience, values do differ. Key should be that the differences are within certain limits so that a proper and competitive design can be made. Note that all data apply to S/C that provide the functions indicated and should for instance not be applied to the estimation of propulsive stages or only with extreme caution wherein validation of the used relationships is extremely important.

Spacecraft subsystem mass estimation relationships

Subsystem mass of S/C can be estimated using:

$$M_i = \left(\frac{\%}{100}\right) \cdot (M_{bus})_{dry} \qquad [2]$$

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 5* for medium to large spacecraft buses and *Table 6* for small spacecraft and planetary spacecraft buses. Note that M_{dry} is bus dry mass (so payload is not included)

Table 5: Medium to large spacecraft bus mass distribution as % of S/C bus dry mass [SMAD]

Subsystem	Subsystem mass	Range
Propulsion	4.75 %	2.5-7 %
ADCS	6 %	3-9 %
Communications	4.75 %	2.5-7 %
Thermal	8.5 %	2-15 %
Power	30 %	20-40 %
Structures	20 %	15-25 %

Table 6: Small and planetary S/C bus mass distribution as % of S/C bus dry mass [Zandbergen]

Subsystem	Subsystem mass distribution	
	Small S/C (S/C mass ≤ 500 kg)	Planetary S/C[12]
Propulsion	6.1 %	19.0% (11.6-26.4%)
ADCS	9.6 %	8.0% (5.2-10.8%)
Communications	9.2 %	7.7% (2.3-12.1%)
Thermal	3.8 %	4.8% (2,6%-7%)
Power	24.5 %	22.4% (12.8%-32%)
Structures	29.3 %	23.0% (18.8-27.2%)
Harness	7.2 %	9.8% (6.9-12.7%)
Command & Data Handling	10.4 %	5.3% (2.4-8.2%)

In the *Table 5* no data are included for the Command & Data Handling system. This does not mean that this system is not present, but that the data is included in some other subsystem.

The data in *Table 6* 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.

[12] Between brackets the 1 SSD mass % range is given. Data has been based on 8 planetary S/C for which a detailed mass distribution was found in literature.

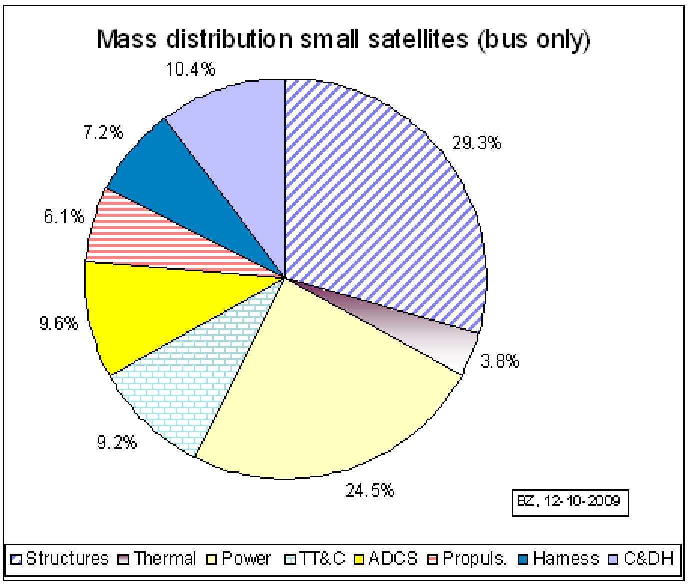


Figure 3: Small spacecraft mass distribution [Zandbergen]

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 Relative Standard Error has been determined. It follows RSE is 21.7%.

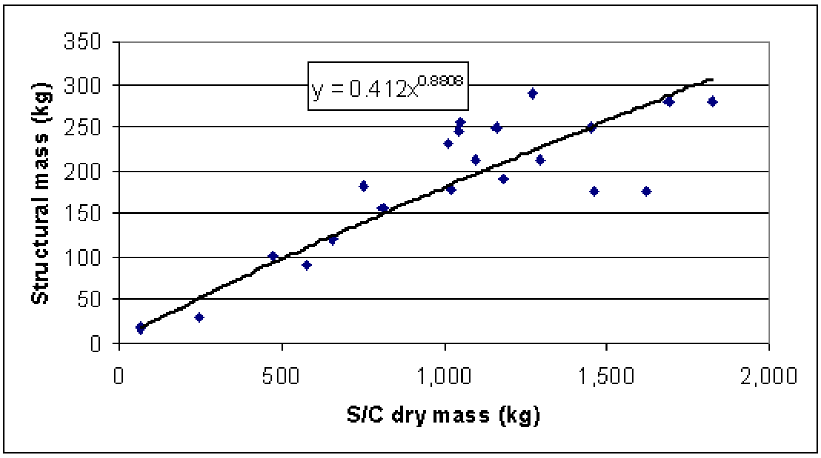


Figure 4: Structures mass for medium to large spacecraft [Zandbergen]

