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Spacecraft Operations

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Foreword

Timing is everything—this is especially true for spaceflight operations. 2014 is a special year for the European space community, the year that started with the wake-up of ROSETTA, ESA's comet chaser, with Philae, the German comet lander, which is on its extraterrestrial voyage since 2004. It has been awoken from its hibernation and is providing us with data during its carefully planned first approach on a comet—67P/Churyumov–Gerasimenko.

In November 2014, the mission culminates in the descent of the lander PHILAE to the surface of the comet's frozen nucleus—the resulting measurements may help us answer some of the fundamental questions about the evolution of life on earth. Comets are considered as veterans of our solar system—their analysis provides insights in the early days of our galactic home.

Another fact which makes 2014 quite special for the European spaceflight community is the mission of two Europeans to the International Space Station—never before have we seen two long duration ESA missions within 1 year. Alexander Gerst has the chance to beat Thomas Reiter's record of logging the longest time in one space mission for a German. Samantha Cristoforetti is only the second ESA female astronaut—and the first Italian woman in space. During their stay in orbit both will collect data for many months of scientific research and definitely awake the public interest in spaceflight in their home countries.

In this fascinating year falls the publication of the book "Spaceflight Operations." It discusses important principles and aspects of the operation of space vehicles. Designated experts of the DLR's German Space Operations Center (GSOC), ESA's European Space Operations Center (ESOC), and the University of Southampton have put together a handbook for operations, which provides not only a good overview, but also the expert background information, to make the book not only a theoretical description, but a vivid testimonial of many years of experience. Both the GSOC and ESOC spaceflight operations centers were founded in 1967 and GSOC's manned spaceflight history dates back to 1985 with the German spacelab mission D-1.

The authors of this book are involved in many of the most exciting space missions and projects currently ongoing: Columbus and the International Space

Station, ROSETTA and the lander PHILAE, TerraSAR-X and Tandem-X, the European Data Relay System EDRS, and the space robotic mission DEOS. They have worked during the preparation and execution phases, acting in their roles as managers, engineers, planners, subsystem specialists, and flight controllers. It is inspiring to read their articles and to listen to their "lessons learnt."

It is my desire that the book will provide both an interest and stimulus for future missions—and may help to improve subsequent operations concepts.

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Jan Wörner
Chairman of the Executive Board of the German Aerospace Center (DLR)

Preface

This book has evolved from the "Spacecraft Operations Course," a 1-week series of lectures and exercises, which has been held annually for the last 14 years at the German Space Operations Center (GSOC) in Oberpfaffenhofen. Originally, our plan was simply to create a handout for this course. However, as we found that there is currently no up-to-date book that deals exclusively with the operations of spacecraft, we extended our project, supplemented, and detailed the chapters, allowing us to complete it in form of a book. That said, most of the 22 subsections are still based on lectures from our current "Spacecraft Operations Course." In addition to the participants of the course, the target group of this book includes students of technical or scientific studies as well as technically interested parties, who wish to gain a deeper understanding of spacecraft operations.

The book begins with a brief summary of the space segment (Chap. 1), introducing the "Space Environment," "Space System Engineering," and "Space Communications," which establishes the connection to the ground segment.

The book now follows the classical fields of operations: mission operations, ground infrastructure, flight dynamics, and mission planning.

The mission operations system is described in Chap. 2. This chapter is based on the life cycle of a mission and is therefore chronologically represented along the phase model used in astronautics. The ground and communications infrastructure, however, provides cross-mission support services; hence, the representation of Chap. 3 is oriented towards the systems. The flight dynamics system (Chap. 4) in turn has the focus on the attitude and orbit control of the satellite platform while the mission planning system (Chap. 5) takes care of an effective management and utilization of the payload.

The two last chapters deal with the details of specific mission types: Chap. 6 describes the operation tasks of the various subsystems of a classic unmanned satellite in Earth orbits. Chapter 7 describes the special requirements of other mission types which are caused by the presence of astronauts, due to a satellite approaching at another target satellite, or by leaving the Earth orbits in interplanetary missions and landing on other planets and moons.

The process of writing of this book had some analogies to the preparation of a space mission: while initially there was a systematic planning process, the implementation was more of an evolutionary process including various mutations and selections. Planned sections were modified, merged, rearranged, or occasionally disappeared, new sections were introduced. Every now and then the book project came in competition with the preparation and implementation of space flight missions, so that chapters had to be put on hold. On the other hand, launch delays in space missions are a regular occurrence and this is something you learn to cope with. However, after 2 years of preparation, our book project is finally on the way to the launch pad.

This is the moment at which we like to thank all our supporters. Firstly, all authors who have endured to the end and somehow found enough spare time in addition to their ongoing mission projects to provide valuable input. A very warm thank you to our layout team Petra Kuß, Adriane Exter, and Juliane von Geisau, who provided significant support, particularly at the end of the project. We also would like to thank Bernd Dachwald of the FH Aachen, who supported us during the early planning process, Sergei Bobrovskyi and Frank Roshani, who gave valuable support when editing formulas, as well as Simon Maslin, who supported us linguistically as a "native speaker." Last but not least, our thanks go to Beate Siek from Springer publishing for her patience during the development of this book.

It is now time to inject our book into orbit—we hope you enjoy reading it.

Wessling, Germany — Florian Sellmaier
Wessling, Germany — Tom Uhlig
Wessling, Germany — Michael Schmidhuber
March 2014

List of Editors and Contributors

Ralph Ballweg is Project Manager and Flight Director in the geostationary group of the Flight Operations Departement of the DLR/GSOC. He is responsible for the operations of two communication satellites and two instruments for Laser-based communication. He has a degree from the Technical University of Munich in Aerospace Engineering and 25 years of experience in spacecraft operations in Germany and the USA.

Jérôme Campan is the head of the European Planning and Increment Coordination team at the Columbus Control Center in Oberpfaffenhofen. After receiving his engineering diploma from the ICAM school (Institut Catholique des Arts et Métiers) in Toulouse, he graduated from the ISAE school (Institut supérieure de l'aéronautique et de l'espace) in the "Space System engineering" field. He started his professional career as a Columbus Operations Coordinator for the first 3 years before moving to his actual position.

Franck Chatel is Flight Director for the HAG1 and EDRS missions within the department mission operations at the German Space Operations Center (GSOC). He graduated in aerospace engineering at the Institut Supérieur de l'Aéronautique et de l'Espace (ISAE) in Toulouse (France) and joined DLR in 2001. He was involved as propulsion and AOCS subsystem engineer and later as flight director, in different LEO (BIRD, SAR-Lupe, GRACE) and GEO (EUTELSAT W24) missions. He supported the early phase of the GALILEO project at spaceopal and DLR GfR before moving to his current position.

Sabrina Eberle is project manager for the EnMAP mission and Flight Director for the SATCOMBw Mission and the on-orbit servicing projects at GSOC. She graduated in Electrical Engineering from the Technical University of Munich (Germany) in 2004 and has joined the German Space Operations Center the same year. She was engaged as subsystem engineer and Flight Director in the SAR-Lupe 1-5 missions. Later she was involved in the TerraSAR-X, TanDEM-X, and the PRISMA missions.

Ralf Faller is project manager for on-orbit servicing projects in the mission operations department at the German Space Operations Center (GSOC). In 1990, he received his diploma in aerospace engineering at the Technical University of Berlin, Germany. He started in 1991 at the DLR Oberpfaffenhofen in the spaceflight dynamics department, being responsible for attitude-related software development and mission support and changed to the mission operations department in 1997 where he focused his work on spacecraft flight operations topics. He was involved in several LEO and GEO missions as subsystem engineer, flight operations manager, and finally project manager.

Paolo Ferri is Head of the Mission Operations Department at the European Space Operations Centre (ESOC) of the European Space Agency in Darmstadt, Germany. After his Doctor's degree in theoretical physics (University of Pavia, Italy), he started working in 1984 at ESOC as visiting scientist on the Exosat X-ray astronomy mission science operations. He moved to mission operations in 1986 and worked on different types of missions, including Eureca (microgravity, deployed and retrieved with a Space Shuttle), Cluster (space plasma physics), Rosetta (comet science), and GOCE (earth gravity). In 2006 he became responsible for all ESA interplanetary mission operations, including Mars Express, Venus Express, BepiColombo, ExoMars, and Solar Orbiter. Since 2013 he leads the Mission Operations Department, in charge of all ESA unmanned missions and operations preparation and execution.

Marcin Gnat received his M.Sc. degree in electrical engineering in 2000 from the Technical University of Koszalin, Poland. From 2001 to 2010 he worked as a Senior Test Engineer in the semiconductor industry at Infineon Technologies and Qimonda, followed by the consultant activities in the same area. Since 2010 he works as Ground Data System Manager for satellite missions at DLR's German Space Operations Center (GSOC).

Tobias Göttfert is member of the mission planning group of the Department for Mission Operations at the German Space Operations Center (GSOC). He received his physics diploma and his Ph.D. while working at the Max Planck Institute for Physics (Munich) on the subject of high-energy physics at the ATLAS detector at the LHC. He joined DLR in 2010 for mission planning tasks at the GSOC. His present work revolves around software engineering for several current and future mission planning systems and telemetry data analysis.

Jacobus Herman (1954) is leading the AOCS group at GSOC since 2001. He joined the DLR in 1989, at first in the flight dynamics department. Wrote a study on space debris commissioned by the ESA in 2000. Worked several years at ESTEC (Noordwijk, the Netherlands) and the Max Planck institute for extraterrestrial physics in Garching (Germany) after getting his Ph.D. in astrophysics in 1983 in Leiden (the Netherlands).

Dennis Herrmann studied aerospace engineering and received his diploma in 2007. Afterwards he joined DLR and worked as part of the Columbus team as

Columbus Operations Planner (COP) and Columbus Operations Coordinator (COL OC). He was assuming the group lead function for that position in 2009, before he was promoted to the Columbus Flight Director group. He acted as Lead Increment Flight Director in 2012 and received a project management certification, before he left the spaceflight area and changed into the automotive safety business, where he to date works for BMW.

Felix Huber is the director of the German Space Operations Center (GSOC). He graduated as an aerospace engineer in 1990 at Stuttgart University and received his Ph.D. on antimatter propulsion in 1994 after a guest stay at CERN/Geneva. He was head of a private research institution where he designed and built several space born experiments including the first commercial experiment on board the International Space Station (ISS) until his appointment of a professorship at the University of the German armed forces in Neubiberg on space operations. He has contributed to several books on aerospace engineering and communications.

Ralph Kahle received the Dipl.-Ing. degree in aerospace engineering from the Technical University of Dresden, Germany, in 2001, and the Dr.-Ing. degree from the Technical University of Berlin, Germany, in 2005. In 2004, he joined DLR/GSOC as a Flight Dynamics (FD) Engineer. He is responsible for the TerraSAR-X/TanDEM-X and DEOS FD system engineering and operation. His field of work also comprises space mission analysis, FD software development, and execution of Launch and Early Orbit Phase and routine FD operations.

Michael Kirschner received the Dipl.-Ing. degree in aerospace engineering from the University of Stuttgart, Germany, in 1982, and the Dr.-Ing. degree from the Technical University of Munich, Germany, in 1990. In 1990, he joined DLR/GSOC as a Flight Dynamics (FD) Engineer. He was and is involved in all satellite mission operations at GSOC. His field of work as senior engineer at the flight dynamics group comprises besides space mission analysis, FD software development, and execution of Launch and Early Orbit Phase and routine FD operations the teaching and training of new colleagues.

Christoph Lenzen is member of the mission planning group of the Department for Mission Operations at the German Space Operations Center (GSOC). His special subject is the automated timeline generation process, where he significantly contributed to the Mission Planning System of the TerraSAR-X/TanDEM-X missions. Current work includes integration of on-board planning within a ground-based planning system. He received his mathematics diploma from the University of Munich (LMU) in 2001. After one year as scientific assistant at the LMU, he started work at GSOC in 2002.

Jürgen Letschnik born in 1979 is a space operation engineer at LSE Space GmbH (LSE) in Weßling, Germany. He studied telecommunication engineering and did his Ph.D. in the field of tele-operations on robotic space missions at the Institute of Astronautics of Technical University Munich, Germany. Since March 2008 he is working on a geostationary telecommunication mission for the German military as a repeater payload subsystem engineer. In 2010 he got the position of the team leader of the whole flight control team for the announced military telecommunication mission above. His main research is in repeater payload communications engineering, in particular intersatellite link communication technologies and tele-operations for robotics space missions.

Sebastian Löw is Project Manager for GRACE-FO and Flight Director for the missions GRACE and PAZ. He graduated in Physics from the Ludwig-Maximilian-University of Munich (Germany) in 2008. Since 2009 he is working for the German Aerospace Center at the German Space Operations Center (GSOC). He was engaged as Attitude and Orbit Control System as well as Power and Thermal subsystem engineer for TerraSAR-X and TanDEM-X. He was also active as AOCS subsystem engineer and Flight Director for TET-1.

Falk Mrowka is head of the mission planning group of the Department for Mission Operations at the German Space Operations Center (GSOC). He received his physics diploma and his Ph.D. while working at the University of Leipzig on the subject of low temperature physics. While working at the GSOC since 2002, he was working for various projects as mission planning system engineer.

Kay Müller is Flight Operations Manager for the GRACE and future GRACE-FO mission as well as PTS Subsystem engineer for Terra-SAR/TanDEM-X and TET-1. He graduated in Aerospace Engineering at the University of Stuttgart in 2011 and has joined the German Space Operations Centre in the same year.

Andreas Ohndorf works in the Mission Operations department of the German Space Operations Center (GSOC) in several roles: as Flight Director in TSX/TDX, as System Engineer in EnMAP and DEOS, and as Deputy Project Manager in DEOS. Graduated in Aerospace Engineering at the University of the Armed Forces of Germany in 2001 in Munich, he served as a Technical Officer in the German Air Force for 12 years before joining DLR in 2008.

Michael Schmidhuber works in the Department for Mission Operations at the German Space Operations Center (GSOC). As deputy lead of the geostationary satellite projects group he is responsible for system engineering of those projects. Additionally, he is managing the satellite operations training activities. He is also involved in the organization of the biannual international SpaceOps Conference. He graduated as an aerospace engineer at the Technical University of Munich in 1994. After working for a subcontractor company as Spacecraft Subsystem Engineer he joined DLR in 2006 as staff member.

Sina Scholz is Flight Operations Manager for the SATCOMBw Mission. She graduated in Aerospace Engineering from the Technical University of Munich (Germany) in 2007. Since 2008 she is working for the German Aerospace Center at the German Space Operations Center (GSOC). She was engaged as power and thermal subsystem engineer for TerraSAR-X and TanDEM-X. She was also active as Flight Operations Manager for GRACE.

Florian Sellmaier is the head of business development at the German Space Operations Center (GSOC). He received his Ph.D. from the Institute of Astronomy and Astrophysics/Munich in 1996 and also spent some time at CALTECH in Pasadena/USA. After working in the radar and navigations area and as team lead of the Siemens Competence Center for e-Business Applications, he joined DLR in 2006. As a project manager he conducted the acquisition and predevelopment of future mission like OLEV, DEOS, and EDRS. He is the author of several scientific publications in the Astrophysics as well as in the Space Operations area.

Adrian Tatnall joined the department of Aeronautics and Astronautics at the University of Southampton in 1988. Prior to this appointment he was a senior space systems engineer at British Aerospace and was responsible for several contracts concerned with remote sensing instrumentation. From 1999 to 2004 he was the Aerospace Teaching coordinator. He has been the organizer and lecturer of spacecraft engineering courses run for industry at the European Space Agency and Southampton for the last 20 years. For many years he served on the Space Committee of the Royal Aeronautical Society and he is currently acting head of the Astronautics group and a senior tutor in the Faculty.

Thomas Uhlig is Columbus Flight Director at the Columbus Control Center in Oberpfaffenhofen and is leading the training for the European ISS flight controllers. He received his physics diploma and his Ph.D. from the University of Regensburg/ Germany, where he worked on electromagnetic imaging of nanostructures. After working as scientist at the Argonne National Laboratory in Chicago he changed into the spaceflight business, where he worked as Payload expert on the Space Shuttle mission STS-122, before he changed in his present position. He was awarded with the Helmholtz price 2005 of the Physikalisch-Technische Bundesanstalt and is member of the International Astronautical Federation Space Operations Committee. He is the author of various scientific articles.

Stephan Ulamec has more than 20 years of experience in the development and operations of space systems and instruments. After finishing his Ph.D. at the University of Graz, 1991, he worked as a Research Fellow at ESA/ESTEC until 1993 and is since then at the German Aerospace Center, DLR in Cologne. Besides his activities in system engineering and project management of the Rosetta Lander, Philae, he is engaged as payload manager of MASCOT for the Hayabusa 2 mission. He was involved in numerous studies for in situ packages for space research as well as the definition and performance of tests for various mechanisms to be operated on planetary surfaces.

Abbreviations

Symbols

3PO	Planning, cPs, iPs Officer

A

ABM	Apogee Boost Maneuver
ACS	Atmosphere Control and Supply
ACS	Attitude Control System
AD/BD	Acceptance Mode Data/Bypass Mode Data
ADR	Active Debris Removal
AFD	Automated File Distribution System
AFW	Astro- und Feinwerktechnik
AGC	Automatic Gain Control
AI	Artificial Intelligence
AITV	Assembly, Integration, Test, and Validation
ALC	Automatic Level Control
ALSEP	Apollo Lunar Surface Experiment Packages
AMCS	Attitude Measurement and Control System
AMF	Apogee Motor Firing
A.N.	Ascending Node
AOCS	Attitude and Orbit Control System
AOS	Acquisition Of Signal
APID	Application ID
APX	Alpha/X-ray fluorescence
AR	Atmosphere Revitalization
ARTEMIS	Advanced Relay and Technology Mission
ASSC	Automatic Satellite Saturation Control

ATM	Asynchronous Transfer Mode
ATV	Automated Transfer Vehicle
AU	Astronomical Unit
AUX	Auxiliary

B

BAPTA	Bearing And Power Transfer Assembly
BAT	Battery
BLR	Bangalore Ground Station
BME	Biomedical Engineer
BOL	Begin of Life

C

CCD	Charge-Coupled Device
CCS	Central Checkout System
CCSDS	Consultative Committee for Space Data Systems
C&DH	Command and Data Handling
CDR	Critical Design Review
CE	Concurrent Engineering
CESS	Coarse Earth-Sun Sensor
CHeCS	Crew Health Care System
CLCW	Command Link Control Word
CLIP	Columbus Lead Increment Planner
CLTU	Command Link Transmission Unit
CMC	Communication Management Centre
CMD	Command Operator System
CMOS	Complementary Metal Oxyd Semiconductor
CNB	Canberra Ground Station
CNES	Centre National d'Études Spatiales
CoG	Centre of Gravity
Col-CC	Columbus Control Center
COL FD	Flight Director
COL OC	Columbus Operations Coordinator
COMMS	Communications
COP	Command Operations Procedure
COP	Composite Operations Plan
COR	Critical Operations Review
COSMO	Columbus Stowage and Maintenance Officer
COUP	Consolidated Operations and Utilization Plan
CPDU	Command Pulse Distribution Unit
CPS	Consolidated Planning System

CPU	Central Processing Unit
CR	Change Request
CRAF	Comet Rendezvous Asteroid Flyby
CRC	Cyclic Redundancy Check
CSC	Communication Service Centre
CSRD	Current Stage Requirement Document
CSS	Crew Support System
CUP	Composite Utilization Plan

D

DC	Direct Current
DDOR	Delta Differential One-way Ranging
DEM	Digital Elevation Model
DEOS	Deutsche Orbitale Servicing Mission
DFT	Data Flow Test
DH	Data Handling
DIM	Digital Interface Module
D/L	Downlink
DLR	Deutsches Zentrum für Luft- und Raumfahrt (German Aerospace Center)
DMR	Detailed Mission Requirement
DMS	Data Management System
DM	Space Debris Mitigation
DMZ	De-Militarized Zone
DNEL	Disconnection of Non-Essential Loads3
DoD	Depth of Discharge
DOR	Differential One-way Ranging
DPC	Daily Planning Conferences
DS	Daily Summary
DSHL	Disconnection of Supplementary Heater Lines
DTM	Digital Terrain Model

E

EAC	European Astronaut Center
EADS	European Aeronautic Defense and Space Company
EC	Earth Center
ECEF	Earth-Centered, Earth-Fixed
ECLSS	Environmental Control and Life Support System
ECR	Engineering Change Requests
ECSS	European Cooperation for Space Standardization
EDAC	Error Detection And Correction
EDA	Electronically Despun Antenna

EDL	Entry, Descent and Landing
EDRS	European Data Relay System
EEPROM	Electrically Erasable PROM
EFN	Electronic Flight Note
EGSE	Electrical Ground Support Equipment
EIRP	Equivalent Isotropically Radiated Power
EMC	Electromagnetic Compatibility
EMU	Extravehicular Mobility Unit
ENVISAT	Environmental Satellite
EOC	End Of Charge
EO	Earth Observation
EOL	End of Life
EOM	End of Mission
EPDS	Electrical Power Distribution System
EPT	European Planning Team
ESA	European Space Agency
ESOC	European Space Operations Center
ESS	Experimental Servicing Satellite
ESTEC	European Space Research and Technology Centre
EUROCOM	European Spacecraft Communicator
EVA	Extravehicular Activity
EXOSAT	European X-Ray Observatory Satellite

F

FARM	Frame Acceptance and Reporting Mechanism
FCLTU	Forward CLTU
FCT	Flight Control Team
FD	Flight Director
FD	Flight Dynamics
FDIR	Fault Detection, Isolation and Recovery
FDS	Fire Detection and Suppression
FDS	Flight Dynamics System
FGAN	Radio antenna of the „Forschungsgesellschaft für angewandte Naturwissenschaften“
FIFO	First In, First Out
FMECA	Failure Mode Effects and Analysis
FOP	Flight Operations Procedure
FOS	Flight Operations System
FOT	Flight Operations Team
FOV	Field Of View
FSS	First Science Sequence
FTP	File Transfer Protocol

G

GAM	Gravity Assist Maneuver
GCM	Gyro Calibration Mode
GDS	Ground Data System
GDS	Goldstone Ground Station
GEO	Geostationary Earth Orbit
GGR&C	Generic Groundrules, Requirements and Constraints
GOCE	Gravitational Ocean Composition Explorer
GPS	Global Positioning System
GRACE	Gravity Recovery And Climate Experiment
Gr&C	Ground rules and Constraints
GRO	Gamma Ray Observatory
GS	Ground Segment
GSN	Ground Station Network
GSOC	German Space Operations Center
GSQR	Ground Segment Qualification Review
GSYS	Ground System
G/T	Gain over Noise Temperature
GTO	Geostationary Transfer Orbit
GUI	Graphical User Interface

H

HCP	Health-check Parameters
HEO	Highly Elliptical Orbits
HF	High Frequency
HGA	High Gain Antennae
HLP	Horizontal Linear Polarized
HMI	Human Machine Interface
HPA	High-Power Amplifier
HPTC	High-Priority (or High-Power) Telecommand
HST	Hubble Space Telescope
HTV	H-II Transfer Vehicle

I

ICD	Interface Control Document
IDRD	Increment Definition and Requirement Document
IEPT	International Execute Planning Telecon
I/F	Intermediate Frequency
IFM	In-Flight Maintenance
IMU	Inertial Measurement Unit

IOAG	Interagency Operations Advisory Group
IO	Intermediate Orbit
IOT	In-Orbit Test
IP	Internet Protocol
IPV	International Procedure Viewer
IRES	InfraRed Earth Sensor
IRIG	Inter-range Instrumentation Group
IR	InfraRed
ISDN	Integrated Services Digital Network
ISO	International Organization for Standardization
ISRO	Indian Space Research Organization
ISS	International Space Station
IT	Information Technology
ITU	International Telecommunication Union

J

JAXA	Japan Aerospace Exploration Agency
JPL	Jet Propulsion Laboratory
JSpOC	Joint Space Operations Center

L

L2	Langraingian Point 2
LAN	Local Area Network
LCC	Lander Control Center
LEO	Low Earth Orbit
LEOP	Launch and Early Orbit Phase
LGA	Low Gain Antennae
LHCP	Left-Hand Circular Polarized
LM	Lunar Module
LNA	Low-noise Amplifier
LOF	Local Orbital Frame
LOS	Loss Of Signal
LRP	Long Range Planner
LS	Logistics Support
LTS	Long Term Science

M

MAC	Media-Access Control
MAD	Madrid Ground Station
M.A.I.T.	Manufacturing, Assembly, Integration and Test

MAP ID	Multiplexing Access Point Identifier
MASCOT	Mobile Asteroid Surface Scout
MCC-H	Mission Control Center Houston
MCC	Mission Control Centre
MCC-M	Mission Control Center Moscow
M&C	Monitoring and Control
MCS	Monitoring and Control Subsystem
MCS	Motion Control System
MD	Mission Director
MEA	Mean Error Amplifier
MedOps	Medical Operations
MEO	Medium Earth Orbit
MER	Mars Exploration Rover
MGA	Medium Gain Antennae
MIB	Mission Information Base
MIPD	Multi-Increment Planning Document
MIPROM	Multi-Increment Payload Resupply and Outfitting Model
MiTEx	Micro-Satellite Technology Experiment
MLI	Multilayer Insulation
MMI	Man–machine Interface
MOD	Mission Operations Director
MOS	Mission Operation System (Flight Director)
MPCB	Multilateral Payloads Control Board
MPLM	Multipurpose Logistic Module
MPLS	Multi-protocol Label Switching
MPPT	Maximum Power Point Tracker
MPS	Mission Planning Subsystem
MRO	Mars Reconnaissance Orbiter
MSL	Mars Science Laboratory
MW	Momentum Wheel

N

NASA	National Aeronautics and Space Administration
NASCOM	NASA Communications Network
NAS	National Airspace System
NAS	Network-Attached Storage
NCR	Non Conformance Reports
NEAR	Near Earth Asteroid Rendezvous
NIS	Network Information Service
NMC	Network Management Centre
NOPE	Network Operator
NORAD	North American Aerospace Defense Command