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 7*.and P_{total} is total spacecraft power (considered here as a known).

*Table 7: 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	5-10
Communications	15 W	5	0-10
C&DH	5 W	5	0-10
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 7* provides specific values of power usage for the various subsystems. As using these values for micro-spacecraft holds quite some uncertainty, it is advised to develop an improved estimation method.

From the work of [Brown] the following table, providing power allocation guidelines for different spacecraft can be obtained. Note that in table no S/C are included that are electrically propelled. In that case, it is expected that propulsion may take a lot more power.

Table 8: 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:

$$C_i = \left(\frac{\%}{100}\right) \cdot C_{SC} \qquad [4]$$

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 9* and C_{sc} is total S/C bus cost (considered here as a known).

Table 9: 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 9* have been used to generate a pie chart.

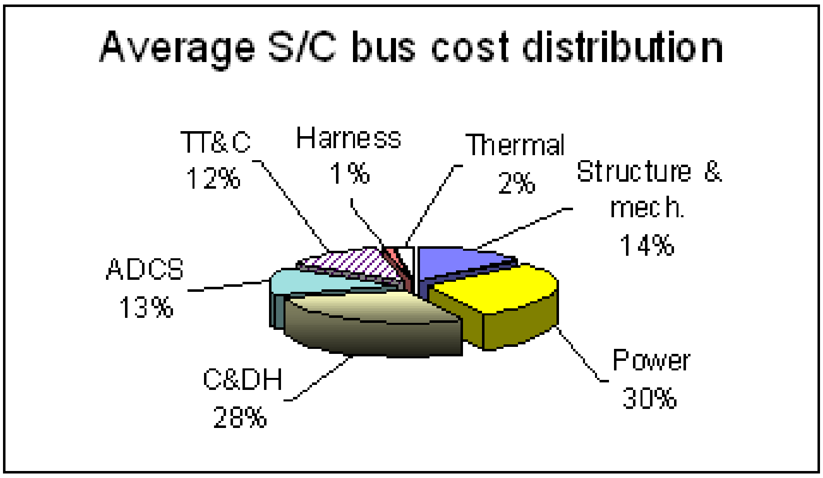


Figure 5: Small spacecraft cost distribution (4 S/C) [Zandbergen]

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 10*.

Table 10: 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 of which 60% are considered bus failures and the remainder payload failures [Sultan].

Table 11: 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

Above data are taken from an industry wide survey. Numbers may vary depending on the country of origin of the spacecraft, the industries involved and the specific design of the spacecraft. This though is left for the reader to explore for him/herself.

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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

$$\text{Mean} = \mu_x = \sum_{i=1}^{i=n} \frac{x_i}{n}$$

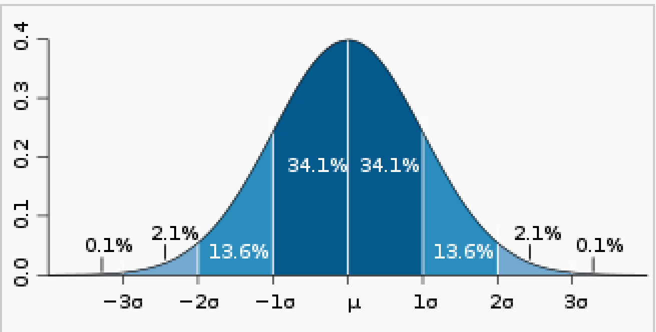
- median, which is the mid-point of the series; half the data in the series is higher than the median and half is lower
- variance, which is a measure of the spread in the distribution around the mean, and is calculated by first summing up the squared deviations from the mean, and then dividing by either 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.