O

OASPL	Overall Acoustic Sound Pressure Level
OBC	On-Board Computer
OBCP	On-Board Control Procedure
OBDH	On-Board Data Handling
OBSW	On-Board Software
ODF	Operations Data File
OD	Orbit Determination
OLEV	Orbital Life Extension Vehicle
OMT	Orthomode Transducer
OOS	On-Orbit Operations Summary
OOS	On-Orbit Servicing
OPS	Operations
ORR	Operational Readiness Review
ORU	Orbital Replacement Unit
OSAT	Roentgen Satellite
OSIRIS	Optical, Spectroscopic, and InfraRed Remote Imaging System
OSTP	On-Orbit Short-Term Plan

P

PAC	Packet Assembly Controller
PCDU	Power Control and Distribution Unit
PCM	Pulse Code Modulation
PC	Personal Computer
PD	Proportional Differential
PDR	Preliminary Design Review
PDU	Power Distribution Unit
PICB	Program Integration Control Board
PID	Proportional Integral Differential
PLL	Phase-Locked Loop
PMC	Private Medical Conference
PMD	Post-Mission Disposal
PM & P	Parts, Materials and Preocesses
PM	Processor Module
PM	Project Manager
POIC	Payload Operations and Integrations Center
PO	Project Office
PPC	Private Psychological Conference
PPCR	Planning Product Change Request
P	proportional
PRN	Pseudo Random Noise
PROM	Programmable Read-Only Memory

PROP-F ПрОП-Ф (Прибор оценки поверхности - Фобос) Russian acronym for "Mobile Robot for Evaluation of the Surface of Phobos"
PR Planning Request
P/T Power Thermal
PTP Payload Tactical Plan
PTS Power and Thermal System
PUS Packet Utilization Standard
PVT Pressure–Volume–Temperature

Q

QOS Quality of Service
QR Qualification Review

R

RAAN Right Ascension of Ascending Node
RAF Return All Frames
RCF Return Channel Frames
RCS Robotic Control System
RDUA Remote Data Unit A
RFE Radio Frequency Electronics
RF Radio Frequency
RF Sensor Radio Frequency Sensor
RHCP Right-Hand Circular Polarized
RID Review Item Discrepancy
RM Reconfiguration Module
RPM Revolutions Per Minute
RSA Russian Space Agency
RTG Radioisotope Thermoelectric Generator
RW Reaction Wheel
RX Receiver

S

SAM Sun-Acquisition Mode
SAN Storage-Area Network
SAR Satellite Anomalies Reports and Recommendations
SAR Synthetic Aperture Radar
SBC Satellite Board Computer
SBM Standby Mode
SCC Satellite Control Centre

S/C	Spacecraft
SDH	Synchronous Digital Hierarchy
SE	System Engineer
SGM	Safeguard Memory
S/G	Space to Ground
SLE	Space-Link Extension
S&M	Structure and Mechanics
SoE	Sequence of Events
SONC	Science Operations and Navigation Center
SO	Security Officer
SPACON	Spacecraft Controller
SP	Sun Presence
SSB	SLE Switch-Board
SSCB	Space Station Control Board
SSIPC	Space Station Integration and Promotion Center
SSMM	Solid State Mass Memory
SSPA	Solid State Power Amplifier
SS	Space System
SS	Sun Sensor
STL	Satellite Team Lead
STP	Short-Term Plan
STRATOS	Safeguarding, Thermal, Resources, Avionics, Telecommunications, Operations, Systems
STS	Space Transportation System
ST	Star Tracker
SV	Space Vehicle
S/W	Software

T

TAFF	TanDEM-X Autonomous Formation Flying Experiment
TAR	Technical Acceptance Review
TCP	Transmission Control Protocol
TCR	Telemetry, Commanding and Ranging
TCS	Thermal Control System
TC	Telecommand
TDM	TanDEM-X Mission
TDRSS	Tracking and Data Relay Satellite System
TDRS	Tracking and Data Relay Satellite
TDX	TanDEM-X Satellite
TEGA	Thermal Evolved Gas Analyzer
THC	Temperature and Humidity Control
TIM	Technical Interface Meetings
TLE	Two-Line Elements
TM/TC	Telemetry/Telecommmand

TM	Telemetry
TRL	Technology Readiness Level
TSM	TerraSAR-X Mission
TSTD-MPS	TerraSAR-X Tandem-X Mission Planning System
TSX	TerraSAR-X Satellite
TTC-RF	Telemetry, Tracking and Command at Radio Frequency
TTTC	Time-tagged Telecommand
TWTA	Traveling Wave Tube Amplifier
TX	Transmitter

U

UHF	Ultra High Frequency
U/L	Uplink
UPS	Unified Propulsion System
UPS	Uninterruptible Power Supply
USOC	User Support and Operations Center
UT1	Universal Time 1
UTC	Universal Time Coordinated
UV	Ultra Viole

V

VC	Virtual Channel
VLBI	Very Large Base Interferometry
VLP	Vertical Linear Polarized
VoIP	Voice over IP
VPN	Virtual Private Networking
VSAT	Very Small Aperture Terminal

W

WAN	Wide-Area Network
WFC	Wide Field Camera
WHM	Weilheim Ground Station
WLP	Weekly Lookahead Plan
WPC	Weekly Planning Conference
WRM	Water Recovery and Management
WSP-C	Weilheim Service-Provider-Cortex

X

XMM-Newton	X-ray Multi-Mirror Mission

Contents

Chapter 1
Overview Space Segment

1.1 The Space Environment

Adrian R.L. Tatnall

1.1.1 Introduction

The space environment in which spacecraft have to operate is an alien world in which we would not survive for more than a few minutes without protection. Fortunately, in this respect, spacecraft are generally more robust than humans and it is possible for spacecraft to regularly operate continuously for more than 15 years. In the case of Voyager 1, launched over 35 years ago, the spacecraft continues to operate and communicate with the earth 18 billion km away. It is interesting to question how this longevity can be achieved when there is no possibility of maintenance and the environment, at first sight, appears so unattractive.

To understand these issues the space environment is considered. The answer is that whilst the space environment is different many of the sources of erosion and wear on Earth are not present in space.

Whilst the space environment is alien, it is only remote in the sense that it is difficult and costly to get into. Space is generally considered to start at the Karman line at 100 km altitude in the thermosphere (Fig. 1.1). The short trip of 100 km does represent a major challenge for rockets and the trip itself subjects the spacecraft to a totally different environment to that which it is subjected on the ground or in space.

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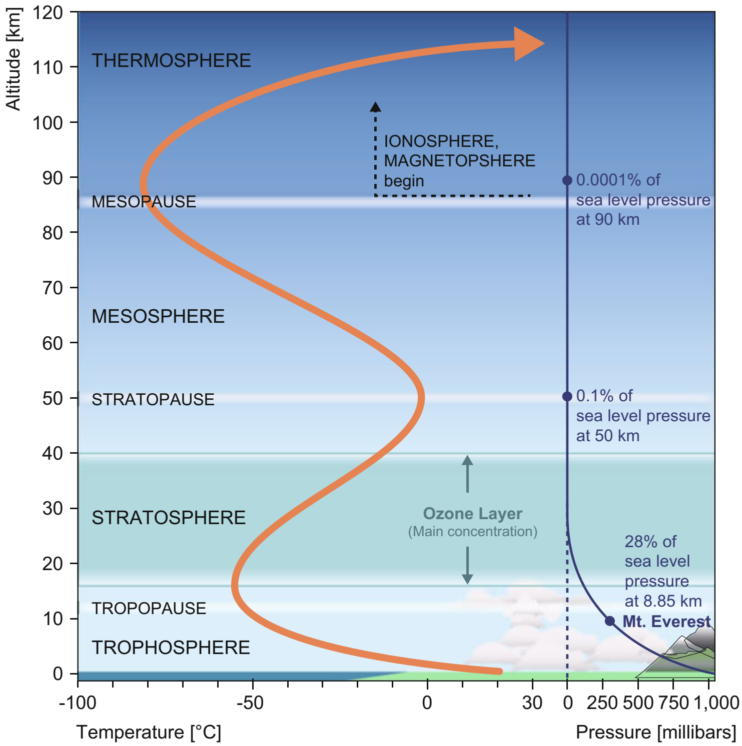

Fig. 1.1 Earth's atmosphere (adapted from Encyclopaedia Britanica Inc.)

1.1.2 Launch Vehicle

1.1.2.1 Acoustic/Vibration levels

Anybody who has witnessed a launch will testify to the noise levels that are produced. At the moment of launch the rocket motor firing and the exhaust products reflected from the ground produce a peak in the acoustic/vibration environment. As the rocket ascends the ground contribution decreases but other mechanical moving parts and unsteady aerodynamic phenomena continue to excite the structure. This excitation of the structure produces a secondary acoustic field within the structure. As the speed of the rocket increases a further secondary peak in the acoustic field occurs during the transonic flight which typically occurs just below Mach 1, the speed of sound. The overall levels experienced within the fairing of the Ariane V and Falcon 9 rocket are shown in Fig. 1.2. Acoustic noise affects lightweight structures, and antenna parabolic reflectors, solar arrays and spacecraft panels are particularly vulnerable.

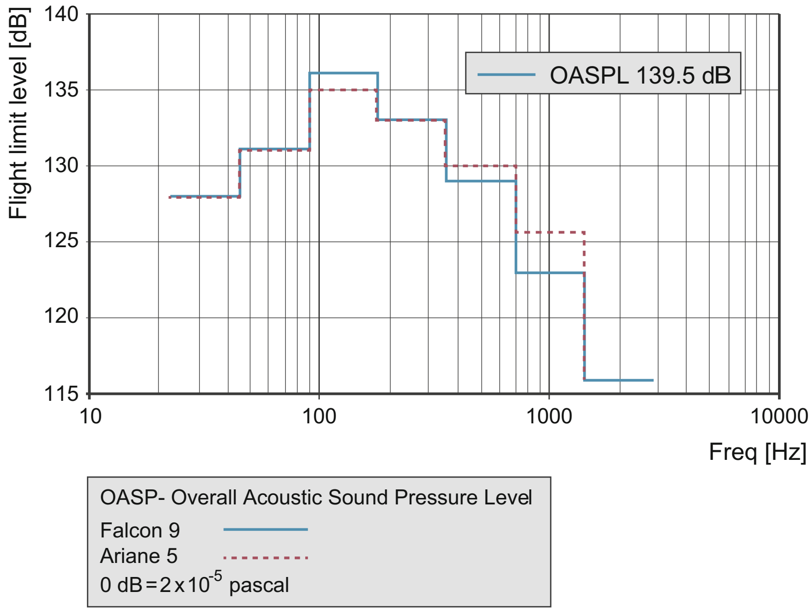

Fig. 1.2 Acoustic noise levels (data taken from Ariane 5 users manual 2011, Arianespace and Falcon 9 Launch Vehicle Payload User's Guide 2009, Space X)

1.1.2.2 Static Acceleration

At the moment of launch the rocket has its highest mass and the acceleration is correspondingly low as the thrust produced is virtually constant. As the propellant is used the rocket acceleration increases until the solid rocket booster flame out and separation occurs. This gives the distinctive launch static acceleration profile shown in Fig. 1.3. Since the accelerations vary with time, the effect on the spacecraft is to generate quasi-static loads. These loads determine the major load bearing parts of the spacecraft structure such as the central thrust tube.

1.1.2.3 Mechanical Shock

A number of events can lead to very high accelerations being produced for very short periods of time. These shocks include

- Ignition and separation of the launch vehicle stages
- Fairing jettison
- Spacecraft separation
- Docking and landing

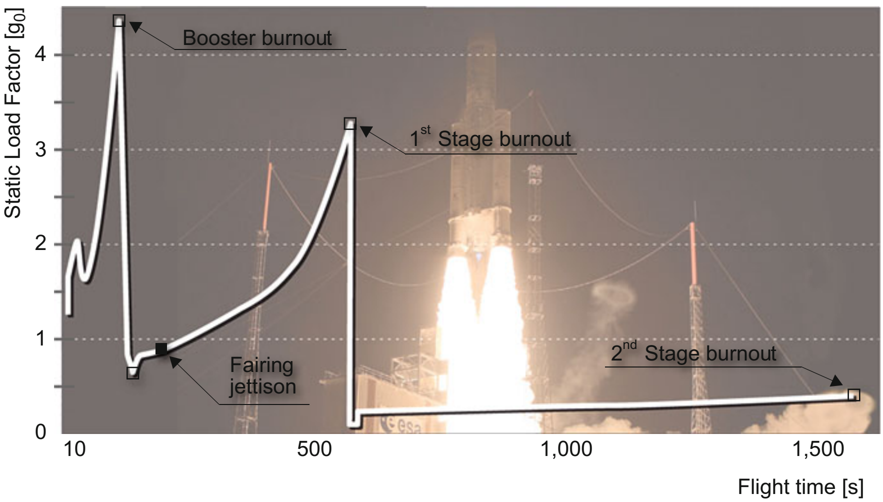

Fig. 1.3 Ariane 5 static acceleration profile

On Ariane and Falcon 9 the peak excitation occurs in the range 1–10 KHz and is 2,000 g_0 for Ariane 5 and 3,000 g_0 for Falcon 9. Despite these very high figures the transient nature of these loads means that they do not usually affect structural strength but they are of concern to the functioning of equipment such as relays.

1.1.3 Spacecraft Operational Environment

1.1.3.1 Vacuum

By the time a spacecraft reaches the low earth orbit at 300 km, the ambient pressure is as low as that which could be achieved by a very good vacuum chamber (about 10^{-7} Pa) on the earth and by the time a spacecraft has reached 800 km the pressure is so low that it cannot be reproduced on the ground. It is therefore important that materials which do not outgas are used in the construction of a spacecraft. Outgassing occurs because the material itself sublimates, gases are released from cracked materials, or gases that are adsorbed by the surfaces are released in a near vacuum. Whilst this will probably not cause a problem to the structural integrity of the spacecraft, it might result in a change to the surface properties and there is always the possibility that the vaporised material will condense if it impinges on colder spacecraft surfaces. It is therefore important that materials such as cadmium, zinc, PVC and many plastics with high vapour pressures are avoided. Adsorbed gases can be responsible for changing the properties of materials. Graphite is a solid lubricant commonly used on Earth but in space the adsorbed water vapour is lost and graphite is ineffective as a lubricant. Alternatives such as molybdenum disulphide have to be used. If a material affected by a vacuum has to be used, residual contaminants can be removed by baking and the application of protective coatings or shielding.

1.1.3.2 Solar Radiation Flux

The spectrum of the radiation from the sun approximates to that of a black body at a temperature of 5,800 K. This is the temperature of the photosphere, the opaque region of the sun which is usually considered to be the surface of the sun. Our eyes have a response which is optimised for the light which is emitted by the sun which peaks at about 550 nm. This radiation is virtually constant and varies by less than 1 % from sunspot maximum to sunspot minimum although there are seasonal variations in the radiation incident on the earth as the earth moves in an elliptical orbit round the sun. Radiation we don’t see, however, is much more variable. Rather than originating from the photosphere, the UV and X-rays originate from the outer regions of the sun, i.e. the chromosphere and the corona. This can be understood by noting that the temperature increases as the distance from the sun increases. By the time the corona is reached at a distance of about 2,500 km from the sun’s surface, the temperature is about a million degrees and so hot that it emits radiation at X-ray wavelengths. The corona is highly variable in timescales of seconds to months and this is reflected in the variability of the UV and X-rays produced. Whilst the terrestrial weather is undoubtedly influenced by variations of the sun, as the overall variations in energy output are very small, it is not easy to distinguish these terrestrial variations from the much larger natural variability of our weather. The space weather, however, is dominated by variations in the sun output as this has a profound impact on the UV, X-rays and particles impacting the earth. UV radiation has a direct impact on the materials used in spacecraft, particularly the solar array. The absorption of UV by the cover glass used to protect the solar cell from particle radiation and the solar cell to slide adhesive can lead to darkening. This has a dual effect; it reduces the cell illumination and so reduces electrical power produced and it also raises the temperature of the cell which leads to a reduction in efficiency. Doping the cover glass with cerium oxide leads to absorption of UV and the prevention of darkening.

Additionally, the solar radiation flux is responsible for the radiation pressure created by absorbed or reflected photons. It is this source which provides the force for “solar sailing”, a means of controlling or propelling a spacecraft. An experimental spacecraft of the Japan Aerospace Exploration Agency (JAXA) called IKAROS successfully flew a spacecraft to Venus using solar sailing (Osamu Mori et al. 2009).

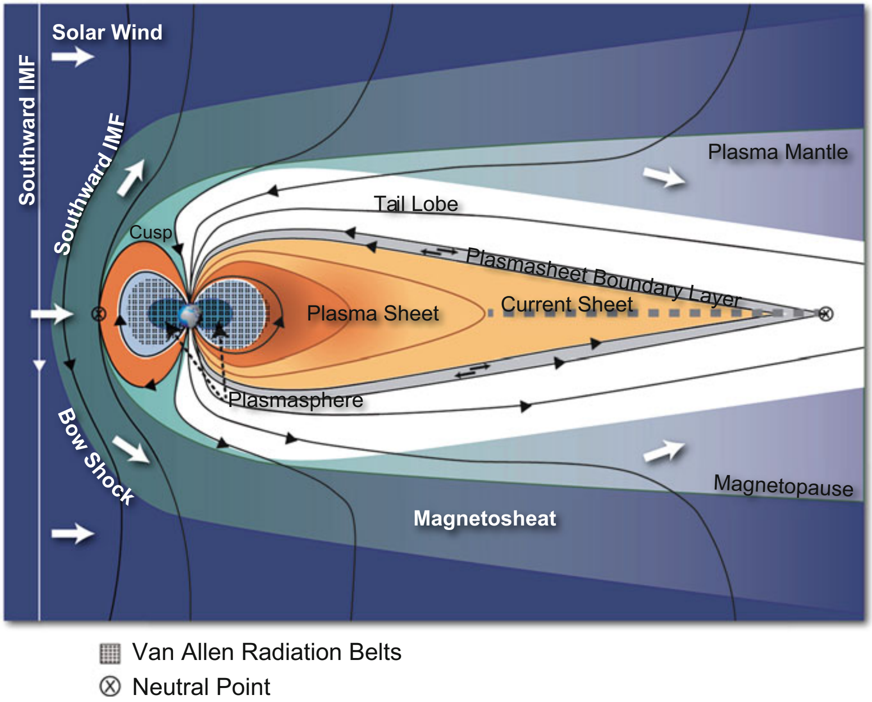

Fig. 1.4 Earth's magnetosphere (Reiff 1999)

1.1.3.3 Particle Radiation

There is a continual stream of high energy particles emitted by the sun. These particles are mainly protons and electrons with an energy of 1.5–10 keV. They are moving at speed of 400–800 km/s. This constitutes the solar wind that pervades our solar system and extends to at least 100 AU from the sun. Despite the high speed of the particles in the solar wind, their density is only 5–10 atoms/cm^3 rising to a few hundred atoms/cm^3 during a time of high solar activity. This provides a negligible pressure on any spacecraft that is impacted by the solar wind. A far greater pressure comes from the light pressure of photons as described above. Whilst the pressure of the solar wind is negligible, the consequences of the solar wind have to be considered, however, because it does have a major impact on the environment of the earth. The plasma of the solar wind interacts with the earth's dipole magnetic field to form the magnetosphere shown in Fig. 1.4. The magnetosphere's distinctive asymmetric shape is due to the pressure of the solar wind. On the sunward side, the magnetosphere extends out to a distance of approximately 10 Earth radii under quiet conditions, whilst in the anti-sun direction it extends several hundred Earth radii. The shape and extent of the magnetosphere depends on the strength and orientation of the magnetic field of the solar wind. This determines the reconnection

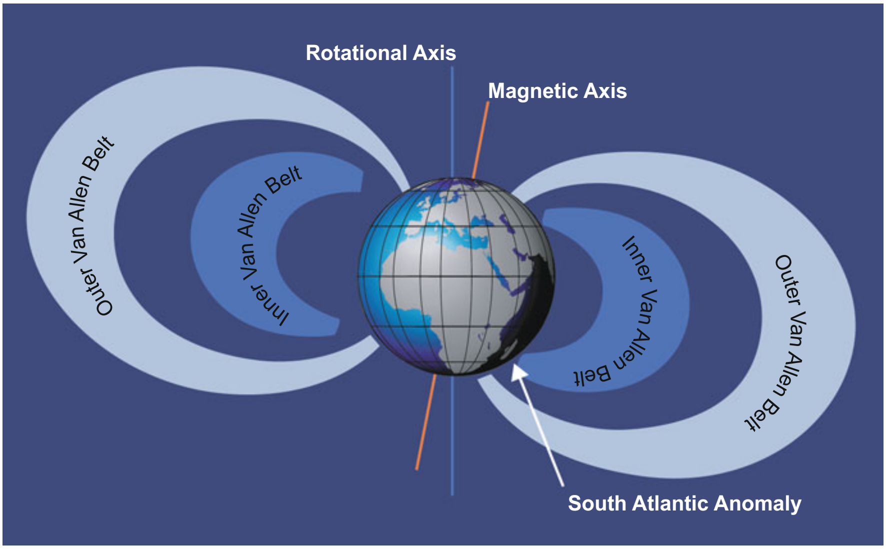

Fig. 1.5 Radiation belts inner and outer Van Allen radiation belts

process of the earth's and the sun's magnetic field that allows energy and momentum to be transferred from the solar wind into the magnetosphere. It may also be the acceleration mechanism for the very high energy particles that can be found in the radiation belts within the magnetosphere.

1.1.3.4 Radiation Belts

In 1958 the existence of a belt of trapped charged particles around the earth was confirmed by Explorer 1 and 3 using instrumentation designed by James Van Allen who had predicted that the belts would exist. The belt detected was the inner radiation belt. In the same year the Soviets—S. N. Vernov and A. E. Chudakov—discovered the outer radiation belt. These belts are doughnut shaped and extend from 1,000 km to 60,000 km above the earth. The outer belt is predominantly made up of electrons with a peak at 15,000–20,000 km, whereas the inner belt consists largely of high energy protons that peak at 3,000 km. Proton energies range from 0.01 to 400 MeV and electron energies from 0.4 to 4.5 MeV. They are shown as two distinct belts in Fig. 1.5 but in practice there is no real gap between the belts and they are highly variable depending on solar activity. The location of the radiation belts follows the magnetic field of the earth and this means that they are not symmetrically placed with respect to the earth. The axis of this field is offset and tilted with respect to the earth's rotation axis and so this leads to a location over the South Atlantic where the magnetic field is anomalously low. As a result the radiation belts are closer to the earth over this region which is commonly known as the South Atlantic Anomaly. A satellite in a low earth orbit is more likely to encounter energetic particles and hence suffer damage in this part of the world.

A geomagnetic storm is caused by a solar wind shock wave interacting with the earth's magnetic field. This leads to measurable changes on the earth's surface of the earth's magnetic field. Accompanying these changes are increases in charged particles in the radiation belts. These particles are subject to the magnetic fields and perform three types of motions. All particles spiral around the field lines, move down field lines and bounce from one hemisphere to another and drift around the earth. This last motion eastward for electrons and westward for protons produces a current known as the ring current which can be measured by observing the associated magnetic field on the surface of the earth. It can lead to a decrease in the magnetic field measured at the earth's surface of $>1\ \%$ during a major geomagnetic storm.

The origin of the particles in the radiation belts is solar, terrestrial or cosmic. Particles of solar origin are injected into the outer belts during magnetic storms. It is believed that the protons of the inner belt originate from the decay of neutrons produced when high-energy cosmic rays from outside the solar system collide with atoms and molecules of Earth's atmosphere.

Radiation effects include total dose effects, e.g. CMOS problems, lattice displacement damage which can damage solar cells and reduce amplifier gain, single event effects and additional noise in sensors and increased electrostatic charging. The charging of a spacecraft relative to the surrounding plasma is not so much of a problem as the possibility of increased discharges that can damage equipment and lead to the generation of electromagnetic interference. This has traditionally been thought to be more of a problem in GEO (Geostationary Earth Orbit) than LEO (Low Earth Orbit) where the plasma is low in energy and high in density, but nevertheless in LEO and particularly over the polar regions high levels of spacecraft surface charging can occur.

1.1.3.5 Atmosphere

When the sun is active and the amount of UV emitted increases, this is reflected in increased heating and expansion of the atmosphere so that the atmospheric drag on spacecraft increases also in region above the actual atmosphere. The change in height of the International Space Station (ISS) is shown in Fig. 1.6. Reductions in altitude caused by atmospheric drag are compensated by boosts using the station's thrusters. The number of boosts required depends on the atmospheric drag and the height variation permissible. For a spacecraft like GOCE which has to be maintained at a constant height and is at a very low altitude in order to measure small changes in the gravity field, this means thrusters have to be used for long periods of time. This is why electric propulsion thrusters are appropriate. Objects and spacecraft which are not controlled, such as debris, will lose height more quickly when the sun is active and the atmosphere has expanded.

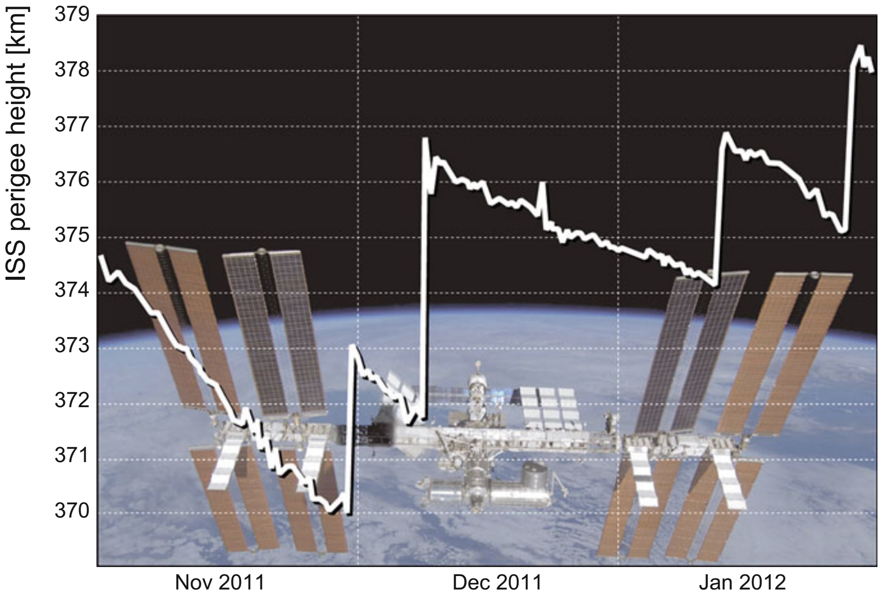

Fig. 1.6 ISS Perigee height change

The effect of solar activity on space debris is illustrated by Fig. 1.7 which shows that the reductions in debris occur at times of solar maximum when atmospheric drag is at its greatest, e.g. 1989. There have been recent suggestions based on the recent low activity of the sun that the sun is entering a new phase which will be characterised by low solar activity. This will have implications for satellites and debris. The effect of drag on a spacecraft provides a force give by

$$\boldsymbol{F}_{\mathrm{D}} = \frac{1}{2}\rho A C_{\mathrm{D}} V_{\mathrm{r}}^{2}\left(\frac{-\boldsymbol{V}_{\mathrm{r}}}{|\boldsymbol{V}_{\mathrm{r}}|}\right) \tag{1.1}$$

where $\boldsymbol{V}_{\mathrm{r}}$ is the velocity vector relative to the atmosphere, ρ the atmospheric density, A the area of the vehicle perpendicular to the flight direction and C_{D} is the coefficient of drag which is typically ~2.5. Spacecraft with high area and low mass are particularly likely to be affected.

The lifetime of a spacecraft as a function of altitude and the mass area ratio m/A is shown in Fig. 1.8 for a representative atmosphere. In order to meet regulations imposed from January 2010 that all spacecraft in LEO must be deorbited within 25 years, this has resulted in a number of proposals to achieve this by deploying a structure that will greatly increase the area of the spacecraft and hence increase atmospheric drag.

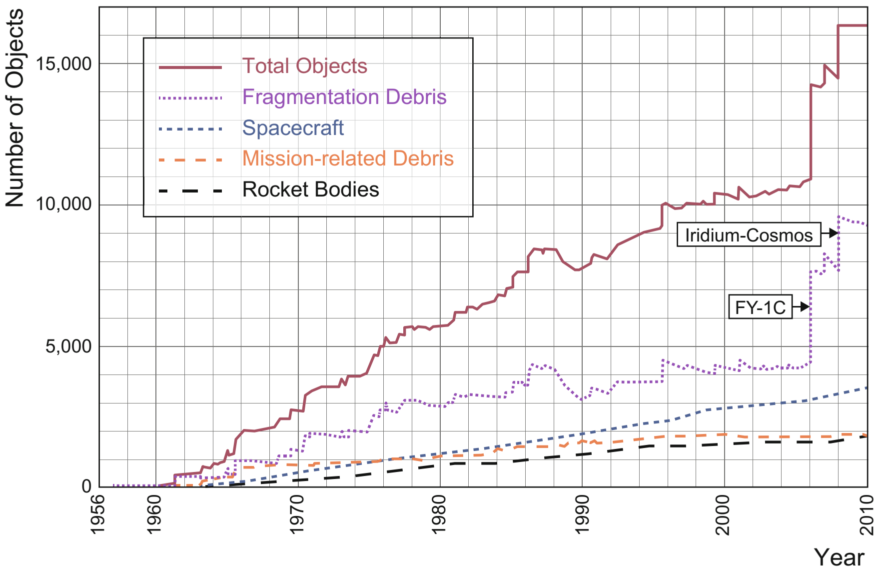

Fig. 1.7 Objects in the earth orbit by object type (NASA)

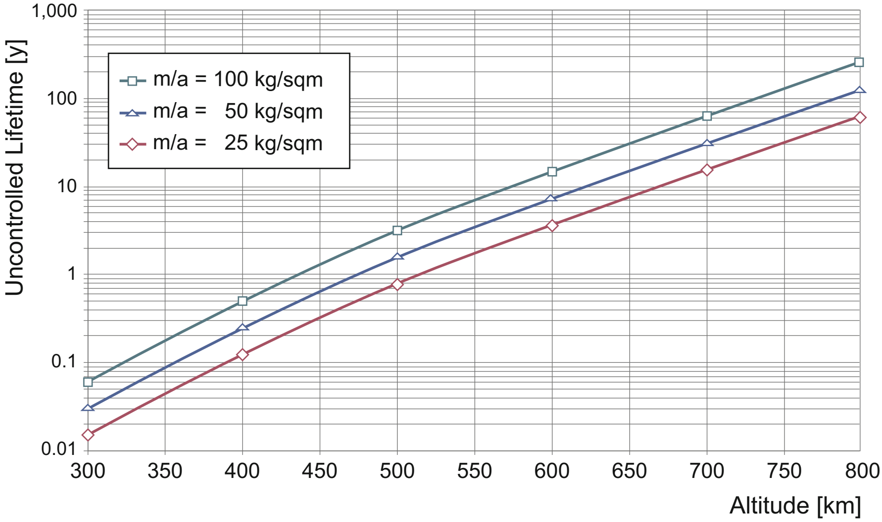

Fig. 1.8 Spacecraft lifetime as a function of altitude and mass/area ratio [m/a] (Kahn 2012)

1.1.3.6 Debris

Ever since the earth has been in existence it has been impacted by material. The mass flux of this material is currently about 10^7–10^9 kg/year. Much of this material is dust-sized objects called micrometeoroids that have a mass of less than 1 g. Their velocities relative to spacecraft averages about 10 km/s and so whilst they are not likely to cause catastrophic damage to spacecraft they do contribute to the weathering process and can modify material properties. Objects larger than 1 g do exist and over the lifetime of the earth it has been hit by many objects of over 1 km in diameter. It is thought that 65 million years ago a 10 km meteorite hit the Yucatan peninsula in Mexico and produced a crater 180 km in diameter and probably caused the mass extinction of the dinosaurs. Of more concern to spacecraft is the increase in natural debris that occurs when the earth moves through the debris of a comet and the earth undergoes a meteor shower. The Olympus communications spacecraft was damaged by one of the Leonid meteoroids in 1993 and subsequently suffered an electrical failure.

Manmade debris is a growing problem as illustrated in Fig. 1.7. Whilst the problem has been gradually growing since man first started launching satellites, it has been exacerbated and highlighted by some recent events that have contributed to large amount of debris. One of these was the Chinese destruction of satellite FY-1C in an anti-satellite test in 2007 and another was the 2009 satellite collision between Iridium 33 and Kosmos-2251. The impact of both these events is shown in Fig. 1.7.

There are several contributors to the debris population:

- Launch and operational debris
- Space vehicle breakup (~215 events to date, 57 of them deliberate)
- Explosions
- Collision induced (5 to date—latest Iridium 33/Kosmos 2251, 10 Feb 2009)
- Upper stage breakup (largest contribution—Breeze M in 2007, 2010, 2011, 2012)
- Shedding of spacecraft surfaces (paint, MLI, etc.)
- Liquid metal coolant droplets
- Sodium–potassium (Na K) droplets from RORSAT reactor cores
- Solid propellant motor firings
- ASAT operations (Fengyun-1C, 11 Jan 2007; USA 193 21 Feb 2008)

The only current debris sink for the low earth orbit is the atmosphere, although there are a large number of innovative solutions being considered. These include electromagnetic methods, momentum exchange methods, remote methods, capture methods, and modification of material properties or change-of material state. If it is possible to remove debris and to enforce a requirement that satellites should have a lost mission lifetime of no more than 25 years, Fig. 1.9 illustrates that it could be possible to stabilise the debris environment. 90 % PMD (post-mission disposal) means that in 90 % of satellites the 25 year rule was implemented and ADR 2020/02 and ADR 2020/05 means that 2 and 5 objects per year respectively were removed after 2020 (ADR-active debris removal). Although the scenario with 90 % PMD and the removal of 5 objects per year suggest stabilisation, it does not take into account the possibility of unpredictable events such as the loss of Envisat, an 8 tonne Earth Observation satellite in April 2012. It is still in one piece but it is not controllable and constitutes a space debris threat as there is a distinct possibility

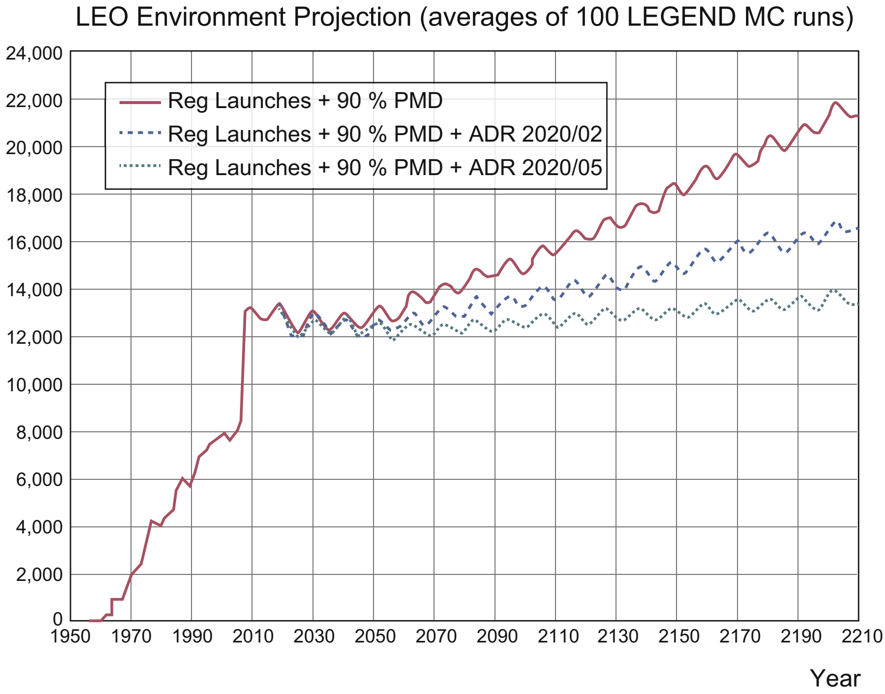

Fig. 1.9 Debris as a function of active debris removal and 25 year rule (Liou 2011)

that it will be struck by other debris. An analysis of space debris at Envisat's orbit suggests there is a 15–30 % chance of collision with another piece of junk during the 150 years it is thought Envisat could remain in orbit. Should this collision happen a very large debris cloud will be produced in a widely used region of space.

1.1.3.7 Gravity and Magnetic Fields

In addition to the environmental torques that can be provided by atmospheric drag and solar radiation, there are also gravity gradient torques and magnetic torques. The former are due to the differential gravity forces between the top and the bottom of the spacecraft and can be used to maintain a spacecraft earth pointing to about $\pm 5°$. Magnetic torques are caused by the earth's magnetic field acting on the residual magnetic dipole moment of the spacecraft. It can be utilised to provide a control torque by generating a controllable magnetic dipole moment on the spacecraft that interacts with the earth's magnetic field and generates a torque. In addition to being lightweight they require no expendable resources. They do require a significant external field and so can only be used for low earth orbiting missions.

1.2 Space Systems Engineering

Adrian R.L. Tatnall

1.2.1 Definition of System Engineering

System engineering requires skills that are traditionally associated with both art and science. Good system engineering requires the art of technical leadership including creativity, problem solving, knowledge and communication skills but it also requires the science of systems management or the application of a systematic disciplined approach. In this section the systematic disciplined approach balance is considered in more detail with the emphasis on the methodology of systems engineering but the main goal of system engineering is to get the right design. This can only be done using skills that cross traditional boundaries between the arts and the sciences.

The definition of systems engineering is an interdisciplinary approach governing the total technical effort to transform requirements into a system solution. The European Standard for Space System Engineering is described in the European Cooperation for Space Standardisation (ECSS) with the document number ECSS E ST 10C. The system can be any integrated product or processes that provide a capability to meet a stated objective. This inevitably means that a system can be a subsystem of a larger system and/or a system of systems. A spacecraft is a system but it is one element of the space mission that will include the launch vehicle and the ground segment and may include other systems such as GPS (Global Positioning System) and a data relay system. The ground segment itself is a combination of systems that is responsible for spacecraft operations and the processing of the data. It is therefore often necessary to consider products at a number of different levels.

The boundaries of the system engineering discipline and its relationship with production, operations, product assurance and management disciplines are given in Fig. 1.10 taken from the ECSS-E-ST-10C.

System engineering encompasses the following functions:

- Requirement engineering, which includes requirement analysis and validation, requirement allocation, and requirement maintenance.
- Analysis, which is performed for the purpose of resolving requirements conflicts, decomposing and allocating requirements during functional analysis, assessing system effectiveness (including analysing risk factors), complementing testing evaluation and providing trade studies for assessing effectiveness, risk, cost and planning.

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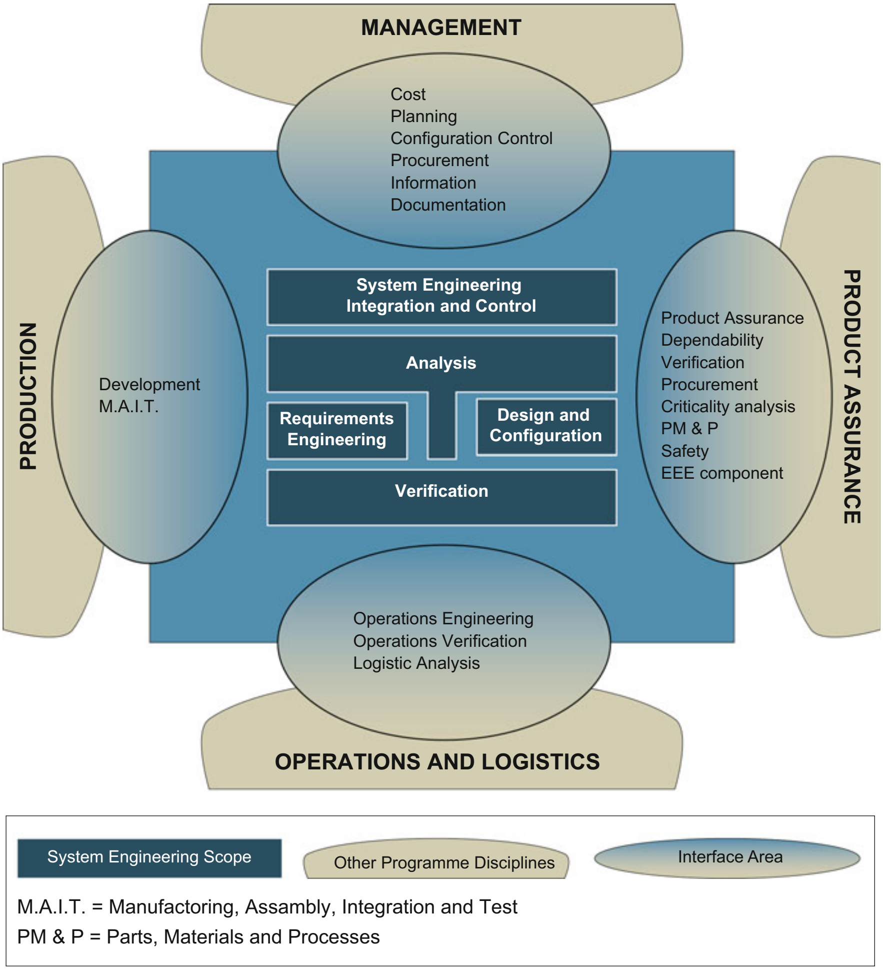

Fig. 1.10 System engineering boundaries (reproduced from ECSS E ST 10C, credit: ESA)

- Design and configuration which result in a physical architecture, and its complete system functional, physical and software characteristics.
- Verification, whose objective is to demonstrate that the deliverables conform to the specified requirements, including qualification and acceptance.
- System engineering integration and control, which ensures the integration of the various engineering disciplines and participants throughout all the project phases.

These functions require the techniques defined in Table 1.1 to be used.

Table 1.1 System engineering techniques (Fortescue et al. 2011)

Requirements identification/analysis	Concept selection
System specification	Budget allocation
Options identification	Performance analysis
Mission assessments	System optimisation
Trade-offs	Interface specification
Feasibility assessment	System definition
Cost comparison	Cost estimation

1.2.2 Objectives and Requirements

The starting point for the mission is the mission statement: a document established by the customer, which reflects the user needs. It is often a single line that describes the mission, e.g. John F. Kennedy in 1960 said that we would put a man on the moon by the end of the decade. The mission objectives are derived from this statement and qualitatively define what this mission should accomplish.

The mission requirements are the top level requirements on all aspects of the mission. They are usually quantitative in nature, specified by the customer or user, and they are an assessment of the performance required to meet the mission objectives. For the spacecraft system design these requirements are translated into engineering parameters. This translation can be complex, depending on the particular application. The requirements drive the rest of the design and determine all aspects of the mission. They are the single biggest cause of project problems.

For a communication spacecraft the translation between the user requirements and the engineering requirements is relatively straightforward since the user coverage and data requirements can readily be used to define the satellite parameters. On an Earth Observation and Science spacecraft, however, they can be considerably more complex. For example, on the GRACE (Gravity Recovery and Climate Experiment) mission (Wiese et al. 2012), the user requirements on geophysical parameters, such as an ice sheet changes into an instrument specification, have to be translated into measurements of the gravity field and ultimately to the measurements of changes in the speed and distance between two identical spacecraft. In this case the process involves assumptions about other related parameters such as the level of processing required and GPS data. At the start of the design process it may not be clear to what extent the requirements are driving the design and so it is an essential part of the space system engineering process that the requirements are re-evaluated when there is a clearer understanding of the impact they have on the spacecraft design.

This iterative process is essential to ensure that the most relevant and realistic requirements are used for the spacecraft design. There are plenty of examples where the engineering requirements have become “tablets of stone” at the start of the design and the overall system has suffered because of unwillingness to question them as the design has evolved. It is always necessary to define how much quality is needed, or how much “science” is enough in order to hold down mission costs and

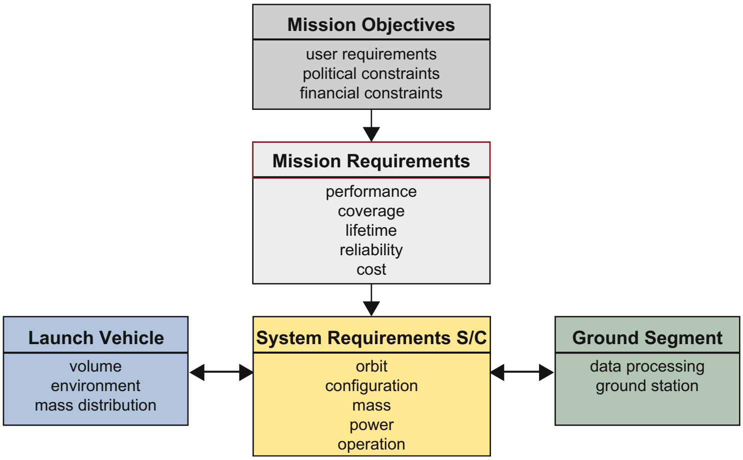

Fig. 1.11 Objectives and requirements of a space mission

avoid unnecessarily restrictive requirements. Whilst Augustine's law that "the last 10 % of performance generates one-third of the cost and two-thirds of the problems" is an oversimplification it does encapsulate the problem of overspecification. In other examples technological constraints such as the inability to space qualify critical parts or processes that may dictate a revision of requirements. The importance of the requirements should not be underestimated. Relatively little of the total project budget is spent on requirement analysis and initial design but it does determine the cost commitment for the rest of the programme. The later the change in a requirement the greater the cost impact on the programme as a whole.

Figure 1.11 shows how it is necessary to expand these top-level requirements into specifications covering the entire range of system and subsystem engineering parameters. It also shows the importance of establishing, in parallel, budget data.

Table 1.2 is a checklist of the full range of parameters that are likely to be specified in later, more detailed phases of a programme.

There are many systems options that have to be considered in the early design phase of a mission. These include the type of orbit, the launcher, the propulsion system, the type of spacecraft configuration and the attitude control concept.

The choice of orbit for an astronomy mission is a good example of the kind of choices that have to be made. This highlights some of the key points that must be taken into account in concept selection and optimisation.

Figure 1.12 is a tree diagram showing the possible orbits about Earth and Sun which could be adopted for an astronomy mission. The mission names of spacecraft flown or due to be flown for the different orbits are shown.

It is clear that the choice of orbit for this class of mission is determined by a large number of factors but there is often an overriding consideration. For example NASA's major observatories—Hubble Space Telescope (HST) and Gamma Ray Observatory (GRO)—had to be in a circular low earth orbit (LEO) in order that they

Table 1.2 Checklist of system requirements [Adapted from Fortescue et al. (2011)]

Mission requirements	**Environmental requirements**
Launch windows	Ground activities
Orbit (Transfer; Operating Nominal and Back-up)	Launch and ascent conditions
Operations mass	Transfer and operating orbit environment (Reentry, Descent)[a]
Launch and early orbit phase	Structural/thermal inputs, loads, ranges
Operational phase	Environmental protection
End-of-life	Cleanliness/Contamination
Lifetime	Electromagnetic compatibility (EMC)
(Retrieval/repair/re-supply)	DC magnetic fields
Autonomy	Radiation
Reliability/availability	Spacecraft charge
Ground segment	Atomic oxygen[a]
	Autonomy
Physical requirements	**PA requirements**
Axes definition	Reliability
Configuration constraints dimensions	Availability
Mass properties	Maintainability
Internal torques	Safety test philosophy
Disturbances	Parts, materials, processes
Power/energy	Cleanliness
	Storage, handling, transport
	Configuration management
	Software
Performance requirements	**AIV programme requirements**
Orbit maintenance	Schedule
Ranging accuracy	Model philosophy
Timing accuracy	Safety test philosophy
Pointing accuracy	Ground segment equipment requirements
Measurement accuracy	Facilities usage
Stability	
Pointing range	
Slew rate	
Data rate	
Data storage capacity	
Onboard processing	
Link budget margins	
Telemetry/telecommands	
Strength/stiffness	
Thermal control	
Reliability	
Cost constraints	

[a]For some missions only

could be launched/serviced by Space Transportation System (STS)/Shuttle and Tracking and Data Relay Satellite System (TDRSS) could be used for data retrieval. As far as the science is concerned these orbits are far from ideal. They suffer from regular eclipse periods and the scope for uninterrupted observation is very limited. Without the constraints of a Shuttle launch, two of ESA's astronomy missions,

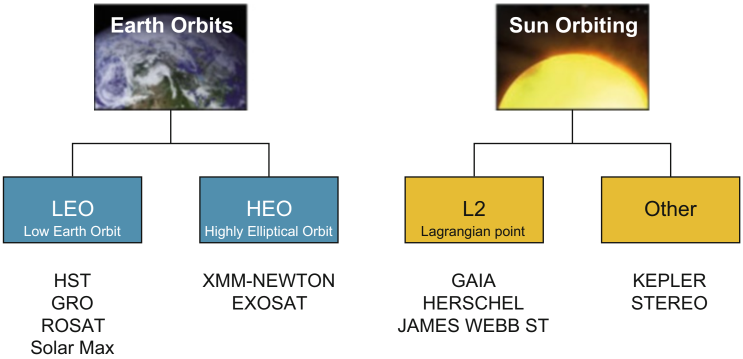

Fig. 1.12 Orbit options for astronomy missions

Integral and the X-ray Multi-Mirror Mission (XMM-Newton), selected highly elliptical orbits (HEO) which can provide long periods of uninterrupted observation away from trapped radiation in the earth's proton and electron belts. More recent missions, such as GAIA, HERSCHEL and First/Planck, have selected orbits around a spot about 1.5 million km from Earth in the direction away from Sun known as the L2 Lagrangian point. In this orbit advantage can be taken of the fact that the benign thermal and radiation environments are ideal for long-distance observations. In addition, by careful choice of the particular orbit around the L2 point, it is possible to have continuous solar power and a continuous communications link. Other spacecraft, such as NASA's Kepler spacecraft, are in orbits around Sun trailing Earth so that a star field can be observed continuously for several years. The importance of the various factors varies with each mission and the current technology.

It is now a common feature of spacecraft that they reuse existing designs of spacecraft equipment. This can offer very significant saving compared to new developments, e.g. the satellite bus used for Venus Express was almost a copy of that used for Mars Express and this in turn was based on the Rosetta bus. The reuse of existing designs and hardware must be treated with caution. Qualification by similarity is a legitimate process but there have been notable failures in the past that have been down to this approach. Examples include the first Ariane V failure because of software inherited form Ariane IV and the loss of the Mars Observer because of the over reliance on hardware qualified for near Earth missions.

1.2.3 Design Drivers and Trade-offs

The purpose of the satellite bus to is to provide the support required for the payload to ensure that it can operate in the required orbit and environment. This makes the payload in most cases, the single most significant driver of the satellite design.

Fig. 1.13 GOCE spacecraft (Credit: ESA)

Power, heating and cooling, structure, power and communication are all provided to ensure that the payload can operate satisfactorily and relay its data back to ground. The propulsion, Attitude and Orbit Control Subsystems (AOCS) and the mission analysis provide the means of getting the payload into the right position to make its measurements. In the case of GOCE, the Gravitational Ocean Composition Explorer, shown in Fig. 1.13, the spacecraft has to fly at a constant, very low altitude of 260 km in order to measure very small changes in the gravity field (Wiese et al. 2012). The effect of the residual atmosphere is very significant and so a main design driver is to minimise air drag forces and torques. As a consequence the satellite body has an octagonal prism shape with two long, fixed solar array wings fitting the launcher fairing dynamic envelope. This requires triple-junction GaAs solar cell technology to generate the maximum power. It also requires an electric propulsion system to ensure the orbit altitude is maintained with the most efficient use of the propellant.