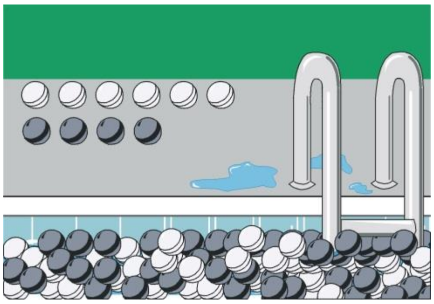
Taking a Small Sample of Marbles

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.

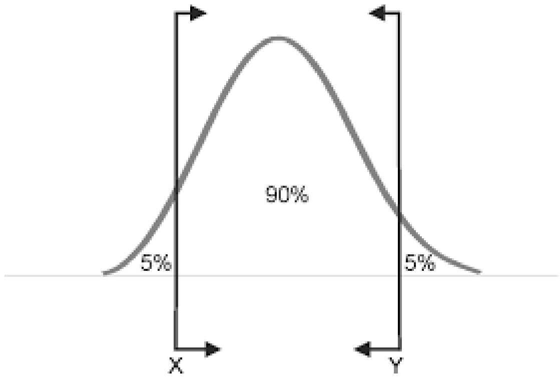


Confidence interval

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.

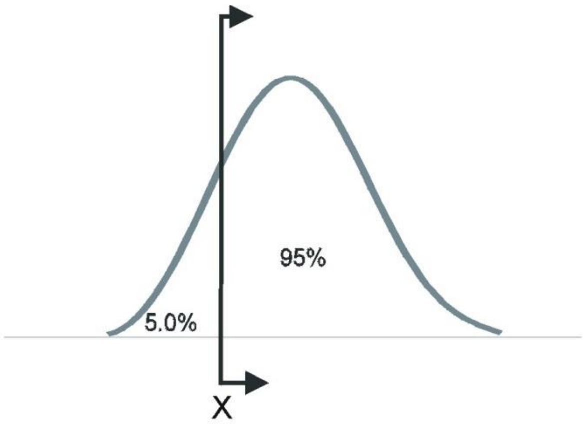
Two-Sided Bounds

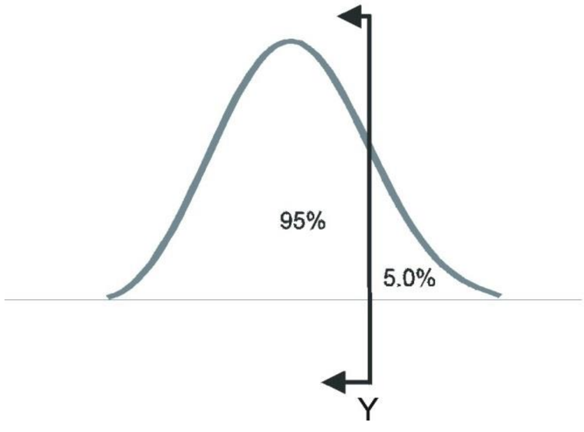
When we use two-sided confidence bounds (or intervals), we are looking at a closed interval where a certain percentage of the population is likely to lie. That is, we determine the values, or bounds, between which lies a specified percentage of the population. For example, when dealing with 90% two-sided 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.

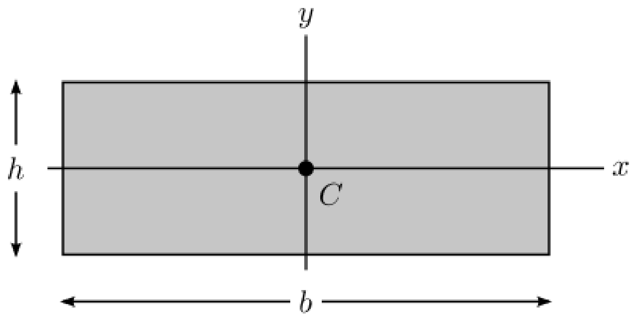
Example: For a S/C you have estimated a payload cost of 40 M€ with a standard error (or standard sample deviation) of ± 4 M€ and a bus cost of 30M€ ± 6 M€. Total cost of the S/C is 30 M€ + 40 M€ = 70 M€ with an uncertainty of:

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 square cross section
See above figure, but now with walls of thickness t and ribs of length b:

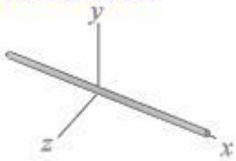
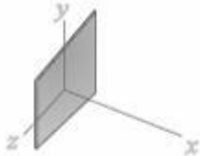
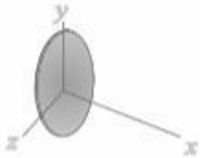
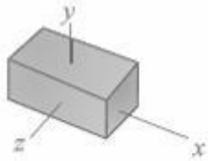
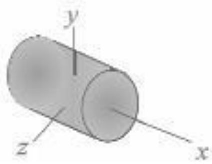
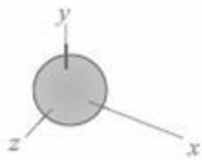
$$I_x = I_y = \frac{t\,b^3}{3}$$

Thin walled cylinder:

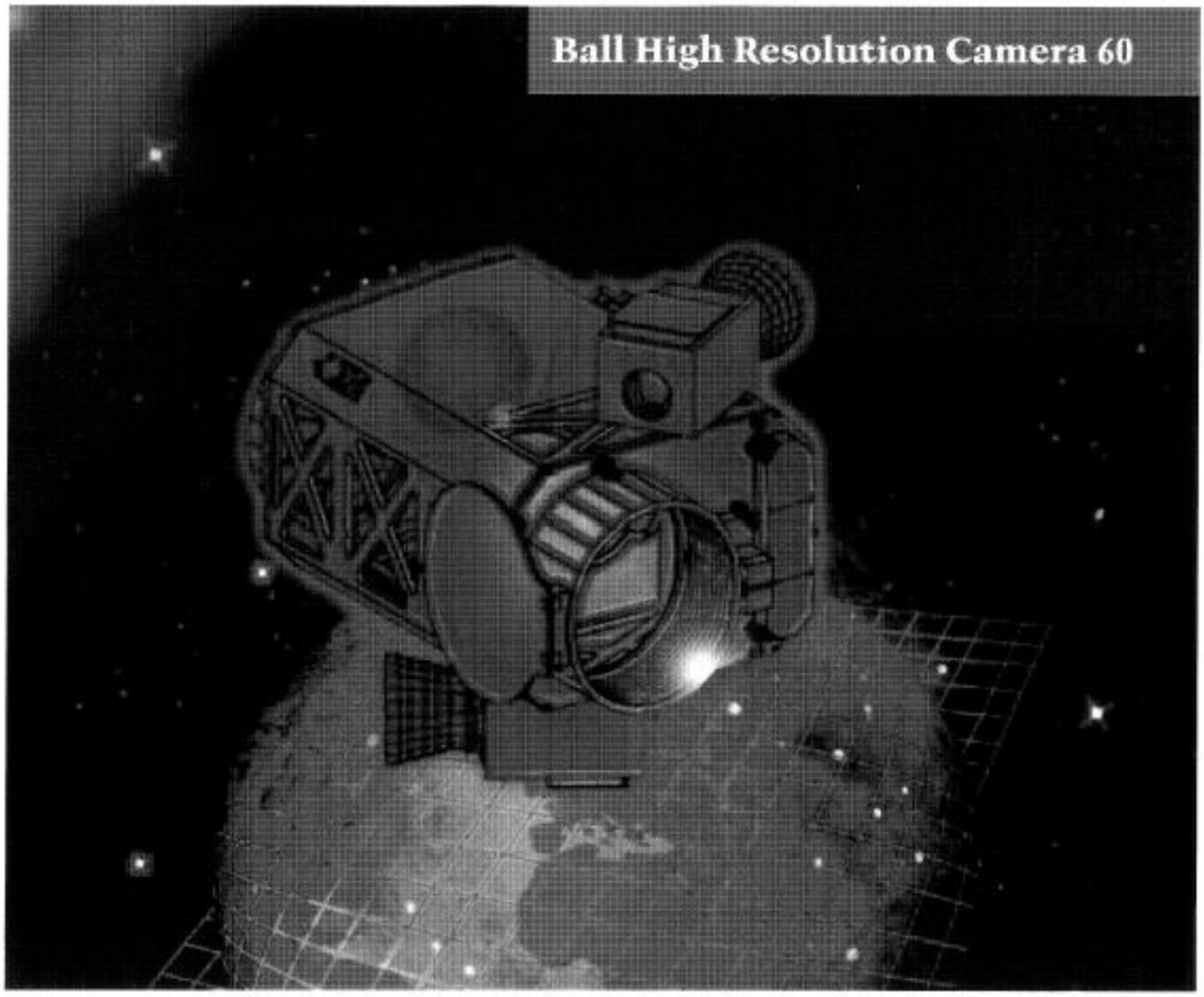
$$I_o = \pi \cdot r^3 \cdot t$$

- 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.