Whilst there may well be a key technological design driver, in a typical space mission there are a number of factors that need to be considered to determine the optimum mission. A trade-off study is an objective comparison with respect to a number of different criteria and is particularly useful if there are a number of possible design solutions. It is common to make use of trade-off tables to "score" the alternative options in early concept studies. Major evaluation criteria for such trade-offs include:

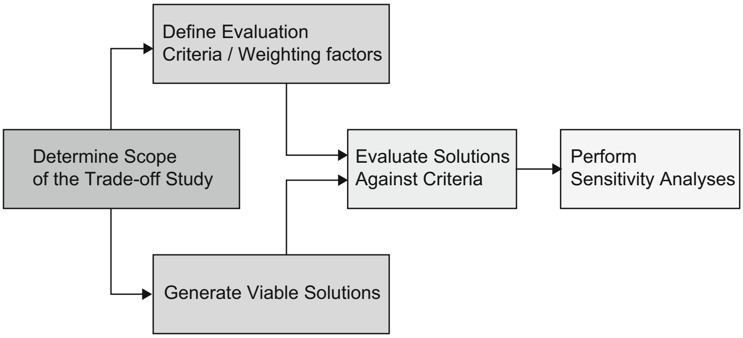

Fig. 1.14 Trade-off process

- Cost, this is generally a dominant factor
- Satisfaction of performance requirements (for example image quality in an astronomy mission)
- Accommodation of physical characteristics, notably mass, size and power which, in turn, impact on cost and feasibility
- Availability of suitable hardware technology and timescales for any predevelopment
- Compatibility with launcher, ground segment and other system elements and the complexity of interfaces
- Flexibility to encompass alternative mission options
- Reliability and availability

Evaluation criteria should be selected that discriminate between the options. If some of these criteria are considered more important than others then a weighted trade-off can be performed. The process is shown in Fig. 1.14 adapted from the National Airspace System (NAS) system engineering manual (NAS 2006). Regardless of whether a trade-off is weighted or not it should only be used as a guide. It is impossible to guarantee that a trade-off is entirely objective and that the evaluation criteria are exhaustive and independent. Cost, for example, is influenced by all the criteria above and its use as an independent parameter is highly questionable. Numeric results are useful but may well give a false sense of accuracy and so should be used carefully.

Whilst some factors can be evaluated numerically many other factors that need to be considered rely on engineering judgement. In addition, quantitative values attributed to factors, can often not be made with sufficient confidence to allow a particular solution to be selected from a number of options. In this case there are often a number of viable solutions possible.

1.2.4 Concurrent Engineering

Concurrent engineering (CE) is a relatively new design tool developed to optimise engineering design cycles. It relies on the principles that all elements of a products lifecycle should be taken into account in the early design phase and that the design activities required should occur at the same time or concurrently. Whilst system engineering has always recognised the value of this approach, the enabling factor for the CE approach has been the rapid development of information technology (IT). Concurrent engineering has enabled design iterations to be performed much quicker and it has enabled the designer to be more closely involved in the design process. ESA's concurrent engineering facility at ESTEC (Netherlands) has achieved the following:

- Studies have been performed in 3–6 weeks rather than 6–9 months
- Cost has been reduced by a factor of 2
- Overall improvement in the quality of the studies by providing consistent and complete mission designs

There are many concurrent facilities around the world and they have become an integral part of the early design phase of a space mission.

1.3 Fundamentals of Space Communications

Felix Huber

1.3.1 Introduction

Radio communication with a spacecraft has to deal with the fact that there are large distances between transmitter and receiver, possible low elevation angles (Fig. 1.15) resulting in a substantial attenuation by the atmosphere and large Doppler shifts due to the orbital velocity of the satellite. Moreover, the ionosphere reflects or absorbs certain frequencies that are thus unusable for space communications.

Two aspects of communications have to be considered:

- Baseband—user aspect
- Carrier—service aspect

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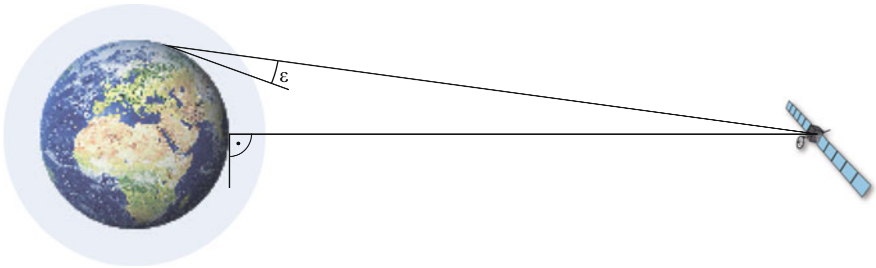

Fig. 1.15 Communication between ground and space at different elevations ε

Both aspects can be handled more or less separately.

1.3.2 Baseband

Signal sources can be discrete values such as switch on–off or pressure and temperature values on the satellite side (“telemetry”) or telecommands in case of a ground station. The path of the signals is shown in Fig. 1.16. Non-digital signals have to be converted into a serial digital signal first in a process called source coding that is described in the next section.

1.3.2.1 Source Coding

Sensors convert physical properties such as pressure or temperature into a normalised electrical voltage for example 5 V. Next, the voltage is sampled at discrete time intervals (sampling) and converted into a binary number by an analog-to-digital converter ADC (discretisation) (Fig. 1.17). If the signal is to be recoverable without losses, it has to be sampled at a speed twice as fast as its bandwidth (not highest frequency!): this is called the Nyquist theorem. The number of steps that the ADC can create (quantisation) has an influence on the rounding errors that occur when the nearest value has to be chosen. This quantisation noise can be made smaller with smaller steps at the cost of a higher data rate that is needed for transmission, thus a trade-off has to be found. The resulting binary numbers are transmitted as a stream of binary pulses, referred to as pulse code modulation (PCM).

If the signal has a bandwidth higher than what the Nyquist theorem allows for, it has to be filtered before being fed to the ADC or frequencies outside the allowed bandwidth will be mapped onto the desired range. This phenomenon is called aliasing and causes a heavy distortion of the signal.

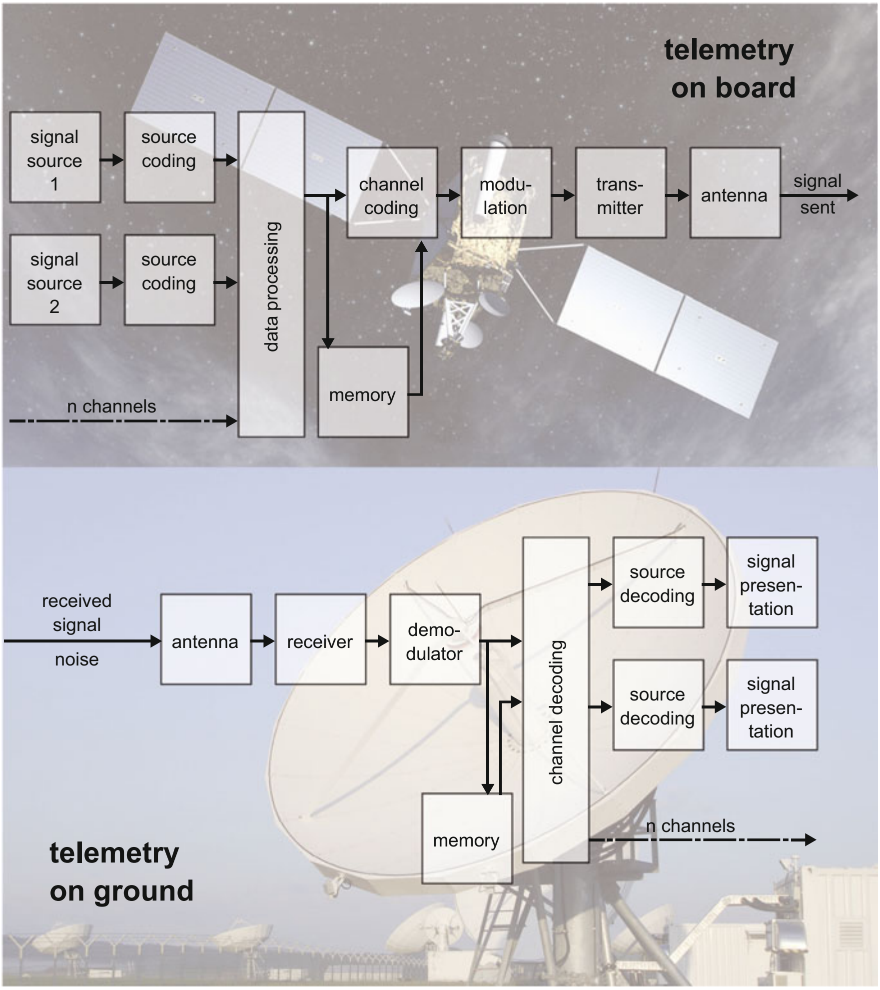

Fig. 1.16 Telemetry signal processing

In a next step, various sources have to be combined (Multiplexing), formatted (range indication, sequential numbering) and perhaps stored for later transmission.

1.3.2.2 Channel Coding

Before the digital data can be sent over the air, precautions have to be taken for errors that can occur during the transmission. In this process of channel coding check sums are added to the data packets (CRC cyclic redundancy check, “inner checksum”) and the resulting serial data stream is run through a convolutional coder

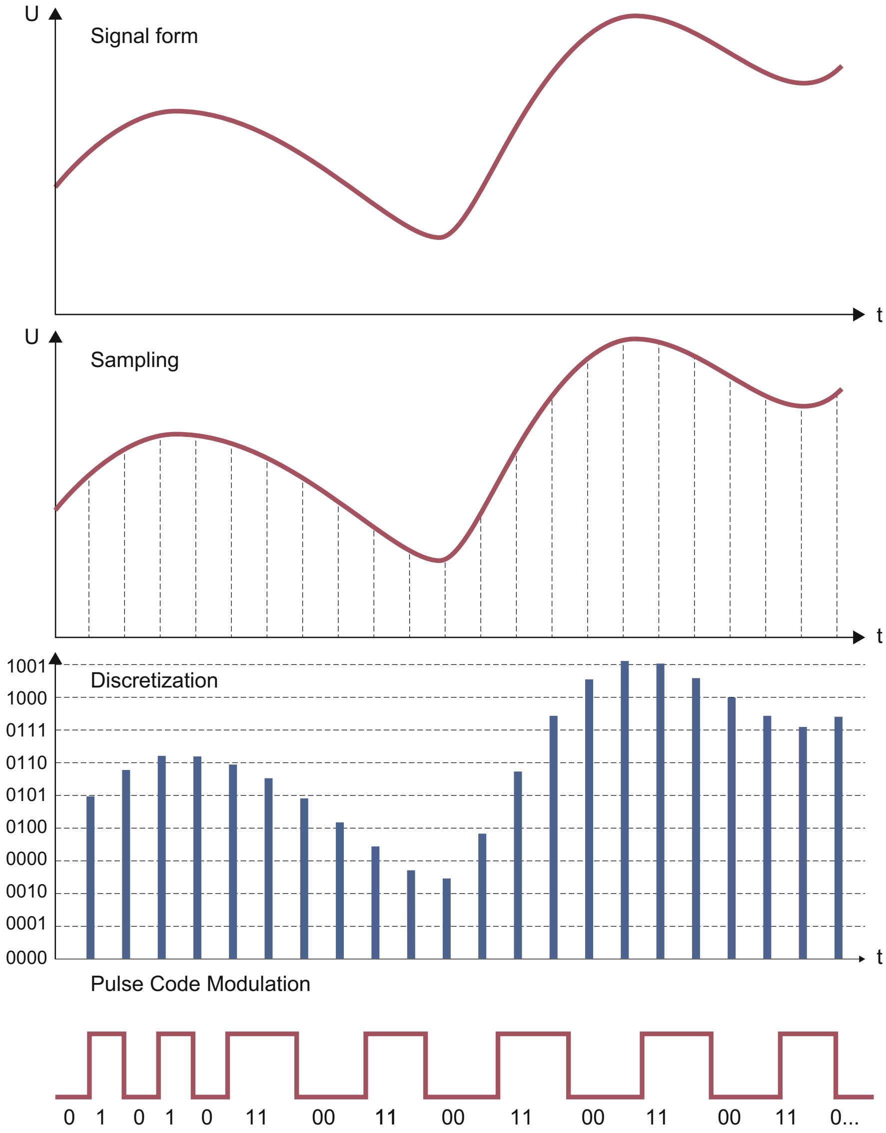

Fig. 1.17 Source coding of a signal

("outer checksum") that adds more check bits in order to recover the distorted bits later after reception. While the CRC only allows for the detection of bit errors, the convolutional coding has enough information to correct bit errors without requesting a retransmission of the data, hence the name "forward error correction" (FEC). This error correction capability is achieved at the cost of a lower net bit rate,

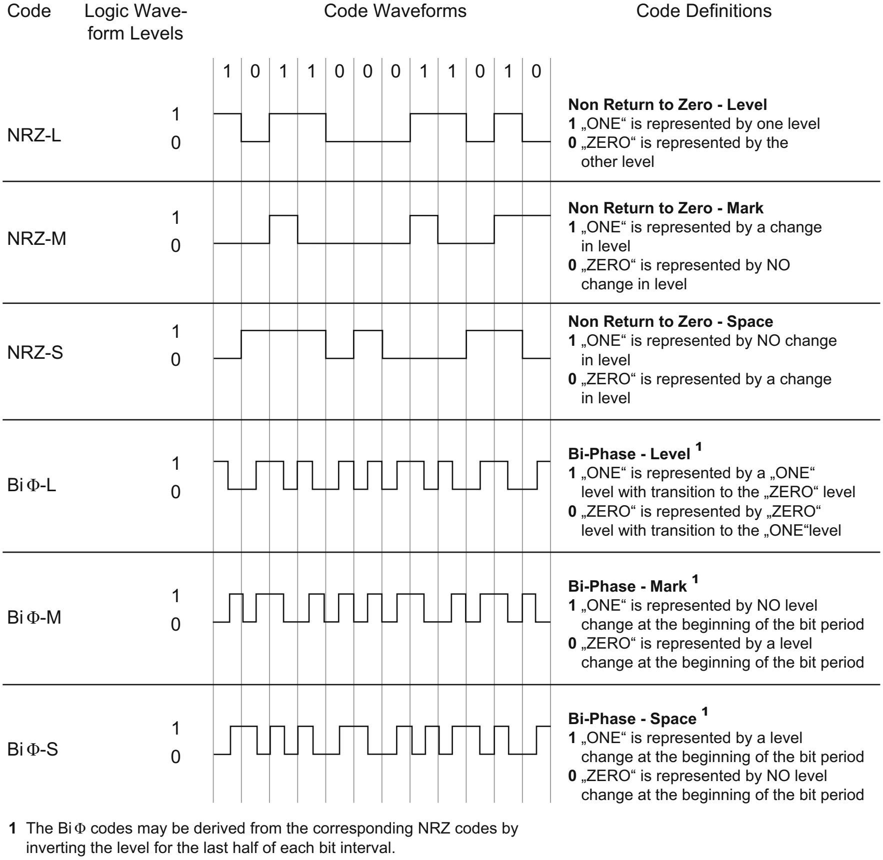

Fig. 1.18 Channel coding according to IRIG 106

referred to rate ½, rate ¾, etc. encoding. It should be noted that the bit error rate could also be lowered by using a smaller bit rate, but the FEC achieves a better lowering of bit errors when compared to the net bit rate; this is called the coding gain.

The final PCM is sent out as a sequence of equally length pulses. However, this sequence of pulses, called non-return to zero (NRZ), can create a DC offset of the average voltage fed to the transmitter that cannot be handled by the system (Fig. 1.18). Therefore, the PCM code has to be converted into a DC-free signal, for example by a bi-phase coding: The signal is multiplied with a square wave carrier of half the bit rate which removes the DC offset, however, at the cost of a higher bandwidth.

It should also be noted that there are two definitions of bi-phase depending on whether a real multiplication is used as an exclusive or logic with the latter being the inverted signal of the first method.

Another critical effect is that if there is an imbalance of zeroes and ones over a certain period of time, a temporary DC offset is generated and the centre frequency will shift causing signal losses due to the bandpass filtering at the receiver. A self-synchronising scrambler is therefore used to smear out periodic patterns that can occur in the data stream. This process, which is also referred to as energy dispersal, creates a uniform spectrum by toggling the PCM bits with a pseudo random number pattern using linear feedback shift registers (LFRS) that have mathematical properties of almost pure randomness. Since the mathematical law of the random numbers is known, the receiver can undo the process of randomising and recover the original bits.

Hence, the transmitted serial signal has no block structure anymore and therefore synchronisation markers have to be added so that the receiver can determine the start of a frame. These synchronisation words are known as Barker codes and have a pattern that has a low cross-correlation since the Barker data pattern could also occur anywhere in the data stream and should not trigger the frame detection. Since the frame length is fixed and known to the receiver, it can check for the regular appearance of the Barker codes and determine the start of a frame.

1.3.2.3 Baseband Shaping

The rectangular pulses occupy a large bandwidth due to their steep edges. If this spectrum is bandwidth limited due to filtering in the signal path, the shape of the pulse gets distorted and spreads over its bit cell time into adjacent cells causing bit errors. This phenomenon is called Inter Symbol Interference (ISI). In order to prevent ISI, the signal would have to be filtered with a bandpass filter with a brickwall characteristic. However, such an ideal filter would have an impulse response that spreads over $\pm$ infinity with a non-causal behaviour and cannot be reached in reality. A more practical approach uses the shape of a raised cosine as the filter transfer function which has also in infinite pulse response, but the corresponding $\sin(x)/x$ shape decays faster at the cost of twice the bandwidth. In practical implementations, a linear mixture of both extremes is used, described by a roll-off factor α, where $\alpha = 0$ corresponds to the rectangular filter transfer function and $\alpha = 1$ to the raised cosine shape with an occupied bandwith of $(1 + \alpha) \cdot$ symbolrate. Typical implementations use α between 0.2 and 0.5.

This filtering can be performed at the analog base band signal where the raised cosine shaping has to be approximated by real circuits or in the numerical domain where the pre-calculated impulse responses are superimposed over several bit cells. Since we have a non-causal filter, the output of the bits has to be delayed using a history shift register in order to make signal causal again. This superimposing of the bits is spread over ± 4 to ± 8 bits, meaning the ideal impulse response is cut off after a certain time, leading to a negligible distortion of the signal.

The resulting signal has zero crossings independently of α always after the bit cell time T, meaning that there is no ISI (see Fig. 1.19). Since we have a symmetric filter, the resulting pulses are also symmetric. A superposition of randomly selected

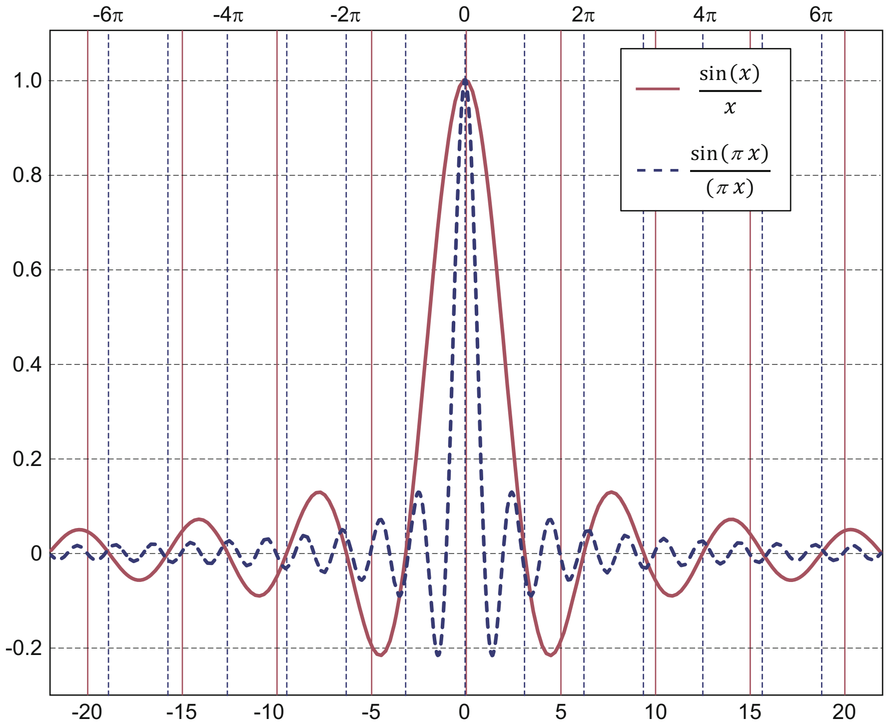

Fig. 1.19 Impulse response of a raised cosine filter

bits leads to a pattern that has the shape of an eye, hence the name eye pattern. It can be used to judge the quality of the received signal: the eye pattern has to be wide open in the centre where the detection of the bits takes place. No zero crossings should occur in the centre; this indicates ISI (see Fig. 1.20).

One more optimisation can be performed in order to maximise the signal-to-noise ratio: The transmitter filter and the receiver filter should have the same conjugate complex transfer function. Since the raised cosine function is symmetric, these transfer functions are the same. However, the eye pattern requires the raised cosine shape at the bit detection in order to avoid ISI; therefore, the filtering is shared between the transmitter and receiver by using the square root of the raised cosine. Such a root-raise-cosine filter system is optimal for both ISI and noise.

1.3.2.4 Modulation

Modulation is the process of applying a (coded) signal onto a higher frequency carrier. A radio frequency signal has the form:

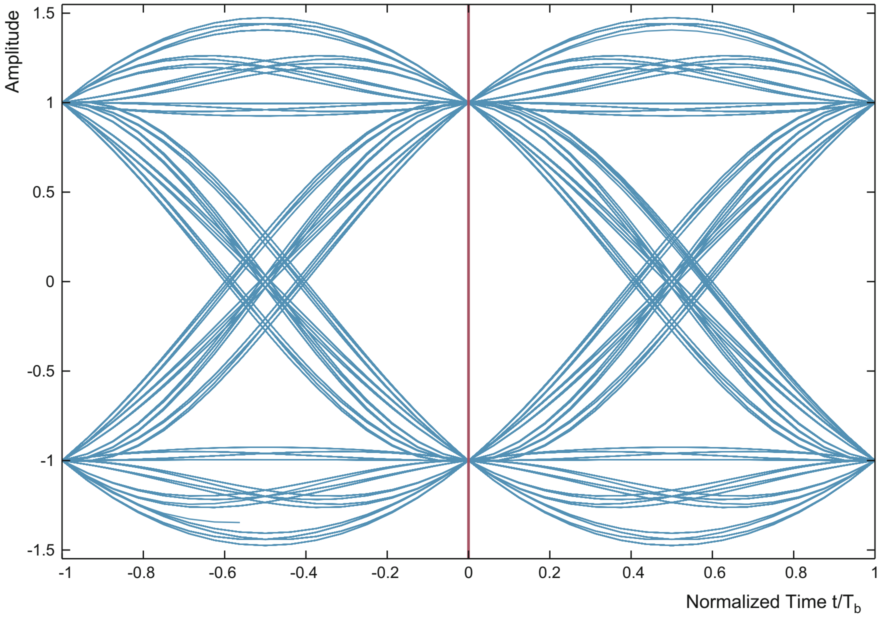

Fig. 1.20 Eye pattern of shaped signal

$$U(t) = A_C \cos(\omega_C t + \varphi_C) \tag{1.2}$$

with the three parameters amplitude A_C, frequency ω_C and phase φ_C that can be influenced by the modulating base band signal. If the base band signal is of analogue type, these changes are named amplitude modulation, frequency modulation and phase modulation, respectively.

The modulation of the carrier converts its single frequency into a band of frequencies that is at least twice as wide as the modulating signal (AM) or even more (wide band FM) (Fig. 1.21).

In the case of a digital PCM signal that has only discrete values, the modulated signal also only takes on discrete values. In this case one speaks of “Keying” instead of modulation (Fig. 1.22) as in the beginning of radio communications the Morse code was generated by pressing the transmission key (beep beep beep, beeeeep, beep beep …). One should note the phase jump after the first bit of FSK. These jumps cause side lobes in the spectra and should be avoided by proper design of the frequency switching circuit.

AM ⇨ Amplitude shift keying ASK
FM ⇨ Frequency shift keying FSK
PM ⇨ Phase shift keying PSK

In the case of PSK, several options are possible, depending on the number of phase values that the signal takes. In the case of only two (0°, 180°) it is called phase

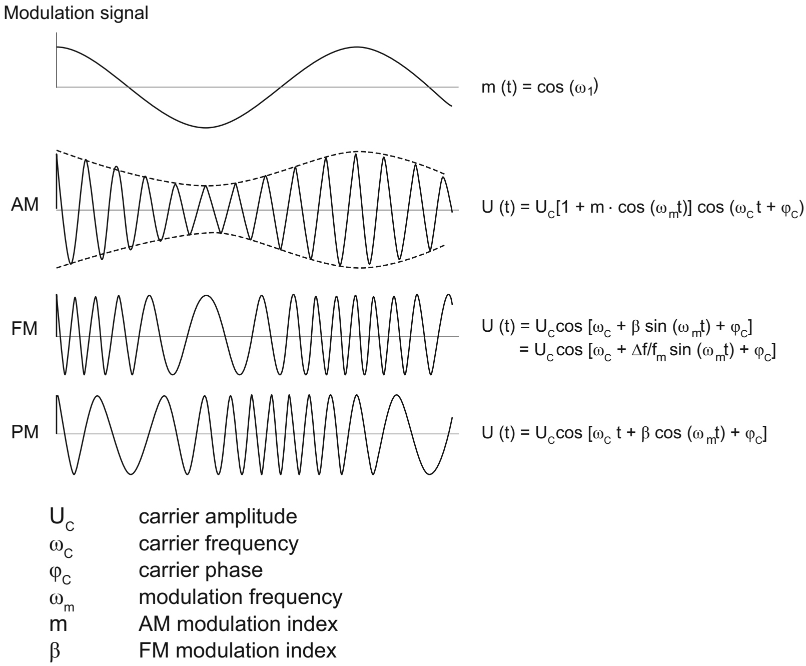

Fig. 1.21 Analog modulation forms

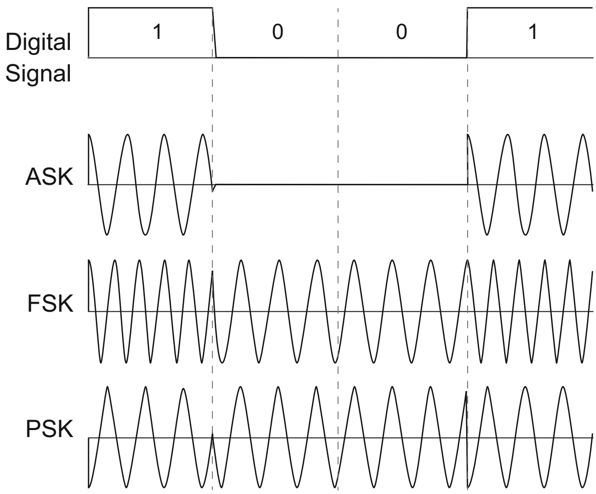

Fig. 1.22 Keying of a digital signal

Table 1.3 Space communications frequencies

Band	Frequency range [MHz]	Application
VHF	150	Voice
P-Band/UHF (lower part)	300–3,000	Military satellites
L-Band/UHF (upper part)	1,215–1,850	GNSS, Satellite telephony
S-Band	2,025–2,400	TT&C
C-Band	3,400–6,725	Future LEOP TT&C
X-Band	7,025–8,500	Payload, Deep space
Ku-Band	10,700–14,500	TV, routine TT&C
Ka-Band	18,000–35,000	Relay
V-Band	37,500–50,200	Intersatellite links

reversal keying or binary PSK (BPSK), with four values it is called quadrature PSK (QPSK), since two carriers (sine and cosine) are used. A combination of AM and PSK is also possible called Quadrature Amplitude Modulation QAM that allows for more bits to be transmitted within a symbol at the expense of a higher noise sensitivity. All *n*-ary PSK modulation schemes suffer from a *n*-ary ambiguity that has to be resolved by the data synchronisation mechanism or by using differential phase encoding.