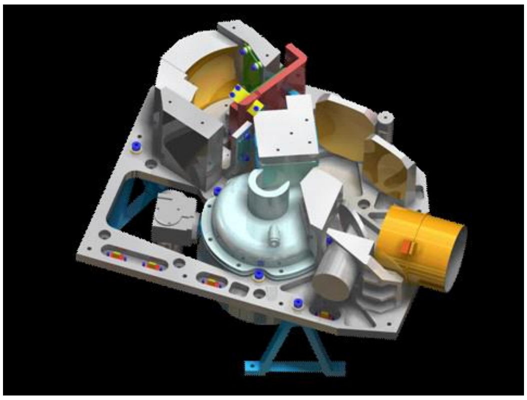


Table 1: Specification of the HIBRIS instrument (taken from Cosine Brochure)

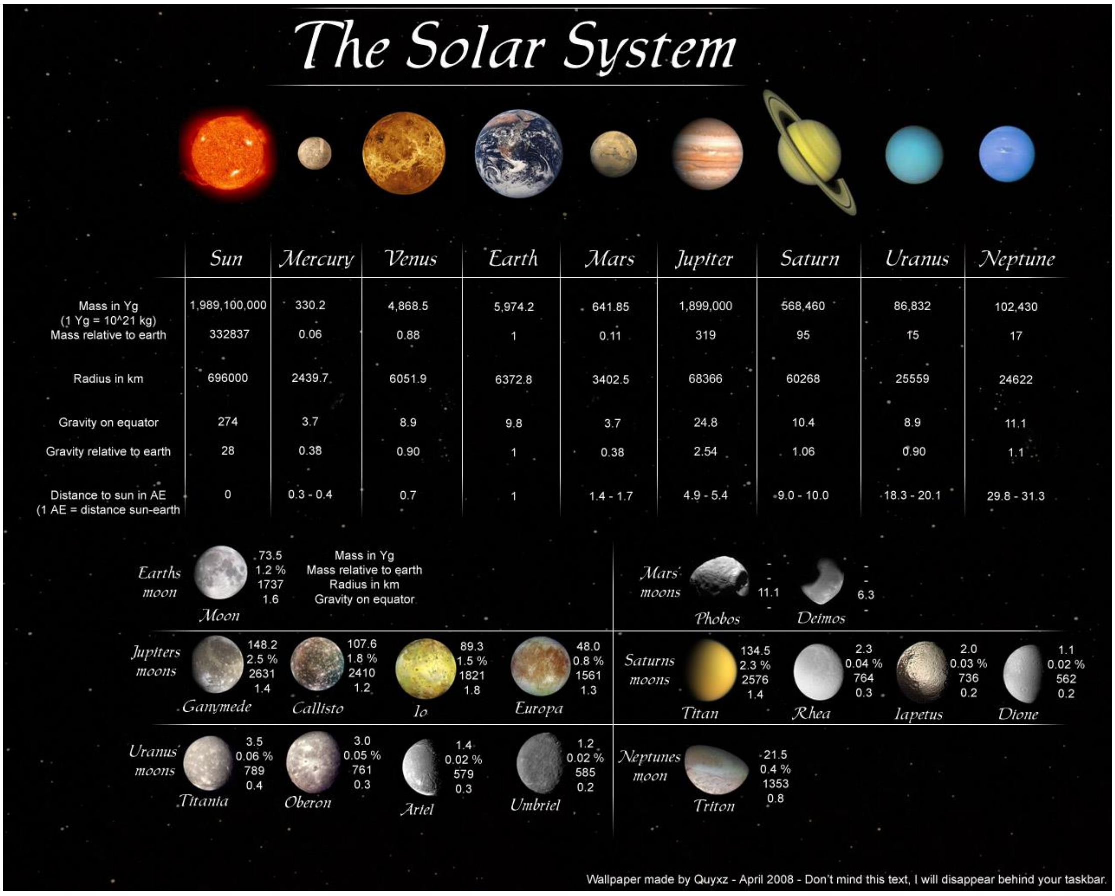
HIBRIS	
Total Mass	7.1 kg
Size (LxWxH)	22x26x21 cm
Avg. Power consumption.	4.11 W
Near Infrared Hyperspectral Imager (NSI)	
Linear Variable Filter Ranges	700 to 1400 nm
	1400 to 2800 nm
	2800 to 5200 nm
Spectral Resolution	1% of Central λ
Spatial Resolution (at 600 km AGL)	50 m
FOV	14.5°
Thermal Imager (TI)	
Spectral Range	10 to 73 μ m
Spectral Resolution	7 μm
Spatial Resolution (at 600 km AGL)	715 m
FOV	6.9°