PSK is used in space communications for its noise immunity and better bits/ energy ratio despite its high effort in the electronics.

1.3.3 Carrier

The space communication frequencies are shown in Table 1.3.

1.3.3.1 Elements of a Space Link

Power Amplifier

Transmits the signal with an average power P_T. Peak power levels can cause a distortion of the signal that has to be accounted for by lowering the input signal (input back-off).

Antenna

Directs the signal into the desired direction by the use of dipoles, horns and reflectors. Since the dimension of the antenna is in the order of the radio signal's wavelength, a Fresnel diffraction occurs that creates a main beam and unwanted side lobes. These side lobes direct energy to unwanted areas in the case of a transmission and collect additional noise in the case of a receiving antenna (Fig. 1.23).

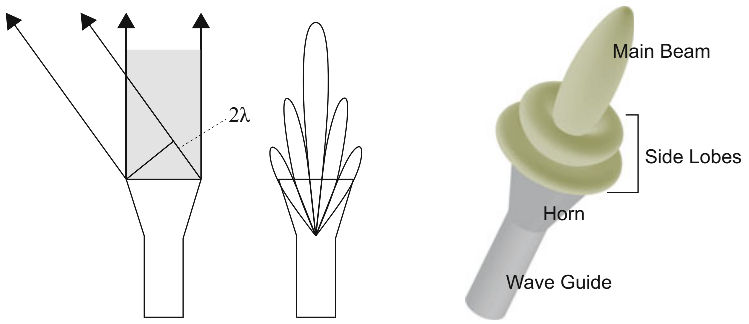

Fig. 1.23 Antenna side lobes

The angle at which the power is at a level of half the maximum value (−3 dB) is called half power beam width and is given approximately by 70° λ/D (in degrees), D: Aperture diameter. The maximum value of the main beam as compared to a theoretical point-like isotropic radiator is called the antenna directivity and its practical value including the efficiency η called "antenna gain" is given by:

$$G = \frac{4\pi A}{\lambda^2} \cdot \eta = \frac{4\pi A_{\text{eff}}}{\lambda^2} \tag{1.3}$$

A: (effective) Aperture area. In the case of a dipole, an effective aperture area can be defined as an area perpendicular to the electrical field lines that still has an influence on the field by "capturing" the filed lines onto its surface. The gain can be seen as the solid angle into which the antenna concentrates the signal compared to a full solid angle.

The product of transmitter power PT and antenna gain G is called EIRP (equivalent isotropic radiated power) and is the power that an isotropic transmitter would have to transmit in order to create the same power flux density at the receiver.

In the case of a receiving antenna, the effective "capture" area for the incoming signal can be calculated from the above equation if the antenna gain is known.

Noise

All warm bodies transmit thermal electromagnetic noise according to Planck's equation. These sources can be Sun, Moon and Earth's ground, atmosphere and clouds, but also galactic sources. In the case of the radio frequency bands, the spectral density of this noise is constant and given by

$$N_0 = kT_{\text{S}} \tag{1.4}$$

where k is Boltzmann's constant and T_S is the system noise temperature as the sum

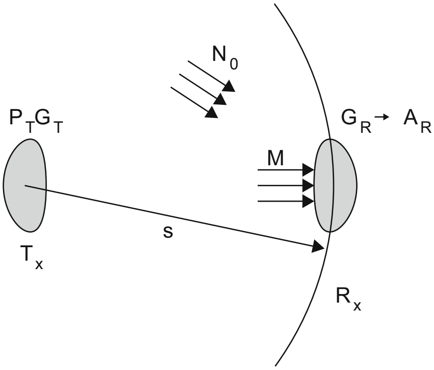

Fig. 1.24 Link geometry

of all natural noises as seen by the antenna and additional artificial signals such as other transmitters or devices (human made noise). Since this noise is purely stochastic, it cannot be removed from the received signal and this limits the sensitivity of the system.

Receiver

The receiver itself creates additional noise in its amplifiers which further decrease the system sensitivity. In a properly designed system, only the first (low-noise) amplifier contributes substantially to the system noise.

1.3.3.2 Link Budget Equation

The performance of the radio link is given by the ratio of the wanted signal (Carrier) and unwanted signals (noise) that has to have a certain value in order to recover the data bits error free (Fig. 1.24).

The transmitter of power P_T and antenna gain G_T is assumed to be in the centre of a sphere of radius s. P_T is the isotropic radiated power and thus uniformly illuminates the sphere's surface $4\pi s^2$. If the transmit antenna has a gain G_T, the power flux density at the receiver is therefore:

$$M = \frac{P_T \cdot G_T}{4\pi s^2} = \frac{\mathrm{EIRP}}{4\pi s^2}. \tag{1.5}$$

If the receiving antenna has a gain G_R, its effective aperture area is

$$A_{\mathrm{R}} = \frac{\lambda^2 G_{\mathrm{R}}}{4\pi} \tag{1.6}$$

and it “captures” at total carrier power C of $M \cdot A_{\mathrm{R}}$.

$$C = M \cdot A_{\mathrm{R}} = \frac{P_{\mathrm{T}} \cdot G_{\mathrm{T}}}{4\pi s^2} \cdot \frac{\lambda^2 G_{\mathrm{R}}}{4\pi}. \tag{1.7}$$

Reordering and adding an additional attenuation L_{A} due to rain, snow, etc. leads to:

$$C = P_{\mathrm{T}} G_{\mathrm{T}} \left(\frac{\lambda}{4\pi s}\right)^2 L_{\mathrm{A}} G_{\mathrm{R}} = P_{\mathrm{T}} G_{\mathrm{T}} L_{\mathrm{s}} L_{\mathrm{A}} G_{\mathrm{R}} \tag{1.8}$$

where L_{S} is called free space loss even though no energy is lost but only diluted over a growing sphere surface area. The reordering of the terms was only done in order to match their position to the transmission path: transmitter, space and receiver.

If the bit rate of the signal is R, the time per bit is $1/R$ and the received energy per bit is C/R. Thus, we finally have the sought ratio of bit energy versus noise power density:

$$\frac{E_{\mathrm{b}}}{N_0} = \frac{P_{\mathrm{T}} G_{\mathrm{T}} L_{\mathrm{s}} L_{\mathrm{A}} G_{\mathrm{R}}}{k T_{\mathrm{s}} R}. \tag{1.9}$$

This Link Budget Equation is usually given in a logarithmic scale using the pseudo unit deci-Bel dB:

$$\frac{E_{\mathrm{b}}}{N_0} = \mathrm{EIRP} + L_{\mathrm{S}} + L_{\mathrm{A}} + \frac{G_{\mathrm{R}}}{T_{\mathrm{S}}} + 228.6 - 10\log R \tag{1.10}$$

where 228.6 is the logarithm of Boltzmann’s constant and $G_{\mathrm{R}}/T_{\mathrm{S}}$, the so-called figure of merit, describes the quality of the receiving system. It describes the radio link on an overall power level but does not take into account the type of modulation and nature of the noise coming from other possible transmitters.

Depending on the modulation and coding used, the required E_{b}/N_0 varies from 1–2 dB for turbo coded PSK to 8–10 dB for uncoded FSK. An additional 3 dB are needed in the case of a non-coherent demodulation.

In the case of orthogonal signals, other transmitters would not even affect the bit detection even though they bring noise power into the receiver as long as they don’t saturate the amplifiers. This is a permanent source of faulty system design.

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Chapter 2
Mission Operations

2.1 Mission Operations Preparation

Andreas Ohndorf

The success of a space mission depends not only on a properly designed and built space segment and the successful launch via a launch segment. It also depends on the ground segment and successful mission operations, carried out by a team of experts using the infrastructure and processes of the mission's ground segment. Its organization and design as well as the assembly, integration, test, and verification (AITV) are therefore equally important as the respective activities of the space and launch segment. A ground segment thereby comprises a ground system, i.e., infrastructure, hardware, software, and processes, and a team that conducts the necessary operations on the space segment.

This subchapter describes the tasks and activities that are necessary for the preparation of mission operations, i.e., what must be done by whom and in which order to enable mission operations of a particular space mission. It gives the questions that arise automatically when analyzing the requirements of a mission and how these questions are answered with a design based on available resources and respecting project-specific constraints.

It is organized as follows:

- With various examples of past space missions, the reader is first introduced to mission operations preparation in general and why this phase is of eminent importance.

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- The input to the entire phase of preparation or the driving factors that influence the design of the ground segment of a particular mission are explained in Sect. 2.1.2.
- Team organization, i.e., who must do what and when, is described in Sect. 2.1.3.
- Section 2.1.4 describes data, products, and tools required for effective mission preparation.
- The particular activities, tasks, and deliveries are explained in Sect. 2.1.5.
- The final Sect. 2.1.6 treats the proven concept of reviews and, in particular, the reviews that are carried out during the preparation phase.

A general remark shall be given here. The following subsections are to be understood as suggestions on how mission operations preparation can be achieved or how it was conducted successfully in the past. They are by no means a "must do," as each space project may exhibit its very specific demands and boundary conditions which can lead to very different realization concepts.

2.1.1 Introduction with Examples

If being asked what "mission operations preparation" is in plain words is or what it means, one could give the following definition:

> Mission operations preparation comprises all measures related to management, development, test, integration, validation, organization, training, certification, and documentation of the ground segment of a space project. The result of successful mission operations preparation is a ready-for-launch ground segment.

The duration of that project phase can thereby vary considerably. Table 2.1 gives examples of past or current missions and the durations of the operational and the preparation phases. These examples cover a range of various mission types, e.g., Earth-bound satellites in low earth orbit (LEO), medium earth orbit (MEO), or geostationary earth orbit (GEO); interplanetary missions to the Moon, planets, and other celestial bodies of the solar system; and deep-space science missions for Sun observation or the exploration of the outer solar system.

The selected examples show that the definition of a general rule for the duration of operations preparation is nearly impossible. Several, mutually influencing factors contribute to space missions in general and to ground segment systems in particular. These systems are:

1. At least one mission control center (MCC).
2. A ground station network (GSN), via whose antennas the ground segment communicates with the space segment.
3. A flight operations team (FOT), which plans and executes operations of the space segment within its parameters.

Table 2.1 Preparation and total project duration of selected space projects

Mission	Type/purpose	Orbit	Spacecraft lifetime (years)	Preparation (years)
TerraSAR-X TanDEM-X	EO, science	LEO	5+	4
EnviSat	EO, weather forecast	LEO	10	12
GPS	Navigation	MEO	12+	22
Galileo	Navigation	MEO	12+	20
Eutelsat W24	Communication	GEO	15+	2
Grace	Science	LEO	12	3
Voyager 1	Outer solar system exploration	Deep space	3 (primary mission) 52 (power limit)	7
Apollo	Human exploration (moon)	Moon	14d	8
Cassini Huygens	Interplanetary exploration	Saturnian system	20	10
Ulysses	Deep-space sun observation	Deep space	17	5 (excluding delays[a])

[a]With a project start in 1979, ULYSSES was planned for launch in February 1983 (Wenzel et al. 1992) but was delayed to May 1986 and again, due to the Challenger accident, to October 1990

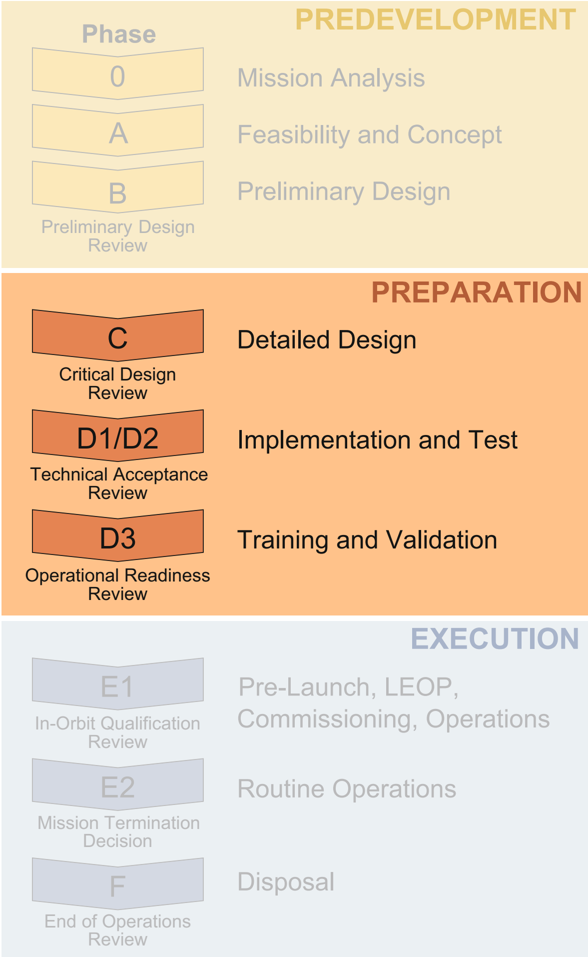
Size, dimension, and complexity of each system depend on a number of mutually affecting parameters and constraints, e.g., mission objectives, technical developments, project phase, schedule, and budget. Striving for an optimal solution therefore inevitably turns into a search for a compromise that is acceptable to the customer. This is the task of mission designers, or, in particular, of those persons responsible for the design of ground segments.

In general, the same (or at least similar) activities and tasks have to be done during every space project, although very different mission-specific requirements need to be met. The European Cooperation for Space Standardization (ECSS) issued a set of management and technical standard documents, which shall harmonize management of European space projects. According to this ECSS phase model (ECSS-E-ST-70C), mission operations preparation is covered in the project phases C “Critical Design” and D “Preparation,” as shown in Fig. 2.1. They are located between preliminary design and mission execution (see also Sect. 2.2).

The result of successful mission preparation is an integrated, validated, and ready-for-launch ground segment.

A generic example of a ground segment, its subsystems, and the data flows between them is shown in Fig. 2.2. It comprises a GSN of three ground stations and a MCC. The main systems of the MCC are the Ground Data System (GDS), the Flight Dynamics System (FDS), and the Flight Operations System (FOS).

Fig. 2.1 ECSS phase model



2.1.2 *Driving Factors*

2.1.2.1 Requirements

The specific requirements of a space mission determine the technical design of the respective ground segment. Their formulation should follow the rules of requirements engineering and in particular those set by ECSS (ECSS-E-ST-10C and ECSS-E-ST-10-06C). These are:

- Performance: Requirements shall be described in quantifiable terms.
- Justification: Each technical requirement should be justified together with the responsible entity.
- Configuration Management: Each technical requirement shall be under configuration control.

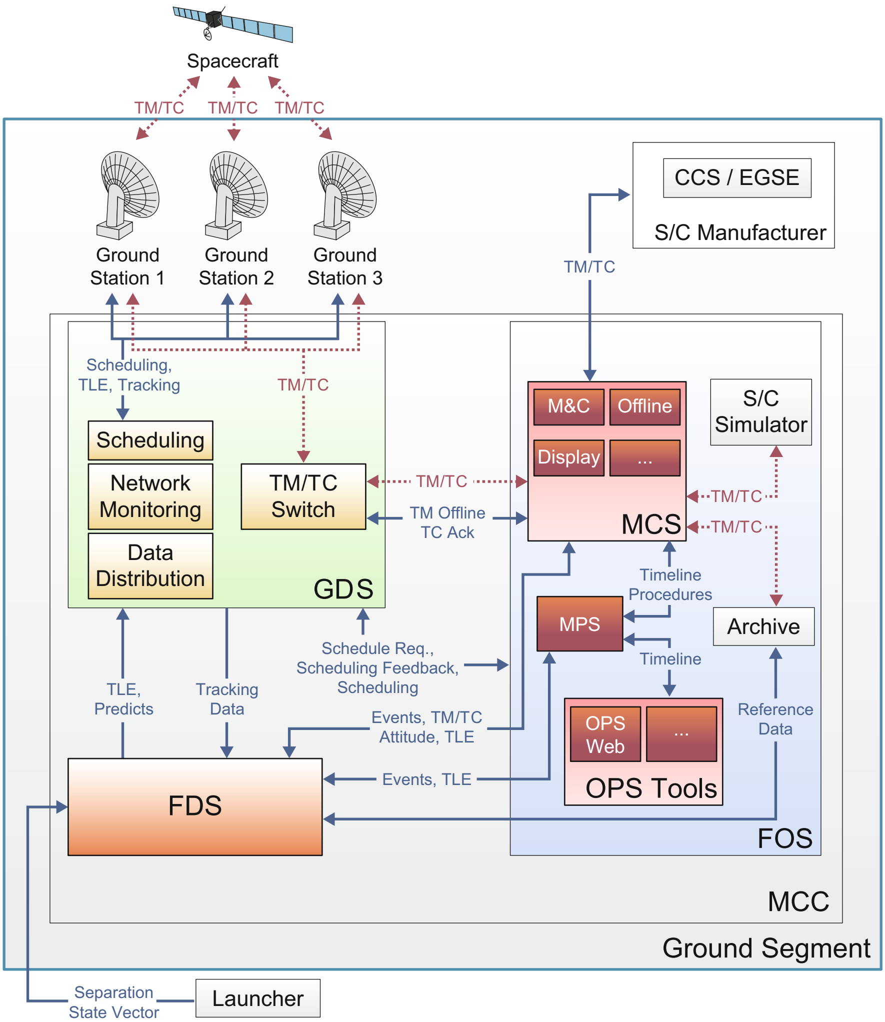


Fig. 2.2 Generic ground segment example

- Traceability: Each technical requirement shall be backwards- and forward-traceable.
- Ambiguity/Uniqueness: Technical requirements shall be unambiguous and unique.
- Identifiability: Each technical requirement shall be identified in relation to the relevant function, product, or system. The identifier shall be unique and reflect the type and life profile situation.
- Singularity: Each technical requirement shall be separately stated, i.e., not a combination of requirements.
- Completeness: Technical requirements shall be self-contained.

- Verification: Technical requirements shall be verifiable using one or more approved verification methods.
- Tolerance: The tolerance shall be specified for each parameter or variable.

These requirements must be analyzed by the engineering and operations responsible (see Sect. 2.1.3) and answered with a detailed design. Ideally, this concept answers each requirement in the best possible way, and its realization takes place within the set project schedule and cost estimate. In reality, however, compromises must be found as otherwise cost and schedule overruns are likely.

Any unanswered or partially answered requirement, i.e., a requirement that is not fully met by a corresponding design, or a requirement whose addressing requires an unjustified high amount of funding, is discussed with the customer. On acceptance by the customer, a temporary deviation or noncompliance to a requirement is documented through a so-called Waiver and thus officially authorized by the customer. A Waiver is, however, not an instrument to document persistent noncompliances, and a requirement alteration should be considered instead.

2.1.2.2 Cost/Funding

Cost-effective design is always required as most projects do not have the funding to develop solutions specifically for a single mission. Nevertheless, the requirements must be satisfied and if several options for realization exist, the one giving the best compromise between risk, schedule, and cost is likely to be chosen.

2.1.2.3 Technology/Complexity

Technical complexity influences cost, schedule, and also the general risk of a space project; this is true for the space segment but also for the ground segment. The necessary degree of complexity, however, depends on a number of factors and drivers, with the most important being the satisfaction of the requirements. This is often achievable through different concepts of various levels of technical maturity expressed with nine, so-called Technology Readiness Levels (TRL). The corresponding TRL definitions according to ECSS are:

- TRL 1: Basic principles observed and reported.
- TRL 2: Technology concept and/or application formulated.
- TRL 3: Analytical and experimental critical function and/or characteristic proof-of-concept performed.
- TRL 4: Component and/or breadboard validated in the relevant environment.
- TRL 6: System/subsystem model or prototype demonstrated in the relevant environment (ground or space).
- TRL 7: System prototype demonstrated in a space environment.
- TRL 8: Actual system completed and flight-qualified through test and demonstrated (ground or flight).
- TRL 9: Actual system "flight-proven" through successful mission operations.

For each subsystem of the ground segment and for each component, the design must be evaluated for alternatives. These alternatives must then be assessed for their effect on schedule, cost, and risk before a decision on the employed technology can be taken.

This shall be explained with the example of a Mission Planning Subsystem (MPS) of an EO satellite.

Let's assume this EO mission requires a defined duration from reception of an image order until delivery of the processed image to the ordering customer of less than a defined number of hours or days. Between these two points in time, several activities are to be carried out. These are the planning for the next available contact for the upload of the image acquisition commands, the prediction of the next opportunity for taking an image of the desired spot on the earth surface, the next downlink opportunity, and the processing of the transferred raw image data. If the required maximum duration between order reception and delivery of the resulting picture is 7 days, for example, the necessary activities might be carried out manually or semi-automatically. However, the shorter the required time frame becomes, the more likely the decision for a fully automatic planning systems becomes. The development of an automatically operating MPS is a complex project in itself and requires significant funding and time. It should consequently be developed as a generic system capable of serving multiple missions. For later missions it can thus be handy to use an existing MPS although it has more features than required. The benefit of reusing an existing system is that, due to its higher TRL compared to a new development, test and validation activities are less.

2.1.2.4 Schedule

The project schedule affects the options for the design, implementation, test, and validation of a ground segment in multiple ways. First, if the schedule is tight, i.e., there is not enough time for the development and test of project-specific solutions, proven technologies of higher TRL must be used. This means to reuse existing software, for example, and to accept possible drawbacks from design concepts that have been tailored to a different mission.

Second, the project schedule also influences the validation and training concept. As there is hardly ever sufficient time for training for each possible and foreseeable situation or emergency, the FOT must concentrate on the most severe ones during training phase. Another influencing factor results from due dates of deliveries by the customer. An essential one is the delivery of the spacecraft simulator. It should be as early as possible, which naturally collides with the fact that the spacecraft design is often not finished. For a series of satellites of the same type, e.g., for a constellation of navigation system satellites, this does hold true only for the first one. However, without having a satellite software simulator of adequate fidelity on time, training of the FOT and validation activities are hampered. Consequently, the delivery of this important item should preferably be fixed contractually.

2.1.2.5 Experience

The short version of this factor is "Whatever it is, it requires less effort if you do it for a second time." A company or space center with decades of mission operations experience can approach future space missions of a similar kind easier than a new competitor. The effort is less because many concepts, processes, and tools do already exist in flight-proven configurations. This depends, however, strongly on the nature of the mission; it cannot be generalized because the requirements of space missions can be of considerable difference. Human-spaceflight missions, for example, pose very challenging requirements in terms of safety, and the cost-effective maintenance of a constellation of 20–30 satellites of a global satellite navigation system may have completely opposite ones. Interplanetary deep space missions are different than Earth-bound satellite projects. A control center specialized on Earth observation is therefore not the first choice for a science mission to one of the moons of Jupiter.

Note that experience may contradict the customer requirements considerably. Early feedback of recommendations to the customer is an important task of the control center as it allows looking for different solutions and the reduction of cost and risk.

Besides the general experience also the time since last mission of the particular type plays a role because experience and proficiency reduces with each year of not operating the particular mission type. Ten years after the last mission of a particular type, one can assume the experience is more or less gone or not applicable anymore and must be reacquired, normally with the same effort as if you prepare for a mission type for the first time.

2.1.2.6 Risk (Also Regarding Launch Delays)

A specific risk analysis must be conducted for each space project, and that analysis must be specific to the respective segments, i.e., also for the ground segment. Risks are primarily w.r.t. cost, schedule, and mission requirements. Naturally, the overall risk for successive mission should be as low as possible; it rarely reduces to zero, though. The minimization thereby results from the reduction of the risk of each system and subsystem to be ready until launch. Preplanned timely buffers and milestones should therefore be part of every project schedule to account for delays in preparation.

Risk reduction is often the reason why aerospace engineering is rather conservative when it comes to employ new technologies. Flight-proven, reliable systems and processes are often given preference before new technologies.

2.1.3 Personnel, Roles, and Responsibilities

The preparation for the operations of a space project requires the organization and assignation of roles and responsibilities. A generic project structure is shown in Fig. 2.3, with the three main branches management, system engineering, and operations engineering. Specific roles are associated to each branch:

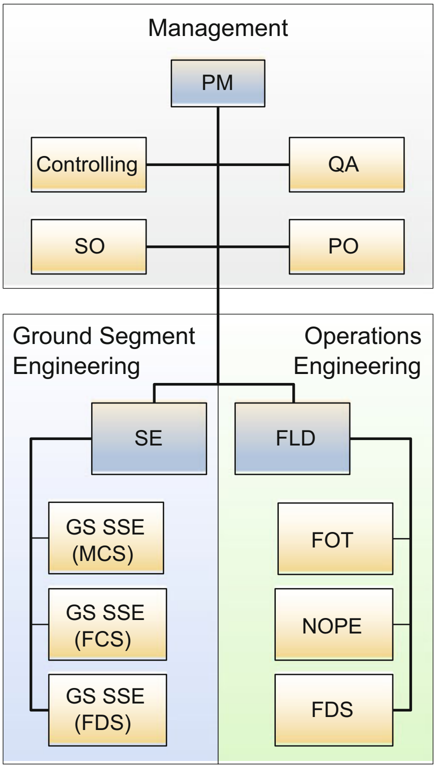


Fig. 2.3 Project organization and roles of a generic ground segment

2.1.3.1 Project Manager

The Project Manager (PM) is responsible for the organization and overall management of the project. He/She is the contact point to the customer and appoints the Flight Director (FD) and the System Engineer (SE). This role is in general given to an experienced engineer who has preferably been either Flight Director or system engineering in an earlier space project. Later, during operations phase, he might take the additional role of the Mission Director (MD).