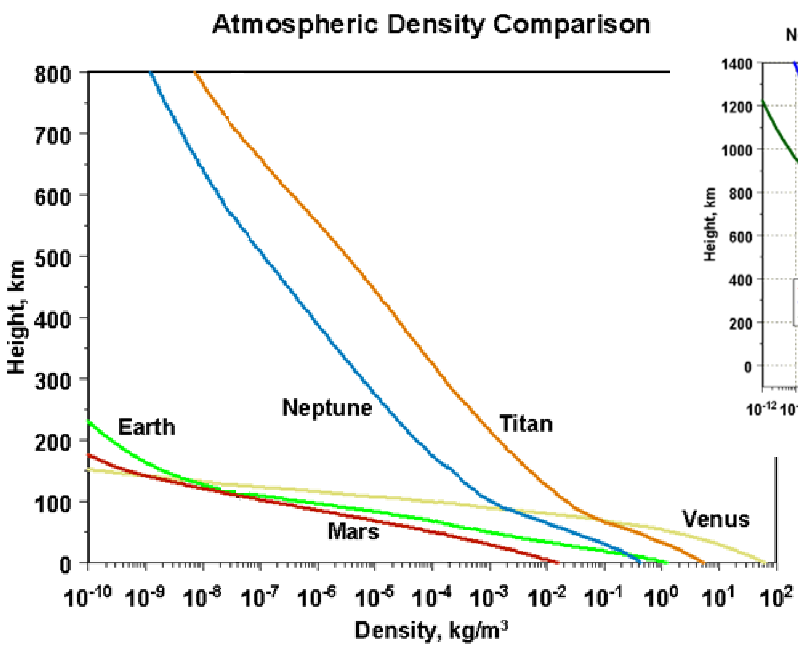
Appendix H: Earth Satellite Parameters

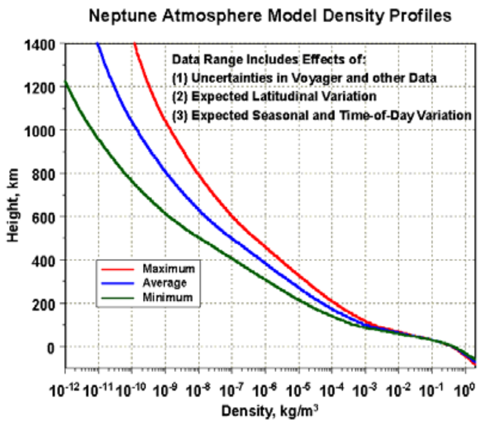
GENERAL PARAMETERS					
Alt (km)	Mean DENSITY (kg/m^3)	Circular Velocity (km/s)	Range to Horizon (km)	Period (min)	Max Eclipse (min)
0	1.2	7.905	0	84.49	42.24
100	4.79×10^{-7}	7.844	1,134	86.48	38.40
150	1.81×10^{-9}	7.814	1,391	87.49	37.76
200	2.53×10^{-10}	7.784	1,610	88.49	37.28
250	6.24×10^{-11}	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.

Appendix I: Solar System Parameters and atmospheric density







Atmospheric density comparison taken from: ATMOSPHERIC MODELS FOR AERO-CAPTURE, by C. G. Justus, Aleta L. Duvall, *and* Vernon W. Keller, *NASA.*

Small insert shows that at high altitude mass density can vary substantially. Also note that values are limited to about 10^{-10} kg/m^3. However, at high altitudes there might still be some atmosphere left that can cause drag.