2.1.3.2 Mission Director

Superior to the Flight Director, the system engineering, and the Lead of the Satellite or Spacecraft Support Team (SST), the MD is overall responsible for the mission execution phase, and, in this function, responsible to the customer once mission operations have started.

2.1.3.3 Flight Director

The FD is responsible for the preparation and the execution of mission operations. He defines size, composition, qualification, and the training concept of the FOT. For this task, he is supported by the simulation officer (SIM). The FD develops the operations concept, formulates technical low-level requirements resulting from that concept, and supervises the FOT during operations. The FD works in close cooperation with the project's System Engineer and reports to the PM.

2.1.3.4 System Engineer

The SE is responsible for ground system engineering and defines the technical concept of a mission's ground segment, which derive from mission requirements. Together and in close coordination with the FD, the SE defines the specification of the ground segment systems and its lower-level components. He supervises the development of new components and adaption of existing ones. He is responsible for the technical implementation of the ground segment, i.e., for integration, test, and validation. In this function, the SE reports to the project manager and supports the FD.

2.1.3.5 Simulation Officer

The SIM is responsible for the planning, organization, execution, and evaluation of the training measures that are required to train the FOT. He reports to the FD and cooperates closely with the SE and the FOT. Common training measures are class room lessons, simulations, and rehearsals. The input of a SIM's work is the required training and skill level, which needs to be defined by the FD. Organizational boundary conditions, such as availability of infrastructure, required data, tools, and specialists, are input for the SIM's planning as well. After an executed training measure and together with the FD, the SIM compiles an evaluation report in which he expresses the success or failure of the particular training measure as well as eventually necessary repetitions or follow-on measures.

2.1.3.6 Quality Assurance Engineer

The Quality Assurance (QA) engineer is responsible for assuring compliance of the project to internal and external quality standards, i.e., he has to monitor the project from a quality and product control point of view and support the PM, SE, and FD during the entire lifetime of the project. External standards are ISO or ECSS standards, such as ISO 9001 "Quality management," ISO 27000 "Information Security," or ECSS-Q-ST-10 "Product Assurance Management."

2.1.3.7 Subsystem Engineer

A Subsystem Engineer (SSE) is responsible for the operations of a specific satellite subsystem, e.g., data handling subsystem. Sometimes, operations of multiple subsystems are combined; the attitude and orbit control subsystem (AOCS) or the power and thermal control subsystem (PTS) are common examples. A SSE must learn the functionality of the respective subsystem, know the telemetry for monitoring that subsystem, and train to apply the subsystem-specific procedures to control its functions, depending on the current situation and intention. They also work with the Mission Information Database to validate and optimize the performance of the MCS.

Additionally to the SSEs of the FOT, the AIT activities before launch are carried out by ground system SSEs. These are specialists for certain components of the ground system, e.g., networks, infrastructure, communication, server integration and configuration, security, software, ground stations, etc. The primary affected subsystems are the ground data subsystem (GDS), the mission control subsystem (MCS), and the mission data subsystem (MDS). The respective SSEs of these subsystems support the SE and the FD from project phase B until launch.

2.1.3.8 Project Office/Project Officer

A Project Office/Project Officer (PO) might become necessary for bigger projects, as the workload resulting from organization becomes too much to be handled by the PM alone. The PO covers documentation management and team organization and provides general support to the PM, e.g., for the organization of reviews.

2.1.3.9 Controlling

A controller supports the PM on contractual and financial aspects of the project during the entire project lifetime. He provides reports and overviews over the project budget in regular intervals and on demand by the PM.

2.1.3.10 Configuration Manager

The Configurations Manager (CM) develops a project-specific configuration management plan during phase C and supervises the implementation of this plan during subsequent phases.

2.1.3.11 Security Officer

The Security Officer (SO) supports the PM in all security related matters, e.g., access control concepts, encryption, clearances, or classification of documents. This can be of importance for military satellite projects whereas for scientific satellite projects this is practically irrelevant. Whether a project needs a dedicated SO or not is the decision of the PM.

The decision on what role should be assigned to which team member needs to be taken by the PM and is specific for each project. For smaller projects, for example, the PM might not need a dedicated PO for project organization and document management. Furthermore, it makes sense to combine the roles of FD and SE for smaller projects; the corresponding workload of larger projects should, however, be shared between two persons.

2.1.4 Required Data, Products, and Tools

Testing the ground segment with its subsystems and components prior launch requires mission-specific tools and data. The validation of developed flight procedures, for example, is strongly impeded without a software simulator that emulates the spacecraft behavior and its foreseen environment, i.e., the conditions in space. The following tools or deliverables are therefore required for mission preparation.

2.1.4.1 Test Data and Data Generators

Specific sets of telemetry data are required for testing of the Monitoring & Control (M&C) system, including the Mission Information Base (MIB), the processing chains (main and backup), and a potential archiving process. Test data becomes of eminent importance if no satellite simulator is provided. Test data may also be needed for other interfaces like file data deliveries.

2.1.4.2 Spacecraft Simulator

A spacecraft simulator is an essential component for mission preparation but is not always provided by the manufacturer of the spacecraft due to cost or schedule constraints, or both. If this is the case, and no simulator is available to the ground segment, then the ground systems team should be granted remote access to engineering models or the flight model of the spacecraft. Access to the flight model can, however, provide less test and validation options as the spacecraft is still on ground. A high-fidelity software simulator provides a representative model of the spacecraft, the spacecraft subsystems, and of the physics of the spacecraft's environment. The implementation of a satellite simulator can be purely in software but also in combination with real spacecraft hardware, e.g., an engineering model of the onboard computer. Such simulators are called hybrid simulators.

The early provision of a stable, high-fidelity satellite simulator eases mission preparation, as it allows early checks of the commands and telemetry data. Furthermore, it enables early familiarization of the FOT with the spacecraft in addition to the spacecraft user manual. The creation and validation of flight procedures is a third mandatory activity during mission preparation that profits from early availability of a spacecraft simulator.

2.1.4.3 Mission Information Base

The MIB contains the definition of the commands, the command parameters, the telemetry, and the position of the telemetry data in the downlink data stream. It is therefore an essential input to validation of the M&C software and also for procedure development and validation. In a preliminary version, it should be delivered on time for the validation phase of the ground segment. At the end of this validation phase, the MIB should be final as well.

2.1.4.4 M&C System Software

The spacecraft is monitored via telemetry monitoring and display software and commanded via command software. Together, these software packets combine to an M&C system. It is an essential component of the FOS and is preferably written in a generic manner so that for each mission only adaption is required. This eases validation as only the changes need to be tested extensively. The functionality of the entire M&C system is then validated during validation of the MIB. Simulations and rehearsals are also used to validate the M&C system.

2.1.4.5 Ground and Flight Procedures for Nominal and Contingency Situations

Control of the spacecraft is basically possible through the M&C software and the validated MIB. However, logical work flows, branching, and timing conditions cannot be described with a simple list of commands. A proven concept to mitigate this deficiency is that of using validated procedures. They are developed for ground and for flight processes and increase mission operations considerably as they reduce the risk of operational mistakes. A procedure describes a validated workflow step by step, together with the required initial conditions, the commands to be sent, the expected response of the space segment, timing conditions, and explaining comments. Such procedures are primarily developed for routine operations, e.g., activation or deactivation of an onboard component or subsystem. Nevertheless, it is important to cover also potential contingency situations with respective procedures. The drop into safe mode is a prominent example of a contingency situation, and the respective procedure should describe the analysis and recovery actions to bring the spacecraft back into normal operations mode.

2.1.4.6 Operations Support Tools

A spacecraft must be monitored and operated. The amount of attention from ground, however, depends on the particular mission. There are space projects that are operated around the clock, e.g., human spaceflight missions, and there are satellites with a single ground station contact per day. Operations of each mission therefore need a customized set of operational tools to make operations as robust as possible. Such tools are anomaly tracking tools, Sequence of Events, telephone lists, shift plans, minutes, recommendation handling tools, links to documentation, procedure lists, etc. The tools should be organized such that they are easily accessible from each control room position, like a mission-specific Web page, for example.

2.1.4.7 Project Documentation, E.g., Spacecraft User Manual, Ground Segment Design Description, Operational Documentation

For safe and robust mission operations, the FOT needs to know the functionality of the space segment and must be trained for situations that are typical or likely to occur. This is facilitated through training and technical documentation, which must be available on time, i.e., before training and operational validation phase. The documentation must be accessible during the entire operational lifetime of the space segment.

2.1.5 Activities, Tasks, and Schedule

A number of tasks must be accomplished during preparation for mission operations. Early in the preparations phase, they are primarily of technical nature, e.g., the integration and test of the ground segment systems and components like software, computers, and networks. Later, when technical work is finished, the composition, training, and certification of the FOT becomes the dominating activity. The following description gives a top-level overview of the tasks of each subphase, the required prerequisites to accomplish these tasks, and the expected output of each subphase.

The first of these phases is project phase C, called “Critical Design Phase,” and the finalization of the ground segment design is its content. If technical developments need to begin before phase D, then they will also start already in that phase (long-lead items). Key roles during this phase are those of PM, FD, SE, and QA.

They coordinate design and documentation activities, and, in this role, they are supported by SSEs and specialists, e.g., flight dynamics experts, network specialists, etc. They all use the design documentation from phase B and elaborate the developed concepts to final fidelity. The interfaces are defined completely and respective test approaches and plans are written. The result of this phase is a detailed design description, including all internal and external interfaces, as well as test plans and schedules.

Project phase C ends with a critical design review (CDR), in which the ground segment provider testifies to the ground segment customer the developed design.

The next phase, phase D, comprises three subphases: the development and procurement of the ground segment systems (D1); assembly, integration, and test of these systems (D2); and the verification and operational validation of the ground segment (D3). During subphase D1, the systems, subsystems, and components of the ground segment are procured or manufactured, including functionality and interface tests. The supervision of these activities is the primary task of the SE, supported by QA. During subphase D2, the ground segment is assembled and integrated. The control room is integrated and potentially required infrastructure alterations are implemented. Networks and automated transfer services are configured and computer hardware is integrated in the operational environment. These activities are again supervised by the SE, supported by QA and the FD.

The following activities are content of subphase D2:

- RF-Compatibility Test (6 months until 12 months prior launch, preferably with flight-model RF components)
- Functional and performance tests of internal and external interfaces
- Functional and performance tests of all ground segment subsystems
- Functional test of the whole ground segment
- Composition of the FOT and first trainings (classroom lessons, handbook study, etc.)
- First flight and ground operations procedures
- Validation of the MIB
- Production of test and validation reports
- Compilation of results in report summary

Subphase D2 ends with the Ground Segment Qualification Review (GSQR) and a Critical Operations Review (COR). Both reviews may also be combined into one review.

The operational validation of the ground segment is primary content of subphase D3, with the main roles involved being FD, SE, QA, and SIM. Intense training of the Flight Operations Procedures (FOP), the finalization of flight and ground procedures, as well as their validation must be achieved. Certification of the FOT's readiness for the upcoming launch is thereby achieved through simulations of increasing complexity. The results of these simulations are documented with corresponding reports for the Customer. These reports and eventual report summaries are subject to review during the Operational Readiness Review (ORR), which is the final review before launch.

Table 2.2 Document deliverables per review milestone

	Domain	GSCDR	GSQR	COR	ORR
Space-to-ground ICD	SS	A			
Space segment operability requirements document	SS	A			
CFI and services requirements document	SS	A			
GS engineering plan	GS	F			
ICDs for external an internal entities	GS	F			
GS design definition file	GS	F			
GS design justification file	GS	F			
GS configuration management plan	GS	F			
GS AIT plan	GS	F			
GS verification plan	GS	F			
GS configuration status report	GS		F		
GS integration and test reports	GS		F		
GS verification reports	GS		F		
Ground systems user and maintenance manuals	GSYS		F		
Logistics support plan	LS		F		
Mission analysis report	OPS	F	F		
Operations engineering plan	OPS	F			
Operational validation plan	OPS			F	
Operations training plan	OPS			F	
Mission operation plan	OPS			F	A
Operational validation reports	OPS			F	F

Essential documents resulting from the preparation phase are listed in Table 2.2, together with the review for which they must be available (see also Sect. 2.1.6). According to ECSS (ECSS-E-ST-10-06C), these documents belong to five levels: space system (SS), ground segment (GS), ground system (GSYS), logistics support (LS), and operations (OPS). The letter "A" designates a document issue for approval by the customer and "F" designates a final document issue, which is approved by the respective supplier and for information to the customer. Note that the documents listed above do not actually have to be separate documents. The relevant information may be embedded in other documents. Furthermore, the actual contract may define only a subset of that list as deliverables.

2.1.6 Review Process

A review is a formal project milestone. Passing this milestone successfully shows that all measures have been taken to complete a certain predefined work package, a project phase, or a subphase. That means the project is stopped to examine, evaluate, and assess the project status and to decide on whether to proceed with the next phase or not. The status is thereby presented with respective documentation.

Several reviews are possible during mission preparation, with not all being necessary or required for every mission. The required project-specific tailoring thereby depends on the particular type of the mission and should be fixed in the contract that covers the phases C and D. This subsection lists the possible reviews during these mission phases, describes their content, common and essential preconditions, and when they take place.

ECSS foresees several major reviews, which are described in the following. These reviews are Ground Segment Critical Design Review (GSCDR), GSQR, COR, and ORR. Additional internal reviews are possible, e.g., test readiness reviews or simulation readiness reviews, and need to be included according to the complexity of the particular project.

Although each of these reviews covers very different topics or elements of the ground segment, they generally follow common principles in terms of schedule and organization. Customization to the needs of a project and also for a particular review is, however, possible. A complete description of the review in terms of "Who, How, and When" should therefore be written. This so-called review procedure is communicated on time to the "review team," which should comprise the project team, i.e., the engineers of the space segment and the ground segment, and the review board. The board should preferably consist of external experts in relevant project fields. These experts can be from other companies, research labs, test facilities, or agencies. The more diverse the knowledge gathered in review board the better.

A review starts with a presentation of the current project status. Venue of that presentation is often at the customer. Each project party gives a brief overview of the current status. This presentation allows the project to describe and explain the specific boundary conditions or constraints under which the current status has been achieved. After the presentation, the so-called review data package is given to the review team for study and assessment. Duration of that phase should be chosen in dependence of the size of the data package. In practice, however, this phase lasts between 2 and 6 weeks.

During the study of the documents of the review data package, all review team members should document any occurred concern via a so-called Review Item Discrepancy (RID). The structure and organization of RIDs are under responsibility of the review board lead and must be described in the review procedure, but for each item at least the following information should be provided:

(1) "Item": Identifies the data package item, for which the remark is valid. It can follow a predefined nomenclature or scheme, but the number of the document, page, section, and/or line number suffice as well.
(2) "Observation": This comment describes what caught the attention of the reviewer. Examples are unclear statements, wrong conclusions, insufficient descriptions, inconsistent analysis, but also typographic errors.
(3) "Concern": The reviewer must express the concern resulting from his observation, e.g., a higher risk for failure of a component due to insufficient testing.
(4) "Recommendation": This is a description of the corrective measures or activities that are necessary for solving the observed problem, e.g., extending a test campaign.

It is the responsibility of the project management and the review responsible to determine a set of RID data items adequate for the particular project. To ease the review process organization, for example, a criterion “criticality,” with possible values “low,” “medium,” and “high” (or “major” and “minor”), helps to group the RIDs and to focus on important ones first.

The review period, i.e., the time during which the review team is allowed to provide RIDs, is limited to approximately 75 % of the entire review period. The RIDs are then provided to the project team for answer. The team then assign a responsible person. This person first decides whether the RID observation is justified or not, i.e., whether the RID is accepted or rejected. The accepted RIDs are then analyzed and answered. The answer is normally an action, the provision of more information, or a correction of existing information. This shall always be reflected with document updates, e.g., update or new issues of project documentation.

The last review stage contains a so-called RID discussion and a closeout, which results in a review report. During the RID discussion, which takes between 2 and 3 days, the project team presents and defends its RID answers to the RID owners. Depending on the answers, the review board then decides on passing or failing of the review and documents its decision with the review board report. A passed review often equals the formal “Go” for the next project phase.

In the following, the specific reviews of the mission preparation phase are explained. Each of these should be organized and carried out according to the given generic description. Project-specific alterations are, however, always allowed but should be agreed between involved project partners. Changes should preferably be not of such extent that the character of a review is changed significantly. In general, and if applicable, the reviews of the ground segment may be conducted together with the reviews of the space segment, e.g., combining the critical design reviews of both segments to a system CDR.

According to ECSS (ECSS-E-ST-10-06C), the reviews during preparation phase are:

Ground Segment Critical Design Review (GSCDR)

Date:	End of project phase C “Critical Design”
Objective:	Customer acceptance of the detailed ground segment design
Precondition:	Design complete, justified, and documented
Content:	Documentation providing description and justification of the ground segment design but also test and training specification as well as interface definitions
Chaired by:	Ground segment customer

Ground Segment Qualification Review (GSQR)

Date:	During phase D, at the end of ground segment AIT and verification (D2)
Objective:	To ensure that the ground segment conforms to the technical requirements and that all conditions are met for proceeding with the operational validation phase (D3)
Precondition:	Ground segment AIT and verification has been finished, i.e., the ground segment is technically ready for usage
Content:	Test documentation, e.g., reports and report summaries of AITV activities on various levels
Chaired by:	Ground segment supplier

Critical Operations Review (COR)

Date:	During phase D, after completion of operational validation
Objective:	To ensure that all mission operations data has been validated and that all documentation is available to start the training of an operational validation phase
Precondition:	Passed GSQR and finished validation of operational data
Content:	Test reports
Chaired by:	Operations customer

Operational Readiness Review (ORR)

Date:	End of phase D, after completion of operational validation, often 3–6 weeks before launch
Objective:	To ensure full readiness of the ground segment for in-orbit operations, and to authorize its utilization for space segment in-orbit operations; to ensure validation of all procedures and readiness of the FOT
Precondition:	FOT training finished; operations procedures validated
Content:	Documentation describing content, course, and results of operations team trainings, simulations, and rehearsals
Chaired by:	Operations customer

The ORR is the final and last review of mission operations preparation before launch. It provides the clearance for the following Launch and Early Orbit Phase (LEOP), during which the ground segment must stand the test with real in-orbit operations. Until launch, however, a few final actions must be done, including regular technical checks of the ground segment elements. Depending on the mission and the control center that conducts the operations, it is also worth considering a so-called system freeze. That means, from a defined point in time, changes to technical systems that affect the mission's ground segment are allowed only under strict configuration control. This is especially recommended in multi-mission environments. The organization of the system freeze needs then to be coordinated with the control center management and eventual other projects. A system freeze can end after LEOP.

If the preparation for operations has been completed successfully, real mission operations presumably will run smoothly and with none or hopefully only a few unexpected or unprepared-for contingencies. However, a ground segment and the FOT can hardly be perfectly prepared and trained for any kind of malfunction or non-nominal behavior of the spacecraft. A high degree of flexibility and the ability of improvisation are therefore essential to maximize chances for mission success.

2.2 Mission Operations Execution

Sabrina Eberle, Thomas Uhlig, Ralf Faller, and Michael Schmidhuber

Although the preparation phase of a space mission can exceed the duration of the actual mission execution phase significantly, the execution can be considered as the most important phase, in which the spacecraft is fulfilling subsequently all of its mission objectives. A dedicated Flight Control Team is in this phase overseeing the operations of the satellite. This chapter will briefly discuss this phase of the satellites lifetime and highlight specific processes and setups.

The chapter is mainly focused on satellite operations; however, some aspects of human spaceflight are also highlighted. For more details of the latter, Sect. 8.1 can also be consulted.

2.2.1 Various Phases During Execution

The execution phase can be broken down into different periods, which can be distinguished by their special operational requirements.

The first part of a mission is called the LEOP. It is followed by the Commissioning Phase, in which the spacecraft as well as the payload on board are prepared for nominal operations. The actual mission goals are then accomplished in the Routine Phase, which is in most cases the longest phase. The lifetime of a satellite ends with the Disposal Phase or End-of-Mission (EOM), which ensures that the satellite is either parked in a dedicated graveyard orbit or is destroyed by a controlled reentry into the earth atmosphere. This phase is also called the De-orbit Phase (Figs. 2.4 and 2.5).

Of course the execution phase is strongly dependent on the mission goal. A scientific LEO mission with experiments on board for only a few months or 1 year has to be prepared in the same way as a LEO mission with an expected lifetime of 5–10 years. Geostationary satellites (especially the commercial communication satellites) often have lifetimes of around 20 years. Interplanetary missions have by comparison very long execution phases because normally it takes the spacecraft a long time to get to its destination in the first place.

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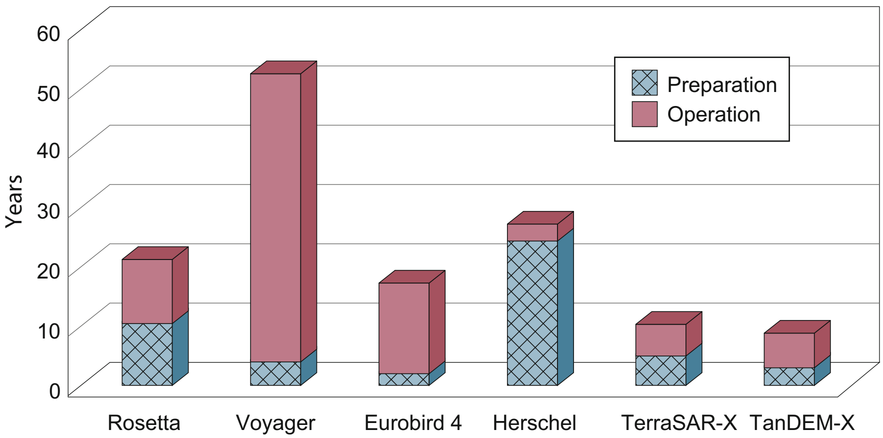

Fig. 2.4 The variable durations of the mission phases using the example of different missions

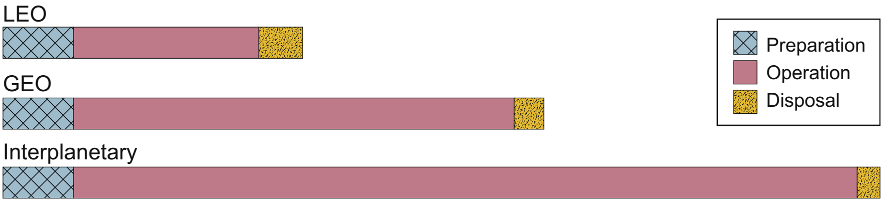

Fig. 2.5 Overview of the differing lengths of mission execution phases on the basis of different mission types. The preparation phase assumed here to be roughly of equal length

2.2.1.1 General Description of the Execution Phase

The abovementioned phases include some common tasks which shall be highlighted in the following section. All spacecrafts have a command and telemetry interface implemented; therefore they can be controlled by telecommands from ground and allow insight into their internal status via the data they send back to the ground stations. This telemetry not only comprises the data from the corresponding payload but also information about important parameters of the satellite's subsystems, like temperatures, currents, status reports from the software or event messages triggered by off-nominal conditions on board. This data – sometimes also referred to as Health and Status or Housekeeping (HK) data—needs to be monitored on ground. Also commanding of the spacecraft is a standard task in the execution phase. Commanding encompasses the operation of the payload as well as the control of the subsystems of the satellite.

Not only the internal performance of the satellite requires surveillance, but also the orbit and the attitude situation of it need to be monitored closely. The active adjustment of orbit and attitude is also required; these maneuvers are described in more detail in Sect. 6.5.

During all execution phases, onboard-maintenance activities like software updates or recalibrations of instruments can take place. Their need is identified during the design phase of the spacecraft and they are usually defined in a maintenance plan, which lists all those activities together with the corresponding time frames, when they need to be conducted.

Satellites are usually designed to be very autonomous. One of the major drivers to involve human Flight Control Teams in the operations is the handling of unexpected situations in the ground or in the space segment. Those events are usually called anomalies or contingencies. Here, humans need to be involved to analyze the sometimes very complex situation—and to put together either troubleshooting plans to identify the root cause of the problem, or to resolve the issue via corrective actions.

Failures can sometimes be prevented, either by preventive actions, which could be part of the regular maintenance activities, or by a detailed short-and long-term analysis of the spacecraft parameters. Tendencies and trends can be observed here, which could result in the decision to take countermeasures to prevent, i.e., further degradation of components or subsystems.

2.2.1.2 Launch and Early Orbit Phase

The LEOP starts after the satellite is launched and released from the carrier. The satellite then has its first time in orbit in which very specific operational requirements take place, which will be discussed in more detail here.

The launch of a vehicle is usually done by a dedicated launch team, which hands over the satellite to the satellite control center after the satellite is released and free-floating. The first major milestone for the controllers is then to establish the first contact with the satellite. Since the ascent phase of the carrier rocket and the release process itself is connected with some uncertainties, the position and orbit of the new satellite is not exactly known; therefore, the first acquisition could be connected with some search activities of the ground antennas. As soon as a radio link is established, the position and orbit can be determined more accurately plus a first checkout of the essential components of the spacecraft follows. In many cases there are also a few very important configuration steps, which need to be performed as soon as possible: The radio link with the satellite is dependent of the satellite's attitude, which determines the orientation of the antennas; therefore, the spacecraft needs to gain attitude control. It is also crucial to ensure power generation capability on board, since the launch phase is usually only supported by the batteries, with limited capacity. This could encompass the deployment of the solar arrays and some reconfigurations of the power distribution system, the proper setup of the battery charge regime, and the switch-on of some vital subsystem components.

With such measures the further survival of the satellite in the harsh space environment is described in more detail in Sect. 1.1. The Space Environment can be ensured. One of the next steps is to bring the satellite to its final destination, be it a dedicated orbit or a dedicated position in the GEO. This might require either several extensive maneuvers with the satellite's propulsion system—or just minor corrections of the orbital parameters. In many cases the maneuvers in this phase of the mission are the main changes of the satellite's orbit over its entire lifetime.

The LEOP phase is probably the most critical phase of the entire mission: After a very demanding ascent in terms of vibrations, acceleration, mechanical stresses, or sound levels, the spacecraft is for the first time exposed to the "real" space environment. The satellite is "unknown" to the operations team and will unveil its characteristics and "features" in this phase—which in most cases leads to several surprises which could lead to a requirement for a redesign of the already prepared operations concept and its corresponding procedures and processes.

Unlike the Routine Phase, which contains usually a series of repeating and well-understood mission activities, a lot of the LEOP activities are singular or even irreversible events. The latter might be due to the fact that the developer decided in the design phase, that only the transition of a given piece of equipment to its nominal ops configuration shall be implemented, but not the transition back into its launch configuration (i.e., only deployment of solar arrays, payload antennas or instrument booms, not the retraction).

The events in the LEOP phase might be time critical, either in the sense that they need to be performed at a very specific time frame (like orbital maneuvers) or that a strict temporal relationship to other activities exist (i.e., a "thermal clock" for equipment, which requires heating after a certain time to prevent degradation or even damage).

Since the LEOP phase contains a lot of transitional states of the onboard configuration, the level of onboard automatisms is usually considerably lower than for the Routine Phase, for which a well-defined configuration of the satellite can be assumed. This also limits the "self-healing" Fault Detection, Isolation, and Recovery (FDIR) possibilities and thus makes the satellite more vulnerable.

All of these abovementioned reasons lead to the requirement to have a good, almost permanent "visibility" of the satellite during LEOP, to have the chance to detect and intervene quickly in case of problems. Therefore multiple ground stations are usually involved to ensure a good coverage, which differs from the Routine Phase, where only one or a low number of ground stations are used for a very limited contact with the spacecraft due to cost reasons.

This setup of multiple ground stations including the coordination of them introduces an additional level of complexity which makes the system more prone to failures. It also requires a high level of redundancy in the ground system to cope with problems in this essential chain link of spacecraft operations.

2.2.1.3 Commissioning Phase

The LEOP phase is followed by the commissioning phase; the transition between them can be blurred. In that phase the satellite is ready to be used; it flies on its designated orbit and its survival is ensured. Now the extensive testing of its payload can be started. This involves checkouts on subsystem level as well as on an integrated level.

In spaceflight it is common to follow a concept of high redundancies. Many subsystems have redundant components—critical ones have even more than one level of redundancy. Elements are called “hot redundant,” if the redundant part is already active and could thus take over the function in a very short time period without interrupting operations. “Cold redundancy” means that the redundant element needs to be activated in a failure case first, which leads to some latency time.

In the commissioning phase also the redundant elements are checked out, to ensure that their performance is sufficient to consider the function as fully redundant and to be able to tune the ops concept accordingly in case of a degraded performance of the redundant component. The test of a hot redundant component is less critical than the test of a cold redundant one, since for the “hot case” the device is already active and some information about its performance is at hand already.

Redundancy testing is important for a future lowering of the operational risks; however, it also constitutes a certain risk in itself: The spacecraft is brought from a good and reliable configuration into a configuration which involves a not-yet-tested component; also the transition process between the nominal and the redundant element can be considered as a more vulnerable phase of the satellite, since it is originally only foreseen to be executed in a contingency case, where the increased risk of switching to another box is justified. For these reasons it is attempted to avoid unnecessary switching processes and to bring them down to an absolute minimum by working on sophisticated checkout sequences.

As already mentioned, checkouts during the commissioning phase are not only done on the subsystems of the satellite platform but also on the payload complement. This subphase is called In-Orbit Testing (IOT). Depending on the payload and its purpose, it might be required to run dedicated configuration procedures or to perform calibration runs with them. The latter might involve additional support or equipment on ground, which is not required during the routine phase. At geostationary communication satellites normally the antenna needs to be positioned to align the antenna beam on the chosen area, the solar panels need to be activated to rotate with the sun, and the payload itself must be started.

In many cases it is also required to prove that the flight hardware meets its specifications or requirements from the design phase. This can have technical or even contractual implications. Therefore there might be test objectives which need to be accomplished during the commissioning phase as well.

All the above tasks which are typically for this phase of the missions require the presence and participation of the corresponding experts, be it from the companies involved in the construction of the components, from the expert teams within the Flight Control Team, or from the side of the payload users, the experimenters, or scientists.

Sometimes a mission control center only conducts the LEOP and Commissioning Phase and transfers the operations to another routine operations center for the comparatively easy routine phase. This transition is called the “handover.” The LEOP control center is still in standby for the first few contacts after the handover and after checking that the routine operations center can operate the spacecraft trouble free, the LEOP control center stops its operations. This scenario is used rather frequently when the satellite manufacturer offers a turn-key delivery to the customer in orbit or when the routine control center does not have the experience or the resources (large control room and access to global ground station network) available to conduct a LEOP.

2.2.1.4 Routine Phase

When the commissioning phase could be successfully completed, the routine phase can be initiated. The operations of the satellite are now usually linked to routine processes, the telemetry is observed and analyzed as already described, and planning as described in Sect. 5.1 governs the day-to-day tasks of the satellite. The payloads are operated to reach the mission objectives, whereas the subsystems are operated to support payload operations and to ensure the well-being of the entire spacecraft.

Planning also encompasses the management of the limited resources the spacecraft has available, like electrical power or also fuel for orbital maneuvers.

The character of the routine phase also allows reducing the manpower utilized during steady-state operations to a minimum; the experts could be assigned to other projects and are only required to be available “on need.” Nevertheless they still monitor their subsystem on a daily or at least weekly basis, dependent on the complexity and flexibility of the subsystem. Especially at the thermal system trend analysis is a very important instrument because the system is rather indolent and it takes the temperatures a while to change, so it is important to keep an eye on the long-term monitoring. At other subsystems like Attitude and Orbit Control (AOCS), the specialists monitor their parameters more often. All SSEs analyze trends, in order to prevent foreseeable contingencies or errors during the whole routine phase.

In addition to the monitoring of the telemetry, there are normally weekly team meetings to discuss special mission topics, occurred events, and upcoming actions. Special but expected events are, e.g., orbit correction maneuvers which are calculated by the flight dynamics team (FDS) or antenna tests at the ground stations. In geostationary satellite missions the orbit correction maneuvers are normally very predictable and follow a certain repeating pattern. For LEO missions mostly not many maneuvers are required and depending on the mission will only be done every few month. Further routine tasks beside the weekly team meetings are the monthly reporting to the customer, the maintenance of the change control, and the constant training of the team members to keep them up to date and always trained even during a long routine operations phase.

Unexpected events are, e.g., a collision avoidance maneuver, a switch to a redundant component on board or a software upload, normally provided by the satellite manufacturer. The daily routine operations, e.g., dumping the telemetry and sending the timetable for the next payload operations, can be taken over by command operators (SPACONS), which are not required to have an in-depth knowledge of the satellite’s subsystems. In case anything unforeseen happens the operator immediately contacts the Flight Director or the relevant SSE.

At the routine operations the ground network, which is required to maintain the contact to the satellite, is reduced significantly. In many cases only one ground antenna can serve this purpose, depending on the orbit parameters, as described below. In that case the visibility of the satellite is reduced to some passes per day only, which are then used to downlink the payload data, to get an insight into the health and status parameters of the satellite and to uplink commands, which might be "time-tagged," meaning that they are not immediately executed, but only at a well-defined time during the following orbit(s). Often, highly automated planning engines on board of the satellite can take over control on board and execute the payload tasks autonomously.

2.2.1.5 End of Mission or Disposal Phase

There are some actions to be completed before the satellite mission can be declared as ended. This is required by national policies or agreements with the customer. All these tasks are summarized in the disposal or decommissioning phase.

It has to be ensured that the satellite is not posing a risk for future spaceflight missions or for anybody living on Earth and—especially for the case of the very crowded GEO—its position in the sky needs to be freed for its possible successors.

The latter can be achieved by two means: In case of LEO satellites the orbital maneuvering system is used to change the spacecraft's trajectory in a way that it enters the earth atmosphere in a controlled manner and is then destroyed by the thermal energy in which the immense kinetic energy of the object is converted to during reentry. In case the satellite is on a GEO, it is brought into a so-called graveyard orbit, a trajectory in which the satellite does not interfere with other satellites for many decades. This orbit is a few hundred kilometers above the GEO. Each mission is committed to leave enough fuel in the tank so the spacecraft can be maneuvered either into the earth atmosphere or the graveyard orbit. This amount of fuel has to be calculated very thoroughly for each maneuver during the routine phase to ensure that not too much fuel is left, which would cost the mission valuable lifetime, and not too little so the final maneuver can be executed completely.

For the de-orbiting itself normally the payload will be switched off and the satellite will be brought in a safe configuration (Skalden 2013). The systems are passivated by shortening the batteries and emptying the tanks to reduce the danger of explosions.

2.2.2 *Staffing of the Flight Control Team*

Each mission has other requirements for the composition of the Flight Control Team—and different control centers follow slightly different philosophies. However, some elements and some considerations have general validity and those will be presented below.

The different functions which are represented in a Flight Control Team are often referred to as “consoles” or “positions”; they are interconnected by modern voice communication systems and use dedicated tool suites to spread information to everybody, to document decisions which have been made, to record the shift events in a dedicated shift diary, to command the spacecraft, and to see its telemetry.

2.2.2.1 Mission Operations Team Lead

A complex, multi-person team requires a clear hierarchy, a coordination function, and a decision-making process, which allows quick reactions to unknown situations. Therefore a team lead function is available in all setups of Flight Control Teams. The nomenclature can differ, typically the terms like “Mission Ops Team Lead,” “Spacecraft Operations Manager,” or “Flight Director” are used. Flight Director (FD) is used in this book.

The person on that position has the full responsibility of all operations conducted by his team, which involves especially all commanding of the spacecraft. Therefore the Flight Director is also the final instance for all decisions and has to approve all commands which are sent to the vehicle.

His authority in real time is usually only limited by the ops documents, which define the operational envelope for the satellite—and under certain emergency circumstances he can also decide to violate those. It is important for the flexibility and adaptability of operations to equip him with extensive power.

Depending on the project setup, the authority of the Flight Director might be limited to real-time processes only. In these cases there needs to be another authority which is not part of the Flight Control Team, but provides the team and the Flight Director in particular with management directives in case need be.

The FOT has the full responsibility for the satellite during operations. During critical operations phases like the LEOP or special tests during commissioning phase, an industry team will assist the FOT. If there are any non-nominal situations which were not described in the handbooks and are not covered with procedures, the industry team may help finding a solution to bring the spacecraft back to the nominal configuration. However, the Flight Director has the overall responsibility of the operations. The SSEs of the industry team only advise their corresponding partners of the FOT. The team lead of the industry team is often called the “Satellite Team Lead” or “STL.” This person will directly communicate with the Flight Director.

Representatives of the customer will have a number of console positions in the control room so they can monitor the operations. They communicate within their own team on a dedicated Voice Communications System loop and of course they cannot send commands. If and how the customer is involved in operations and decisions is dependent on the mission.

2.2.2.2 Subsystem Specialists

All subsystems of a spacecraft are usually reflected as positions in the Flight Control Team. That way it can be ensured that the team has sufficient expertise and the manpower to concentrate on their subsystems, to decide in critical situations, where a deviation from the standard processes is required. The subsystem specialists monitor the data of their corresponding subsystem, analyze it, and ensure that possible anomalies are detected and resolved. The level of resolution is again dependent on the overall concept: The Flight Control Team might be empowered and able to bring the spacecraft back into a fully nominal configuration—or they just conduct a first contingency response, which brings the satellite into a safe mode, and have now enough time for the further analysis of the problem. This will then be discussed and forwarded to engineering support teams, which finally provide the team with advices how to recover the anomaly and resume nominal operations. The industry team normally will assist the Flight Control Team.

The Flight Director as the final decision instance in real time is supported by his specialists in his decisions.

For human spaceflight missions, the crew can be considered in a first approximation as (an) additional subsystem(s); therefore additional positions in the Flight Control Team like “spacecraft communications” or “medical operations” are available here.

2.2.2.3 Command Operator

In many teams the actual commanding activity is performed by a dedicated command position. This ensures a good coordination of all commanding, since it is done centralized. The command operator or SPACON (spacecraft controller) takes instructions from the Flight Director. In that way the flight team is relieved from the technical aspect of the MCS and the communication with the network operator (NOPE) and the ground stations (cf. Sect. 3.2).

In routine phases the presence of the Flight Control Team in the control room can be reduced to only the command operator, who receives pre-generated and pre-approved command tasks from the subsystem specialists and the Flight Director. The operator then prepares the command sequence, uploads it, and checks its successful execution. Should he detect any anomaly, he could then alert the Flight Director or the subsystem specialists which are usually on call for that purpose. His autonomy is usually constrained to well-understood and strictly defined situations.

2.2.2.4 Planner

The scheduling of the often very complex activities and a quick and profound reaction in case of malfunctions to ensure that the mission can continue under the new boundary conditions requires the presence of a planning function in the Flight Control Team. The planning concepts for satellites and human spaceflight operations are described in more detail in Chap. 5. The inputs of the mission planning team, the timeline, will be provided as ready-to-send telecommands and will be part of the daily command stack. Conventional geostationary communication missions usually need no dedicated mission planning. The planning tasks are distributed within the FOT.

2.2.2.5 Flight Dynamics

The maneuvers which have to be performed will be calculated and initiated by the flight dynamics group. These can be normal orbit maintenance maneuvers to keep the satellite in its nominal orbit, or an unscheduled maneuver like a collision avoidance maneuver. These maneuvers have a high priority and all the other planned tasks will be canceled and rescheduled in order to prevent the satellite from a possible crash with another spacecraft or an impact with an uncontrollable component. The planning and development of an orbit maneuver is described in more detail in Sect. 4.1.

Also other orbit related information is provided by the Flight Dynamics Team: Contact Times, S/C sensor usability prediction, maneuver calibration, and the collision risk estimation.

2.2.2.6 Ground Data Systems

To communicate with the spacecraft, the data links from the control center to the ground station and on to the satellite and back have to be established and maintained. This is done by the Ground Data Systems Group. The communications with the ground stations from the control room is executed by a special network communications operator. He will also inform the Flight Director about difficulties or changes concerning the antennas. The communications concept is described in more detail in Sect. 3.2.

2.2.2.7 Assistance Team

Essential support to the Flight Control Team comes from the Engineering Support Team, which is in most cases staffed by experienced engineers of the satellite supplier companies. They have the expert knowledge to analyze problems which lie beyond the knowledge and the expertise of the Flight Control Teams. In critical phases (e.g., LEOP) representatives of this team need to be present in the control room; in routine phases they can be contacted remotely if need be. They can be considered the second—and in many cases also the last “line of defense.” Special attention has to be given to the fact that naturally the industry experts are moving on in their careers and the original knowledge is dissipating over time. In long-term missions the Project Manager and the Flight Director need to secure the necessary knowledge and eventually bring this to the attention of the customer.

2.2.3 Interactions within the Flight Control Team and Flight Procedures

2.2.3.1 Interactions within the Flight Control Team

A typical scenario in a full-fledged Flight Control Team is described here for better illustration of the baseline concept.

During the Acquisition of Signal (AOS) one of the subsystem components on board shows a high spike in the current of one of the power devices. The values are sent down to the control center, where the reading appears on the telemetry displays of the Flight Control Team. The out-of-limit condition might be automatically detected by the ground software and the team could have been alerted via a visual—and under certain circumstances maybe also an audible alarm.

Either the Flight Director now prompts his team—or the responsible subsystem specialist proactively approaches the Flight Director and provides some information about the failure signature he has seen, an ad-hoc analysis about the root cause, and—with a reference to the ops documentation—a recommendation on how to react in that specific case. The Flight Director might involve other affected disciplines or might consult an assistance team, in case it is available and then base his decision for the problem response on the information he gathered. He would then advice the command controller to prepare the corresponding telecommands and send them to the vehicle, in close coordination with the corresponding ground station. The success of the commanding and of the problem resolution approach is then checked by the command operator and the subsystem specialist, respectively. Based on the results, further steps of recovery, some troubleshooting measures, analysis by further experts, or the documentation of the anomaly will follow.

All the interactions between the various positions are usually described by ops documents, which are also the foundation of the work and responsibility sharing within the team.

2.2.3.2 Flight Operations Procedures

Safe and reliable operations of a spacecraft in orbit require sufficient knowledge about how to fly the spacecraft. Detailed information about the spacecraft itself and the ground system used for the operations is provided by handbooks, telemetry and telecommand databases, and other reference lists, but the basis for the operations is built on the so-called FOP.

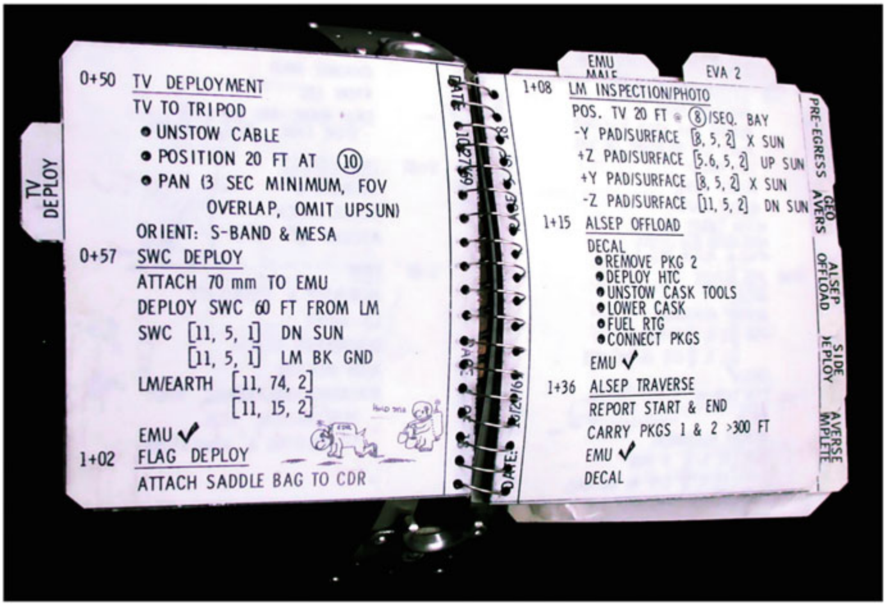

Fig. 2.6 Apollo 12 astronaut cuff checklist, NASA

Definition and Applications

FOP are a prepared, tested, and validated set of work instructions that list in the very detail all activities and checks to be performed for a specific purpose. This includes the exact sequence and timing of the different steps, complemented with comments about the activities and go/no-go criteria for critical events.

Depending on the mission characteristics, e.g., operating a manned or unmanned spacecraft, long (in GEO) or short (LEO) ground station contact times, the kind of monitoring and control system (MCS), level of spacecraft autonomy, or mission budget, different types of FOPs can be used. For manned missions with astronauts in the loop, it may be sufficient to have FOPs in a free text form on paper or screens (see an example of an astronaut checklist on Fig. 2.6). In unmanned missions, most actions are done through telecommands and telemetry. Therefore flight procedures concepts are mostly table-based (see Fig. 2.7). This allows an automated processing of the instructions and thus, safer and easier preparation, and shorter execution times. More sophisticated systems are using script languages, which allow partly- or fully-automated execution of procedures by the ground system with only a minimum of supervisory activities by the operations personnel.

FOPs are typically grouped into procedures for nominal and non-nominal activities and tasks. Nominal procedures might be used for standard and planned situations (e.g., boost maneuver during LEOP), while non-nominal, often called contingency procedures, are prepared for anomalies, error detection, and trouble-shooting (e.g., no telemetry at signal acquisition).

PX1	NP120		Solar Generator Partial Deployment					TPX-Flight Operations Plan Vol 3		Issue 1.3	
STEP	TIME		ACTIVITY				DISPLAY	TM			
n°	ABSOLUTE	RELATIVE	EVENT DESCRIPTION		CODE	DATA	FMT	CODE	DESCRIPTION	RESULT	
1		T0-8min	**Verify Entry Conditions**								
			Check S/C Status								
			VERIFY:								
			AOCS mode				5200	AS100	AOCS mode	**SAM**	O
			Sun vector S/C x				5200	AD451	Sun S/C x	**0 +/- 0.02**	O
			Sun vector S/C y				5200	AD452	Sun S/C y	**0 +/- 0.02**	O
			Sun vector S/C z				5200	AD453	Sun S/C z	**-1 + 0.02**	O
			Main battery power level				6122	PD300	Bat A PWR	**> 50Ah**	O
			Deployment hinge temperature				2200	TP280	Depl H tmp	**> 10 deg**	O
			IF entry Conditions are not correct								
			THEN postpone deployment								
			ELSE proceed with step 2								
2			**Partial Deployment**								
									UPS: make hardcopy of #4210		O
2,1		T0-5min	**Prepare Pyro System for Deployment**								
			SEND:								
			Activate_Pyro_System		11PAD						[]
			Set_Pyro_System		10PAD						[]
			Arm_stage			12					
			VERIFY:								
			Pyro system arming				6122	AS230	Pyro Arming	**ACTIVE**	O
			Arming stage selection				6122	AS235	Arm Stage	**12**	O
2,2			Request Go/NoGo for deployment from all subsystems								O
2,3			Request Go/NoGo for deployment from STL								O
2,4			**Deploy Generator**								
		T0-1min	Set TC system in BD mode						**TC operator: verify TC system in BD-mode**		O
			SEND next 2 commands in sequence:								
		T0			20PAD						[]
			Fire Pyro		21PAX						[]
									TC operator: verify TC system in AD-mode		O

Fig. 2.7 Table-based flight procedure example

Flight Procedures can be "atomic" or "elementary" and contain only a few instructions around one activity (e.g., switch on of S-band transmitter) or they can be encompassing to cover a long complex activity (e.g., in-orbit test of a repeater payload).

FOP Life Cycle

The initial input for the FOP generation is provided by the spacecraft manufacturer, because he has designed the spacecraft and the basic algorithms to operate it. The initial procedures consist of the basic activity flow, the commands to be sent, telemetry checks to be performed, constraints, and basic timing specifications. In the next step, the flight operations specialists complement the procedures by adapting them to the provided Monitoring & Control System and insertion of additional information, like display page references for telemetry checks or ground-based activities. Finally the procedures are validated by running them in simulator sessions or comparable test environments, so the correctness of foreseen commands, TM checks, and timing is confirmed. With the release of the validated FOPs, the utilization phase starts. Caused by updates of the spacecraft database, flight experiences, or changed S/C hardware characteristics, maintenance of the FOP might be required. The procedures need to be updated including additional validation sessions and an official release. A crucial factor is to keep the FOPs under strict configuration control to ensure that only validated and released procedures are used for flight.

Satellites that are built as a series and have a commercial background have the advantage that the flight procedures will be prepared by the manufacturer in a good maturity state. One-of-a-kind missions like scientific satellites or new models will require substantially more work to be done on flight procedure development and the control center may be asked to contribute in that work.

2.2.3.3 Anomalies and Recommendations

In the previous chapter, flight procedures were introduced as the central element for flight operations. It is obvious, however, that in spite of optimal mission preparation, a comprehensive set of FOPs and intensive testing of all space and ground components beforehand, anomalies during the mission cannot be avoided. All kinds of glitches and malfunctions, in the space segment or on ground, might cause a disruption of the running activities and require proper reactions to resume nominal flight operations again.

Anomalies

Anomaly handling is the process of a controlled reaction on problems and anomalies, which is required by common quality management standards. A usual approach is to process ground related problems, e.g., control room or ground network hardware or software problems, by the established issue tracking system of the ground facility by generating nonconformance or discrepancy reports for logging and troubleshooting. Space segment related problems are covered by the FOT. A work flow has to be established to issue a corresponding anomaly, to inform the involved persons for problem solving and to decide the corrective measure (see next paragraph). All steps of the process will be logged. The size of the work flow and the number of roles involved in the process depends on the project size and its complexity, but at least the Flight Director and a responsible subsystem specialist need to be involved.

Recommendations

Recommendations are the controlled way to introduce and process additional actions or changes to the planned flight operations. A recommendation typically consists of a short description of the context and purpose of the desired action and step-by-step orders to be performed. All kinds of actions can be addressed, e.g., sending of an additional command and altering a pending or execution of a so far unplanned flight procedure. Within the process of anomaly handling, recommendations represent the corrective measure.

Key element for the recommendation is the at least a four eye principle, i.e., the recommended action needs to be checked and approved by involved engineers and the person in charge of the flight operations (Flight Director). An example of a recommendation work flow is described as follows: A member of the flight ops team prepares the recommendation, which is checked and complemented if necessary by the affected subsystem specialists or an assistance team responsible (e.g., during LEOP), and finally approved for execution by the Flight Director. The recommendation is completed after its execution and the confirmation of the expected results by the SSE and the Flight Director.

All recommendations have to be noted in a written form, either on paper, or using a dedicated tool. The steps have to be signed by the corresponding persons/roles. In critical situations, where a quick response is needed, a recommendation can be processed verbally first, but shall be noted and fully logged later.

2.2.4 The Mission Type Defines the Operational Concept

Basically there are four different types of operational concepts which are operated at GSOC: the LEO satellites, the GEO satellites, the deep space missions, and the Human Spaceflight. A typical example of LEO satellites are the Grace twin satellites (Tapley et al. 2004a, b). They were designed to measure the gravitational field of the earth. They were launched in 2002 and use changes in their relative distance and speed to derive information about the local gravity forces.

SATCOMBw is a communication satellite family consisting of two satellites identically constructed. The first one was launched in 2009, the second in 2010. Their representatives are examples for geostationary satellites.

Galileo was a deep space probe whose science objective was to explore Jupiter and the Jovian system (Belton et al. 1996). It was launched in 1989 by the Space Shuttle and its lifetime ended by the deliberate entry into Jupiter's atmosphere in 2003. Its mission significantly contributed to our understanding of the Solar system. A GSOC team was located at the JPL to support operations during the mission duration.

For more information on Human Spaceflight, please see Sect. 7.1.

2.2.4.1 Low Earth Orbit: GRACE

The GRACE satellites are orbiting the earth at an altitude of approximately 430 km in a polar orbit with an inclination of 89°. The orbital period is approximately 93 min (Fig. 2.8).

For LEO missions one of the typical operations tasks during the routine phase is the "housekeeping" of the spacecraft. Nearly all the telemetry parameters are monitored. Each SSE monitors his subsystem and reports to the Flight Director in case any of the parameters do not behave as expected. The team will not only react when yellow or red alarm situations are indicated in telemetry, the SSEs will also perform long-term monitoring where the data will be recorded and plotted over an

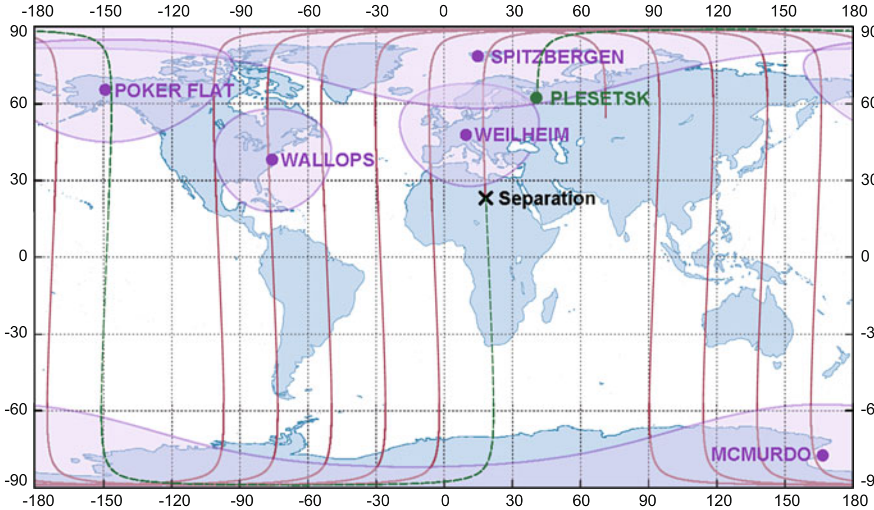

Fig. 2.8 The ground track after launch (the *green line* is the track before first acquisition which was over Weilheim) of GRACE shows that only a few contacts are possible

extended time frame, sometimes even years, and evaluated by the experts to perform predictions and react in advance to trends and tendencies. During the nights and weekends Command Operators will watch the telemetry during the station passes and inform the Flight Director and the responsible SSE immediately in case any anomalies occur.

Another major task during LEO routine operations is the attitude and orbit determination. The flight dynamics team collects the orbit measurement data (typically GPS measurements) and calculates the exact orbit and generates the information for the next maneuver. The FOT will command the satellite with the orbit maneuver data and execute the maneuver. Afterwards the specialists of FDS will check the satellite orbit again und evaluate the accuracy of the orbit maneuver. For the detailed description of the execution of the maneuver please see Sect. 4.1.

The payload operations are depending on the corresponding payload the satellite is carrying. In case of the GRACE satellites there are scientific experiments which have to be commanded and monitored. The recorded data of the satellite payload are dumped over the ground stations and will be distributed to the scientists afterwards. A dump is the download of a data storage that contains previously recorded telemetry. All these tasks have to be organized and scheduled and this will normally be done by the mission planning tools. For further details see Sect. 5.2.

Because of the low altitude of the LEO satellites, the contacts with the ground stations are quite short. Normally at an altitude of ca. 500 km the time that the ground station has contact to the satellite is only around 10 min. With scientific missions like GRACE, the number of ground stations is limited due to the costs of each ground station contact. As a consequence the number of ground station contacts with one or two ground stations leads to about five contacts per satellite

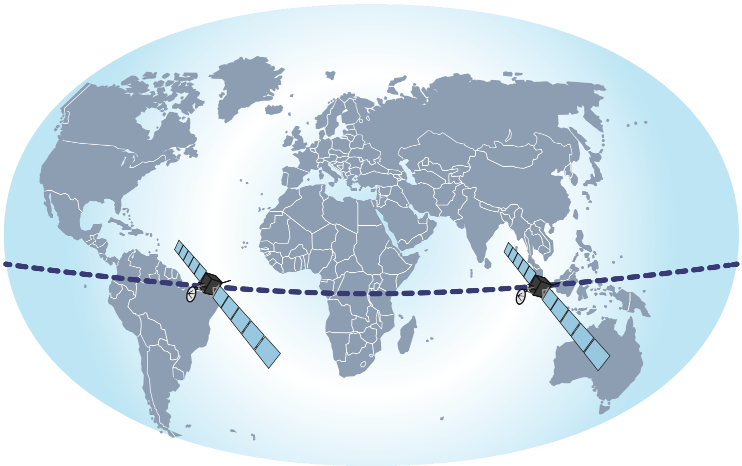

Fig. 2.9 In a geostationary earth orbit the satellite seems to "stand still" over one location; that means the ground track is only a spot. The *shaded area* indicates the regions on the earth from which the satellite can be seen

per day. This is the reason why most of the routine operations are conducted with time-tagged commands. These commands are sent to the spacecraft during the contacts. They have a certain time stamp which means they will not be executed right away but be stored on board until their intended execution time. This allows the execution of orbit maneuvers, payload-specific actions, or software uploads, which normally take a long time, any time and not only during ground contacts. Without the option of time-tagged commands, the operations would be much more complicated and insufficient.

The command verification can either be done via mechanisms provided by the data handling system (see Sect. 6.2) or by checking if it was executed in the telemetry during the next pass, which would be the indirect verification.

During the LEOP the ground station network is of course more extended than during the routine phase, although there normally still is no full coverage. Depending on the mission there are four or more ground stations involved. In case of emergency operations there will be booked additional ground stations to support the mission and bring it back to normal operations as soon as possible.

2.2.4.2 Geostationary Earth Orbit: SATCOMBw

At the altitude of approximately 36,000 km above the earth and at an inclination of 0° against the equator, the geostationary satellite seems to stand still over one location. With this fact the control center has a 24 h per day visibility of the satellite with a single antenna located in a suitable region of the earth (Fig. 2.9). The two geostationary satellites of the German Bundeswehr (Armed Forces) called

COMSATBw1 and COMSATBw2 are located at two positions of the GEO so that they both can be operated through ground stations located in Germany. Due to the permanent visibility, the GEO satellites can be monitored all the time. The spacecraft surveillance including the monitoring of the spacecraft housekeeping is performed with a 24-h 7-days-a-week Operator shift concept. The telemetry surveillance and the long-term monitoring of the data are conducted by the SSEs as described in the chapter above. The orbit maneuvers, the payload operations, and the software upload can be monitored in real time. The use of time-tagged commands is reduced.

The first days of the GEO-LEOP are not fundamentally different from a LEO-LEOP. The control center takes over the satellite after separation from the launcher and checks out the bus systems. Notable differences from a LEO are the facts that a propulsion system has to be prepared and that solar panels have to be unfolded. Usually the satellite is released by the launcher into a highly elliptic trajectory, the geostationary transfer orbit (GTO). As shown in Fig. 2.10, this orbit has a height above the earth ranging from 300 km in perigee (point closest to the earth) up to 36,000 km in apogee (highest point of the orbit). Variations are possible. Some launchers allow to release the satellite directly in the GEO; others are using a super-synchronous orbit (apogee higher than 36,000 km height above the earth) to be able to change the orbit inclination more efficiently. The orbit duration in this phase is about 11 h. Station visibilities are several hours long. Interestingly, seen from the ground station the satellite can change the apparent direction of movement in the sky, as depicted in Fig. 2.10. The general approach is to be able to monitor and control the satellite as continuously as possible and therefore to include many ground stations in the LEOP network. For critical operations (e.g., for maneuvers) it is advisable to even have redundant stations available. The maneuver sequence also dictates that critical events are often happening at night hours and are possibly not compatible with convenient work shift arrangements.

When the satellite is stable and the main bus components have been checked, orbit maneuvers with the satellite motor are performed around apogee position. This maneuver sequence will increase the perigee height in several steps to an altitude of 36,000 km and to the desired longitude. Once the spacecraft reaches its final position in the GEO, only one ground station is needed for continuous visibility (Fig. 2.11). In case of SATCOMBw there are additional ground stations available from which the satellites could be operated in the event of contingency situations. These ground stations have to be tested on a regular basis to make sure that the handover works flawless if needed.

The limited possibilities of the spacecraft of the 1990s to determine their orbit and attitude influence the classical GTO operations scenario. In the time when the classical GEO communication satellite was designed, no GPS was available and its use is limited, as during large parts of the orbit (above approximately 3,000 km height above the earth) the GPS signals are not continuously receivable. Ranging and angle tracking from ground has to be performed. Also the attitude was usually determined with sun and earth sensors only. This influences operations during

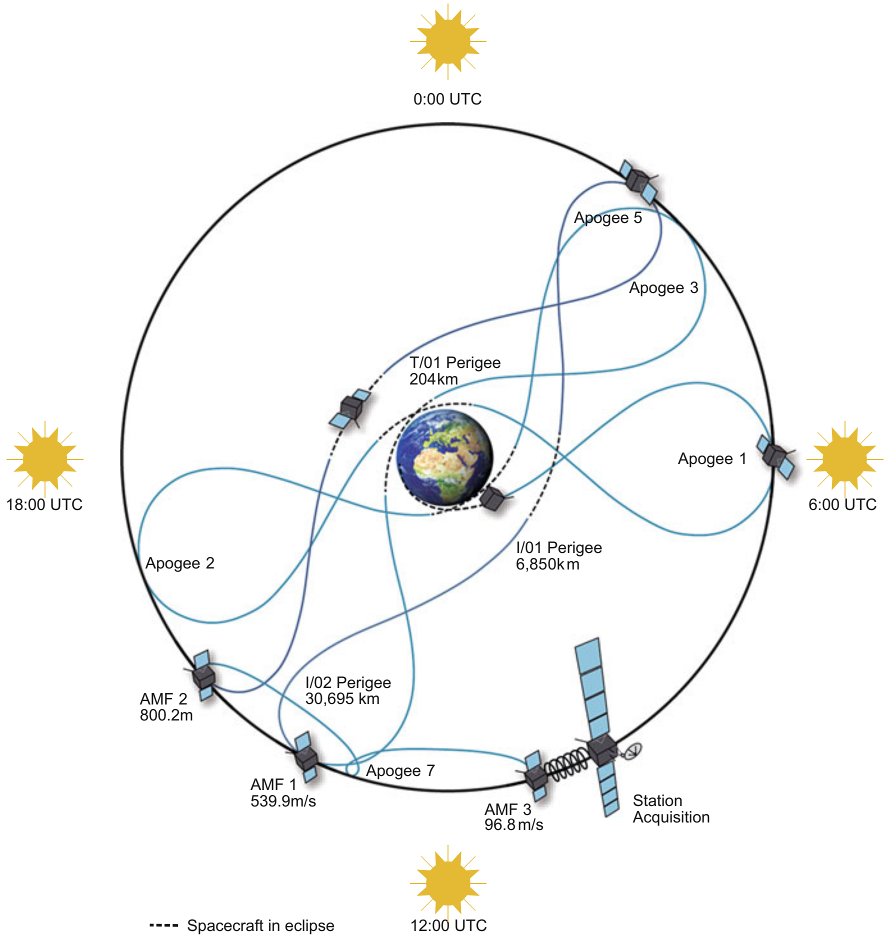

Fig. 2.10 The LEOP trajectory of Eutelsat II-F2 in the earth-fixed coordinates as seen from the north. This illustration allows to see the movement of the spacecraft across the sky as seen, e.g., from the ground stations (Illustration based on P. Brittinger, H. Wobbe, F. Jochim, DLR)

eclipse phases and during maneuvers, resulting in costly ground networks, complex activities, and long waiting periods. Modern spacecraft will be equipped with star sensors and high-sensitivity GPS receivers, which will reduce some of the limitations as described in Zentgraf et al. (2010).

The routine phase of a geostationary satellite is mostly focussed on payload operations. In regular intervals of a few weeks, the satellite's position has to be corrected, as perturbation forces influence the orbit. Short boost maneuvers will be executed that normally do not disturb payload transmission. Repeater payload activities have to be performed in long intervals. They are described in Sect. 6.6. Apart from long-term trend analyses of the equipment, the remaining fuel mass has to be calculated. Comparatively small teams can easily supervise entire fleets of satellites at the same time.

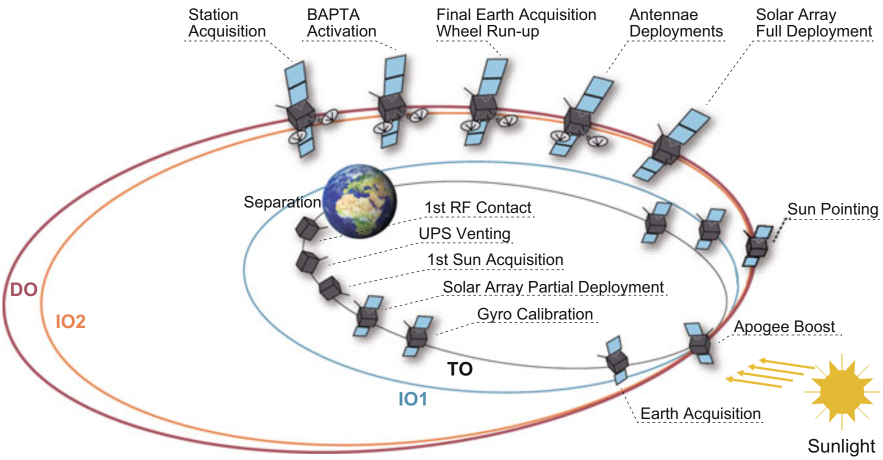

Fig. 2.11 The LEOP orbit of SATCOMBw in inertial coordinates. This view shows the successively larger orbits after the motor firings

Finally an aspect that should not be neglected is the fact that GEO missions are different from all other space missions in that they are often highly commercial in nature. Serious business plans and large sums of money rely on a timely service entry. The spacecraft involved are mostly from an established series with proven equipment. The spacecraft are extremely expensive and expectations are high. On the positive side, manufacturers will provide a complete set of documentation including flight procedures along with a convenient full scale software satellite simulator.

2.2.4.3 Deep Space Missions: GALILEO

The typical deep space mission spacecraft is operated outside of the earth's gravitation field at a long distance from earth. It normally takes a very long time, compared to LEO or GEO missions, for the spacecraft to reach its destination. It took the deep space mission GALILEO nearly 6 years to reach the Jupiter orbit with the goal to study the planet and its moons. It took the spacecraft five flyby maneuvers at Venus and Earth to gain the necessary speed to reach Jupiter. The satellite needs a very high degree of autonomy to detect failures by itself and to autonomously recover from them because the response from ground can be delayed by several hours due to the radio signal travel time. Because of the geometry of the orbit, the contact durations are often several hours long. Depending on the mission phase only one antenna may be used, which results in daily repeating periods without contact (Fig. 2.12).

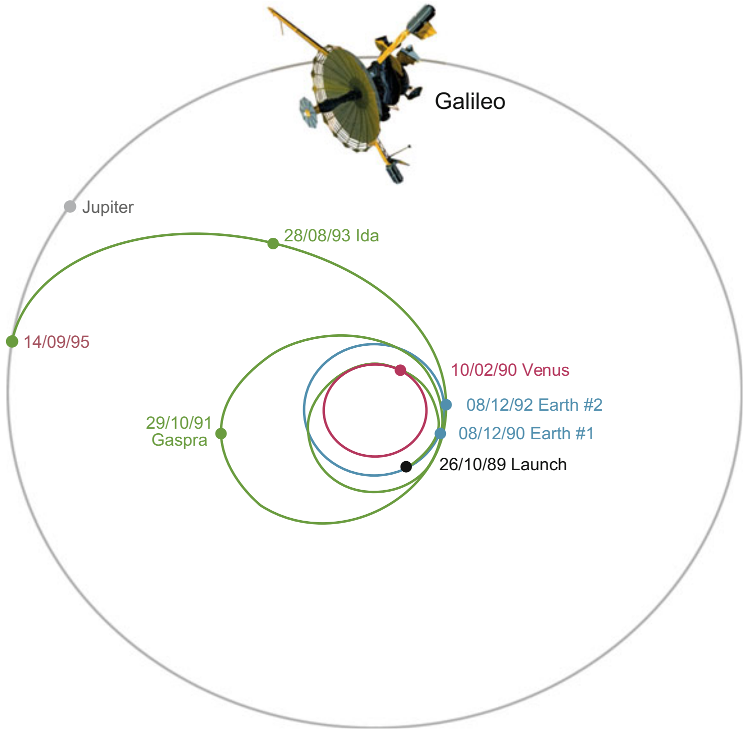

Fig. 2.12 GALILEO's long journey to the Jupiter System

A challenge coming from the long transfer times is to keep up the expertise in the operations team, from the manufacturer side and in the scientific community. Projects like Rosetta can span entire careers. It is wise to build up the necessary spacecraft knowledge inside the control center and to preserve access to the engineering model. For more details on deep space missions, please see Sect. 7.3.

2.2.5 *Summary*

The operational concept is very much dependent on the mission type and may vary for the different missions. The composition of the Flight Control Team is in turn driven by the requirements derived from the operational concept; however, some basic commonalities can be deduced.

2.3 Flight Experience

Ralph Ballweg

This section will cover examples of lessons learnt at DLR/GSOC, particularly in the course of multiple LEOP phases of communication satellites. It will then handle the process of dealing with system contingencies, mostly on spacecraft side and wrap up with several spacecraft anomalies and the attempts to deal with them.

2.3.1 Statistics

From 1987 to the year 2002 GSOC supported on an average of one launch per year LEOPs of geostationary communication satellites. Among these were two series of six almost identical satellites for the provider EUTELSAT from 1990 to 1995 and 1998 to 2002. We will use the first series of EUTELSAT II satellites to demonstrate how leaning curves can develop. The satellite TV-SAT 1 will be used to illustrate problems that occurred in the course of a mission.

At first we will take a look at the LEOP duration (Fig. 2.13). The LEOP operations for a geostationary communication satellite can typically be performed within 2–3 weeks. These operations include the positioning, configuration, and in-orbit testing of the satellite bus. Short LEOPs are desired by the customer in order to bring the satellite into service as early as possible.

Proficiency and reliability of the ground system, its components, and the team have a direct impact on LEOP duration as experience grows which can be seen in the next paragraphs. Over the course of the six Eutelsat II missions, the LEOP durations were cut from 18 days with EUTELSAT II F1 in 1990 to 11 days for EUTELSAT II F6 in 1995. Flight F5 was lost due to a launcher failure.

There were several areas where refinements and optimization improved the performance:

(a) Station acquisition strategies
(b) Procedures
(c) The SOE (sequence of events)
(d) And within the control center hardware and software tools which for example allowed a faster analysis of data for further processing like ranging data and expedited maneuver calculation

Another indication of the level of maturity of the operations concept is reflected in the number of Engineering Change Requests (ECR) and Non Conformance

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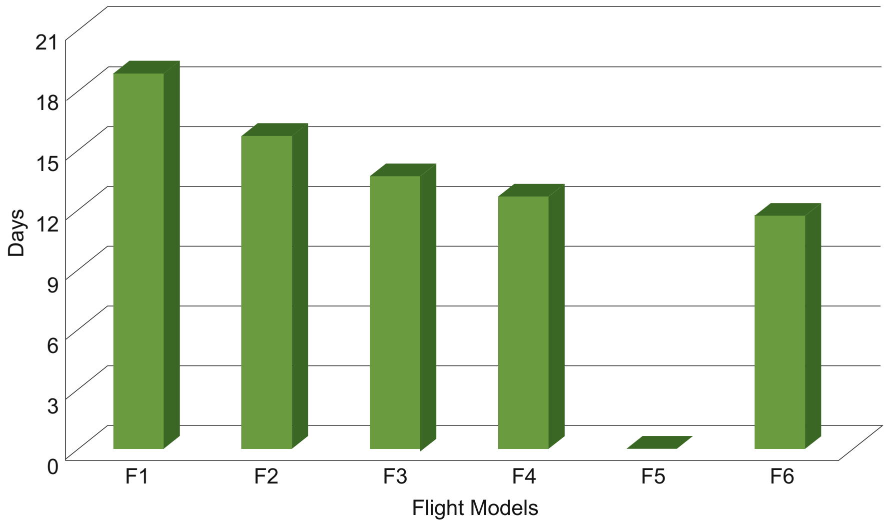

Fig. 2.13 Duration of LEOP for a series of EUTELSAT II satellites

Reports (NCR) that were issued in the course of the mission preparation (Fig. 2.14). Configuration control during mission preparation and mission performance phase is formally managed by ECRs and NCRs. An ECR is raised whenever it is intended to modify a certain topic to the specifications or the existing configuration. An NCR is issued whenever a deviation with respect to the specifications is observed or if a subsystem differs from the expected behavior during the mission preparation phase.

The number of ECRs dropped from Flight 1 with ~170 to ~50 for Flight 6. The high number of change requests at the beginning is easily explained with the fact that the ground segment had to be configured for a completely new mission. For the next launch the change requests already dropped to about 60 CRs as a result of the previous experience. The slight increase for Flight 3 was due to the fact that for this mission the launcher was changed. While for the previous two launches the satellite was mounted on an Ariane; for this mission a Lockheed Atlas was selected. This launcher placed the spacecraft into a super synchronous transfer orbit and the launch took place from the Kennedy Space Center in Florida. The differences that incurred were different interfaces to the launcher and launch site, different ground station selection and schedule, and considerable updates to the flight dynamics software to include a perigee maneuver to lower the apogee.

Again a small increase can be detected between Flight 4 & 5. At that time GSOC made a change in its control facility. It moved to a different building and also implemented new hardware with the corresponding operating systems being adapted.

The decrease in ECRs from 5 to 6 again is not as large as can be expected because there a change in the spacecraft hardware was implemented that resulted in updates to the ground software. Those were in particular in the satellite power subsystem.

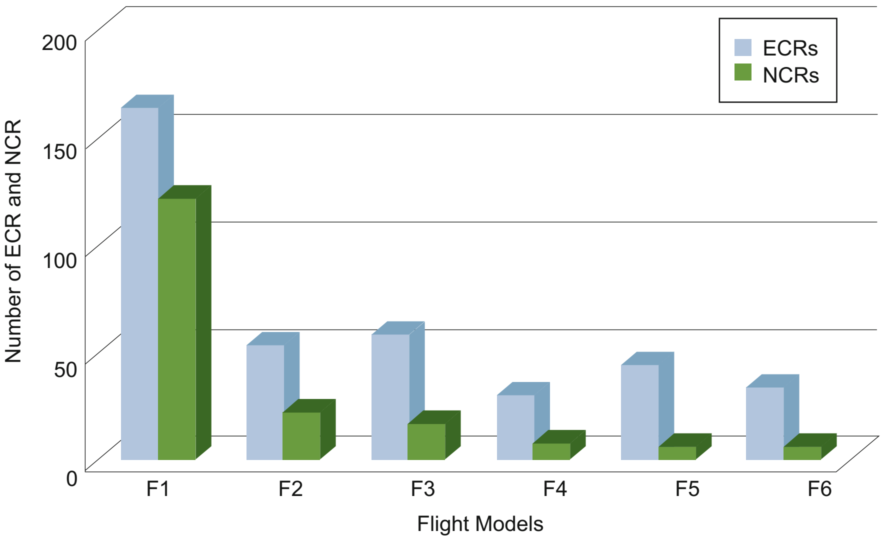

Fig. 2.14 Number of ECR/NCR for EUTELSAT II series

During mission execution all deviations from the nominal procedures and actions caused by unexpected and non-nominal satellite behavior are handled by Satellite Anomalies Reports and Recommendations (SAR). SARs can be issued by any person on the mission control team, control center personnel, as well as representatives of the satellite manufacturer or the end customer. SARs are issued in case of:

- Unexpected and non nominal S/C behavior (not covered by prepared procedures)
- Online procedure changes
- Online mission sequence changes

In the course of the launches from Flight 1 to Flight 4 one can see a significant decrease in the numbers of SARs from roughly 200 down to 60 (Fig. 2.15). Flight 5, even though it was a launch failure, had some SARs because last minute changes to the database and procedures were introduced, which caused changes to the operational system at GSOC.

Flight 6 had an increase in the SARs because there were several modifications to the spacecraft bus, in particular to the power subsystem.

2.3.2 *Interpretation of Telemetry*

This next chapter shall give an example of which kind of information can be gained through detailed analysis of telemetry. As an example we are using a plot of the spacecraft receivers Automatic Gain Control (AGC) (Fig. 2.16).

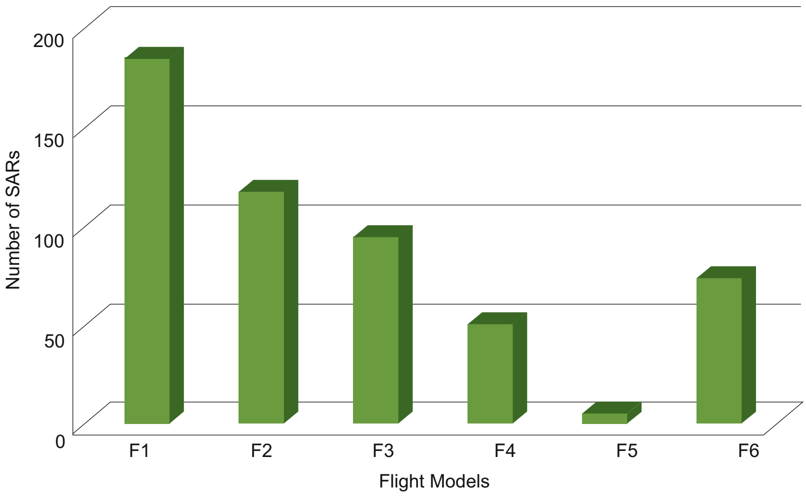

Fig. 2.15 Number of SAR for EUTELSAT II series

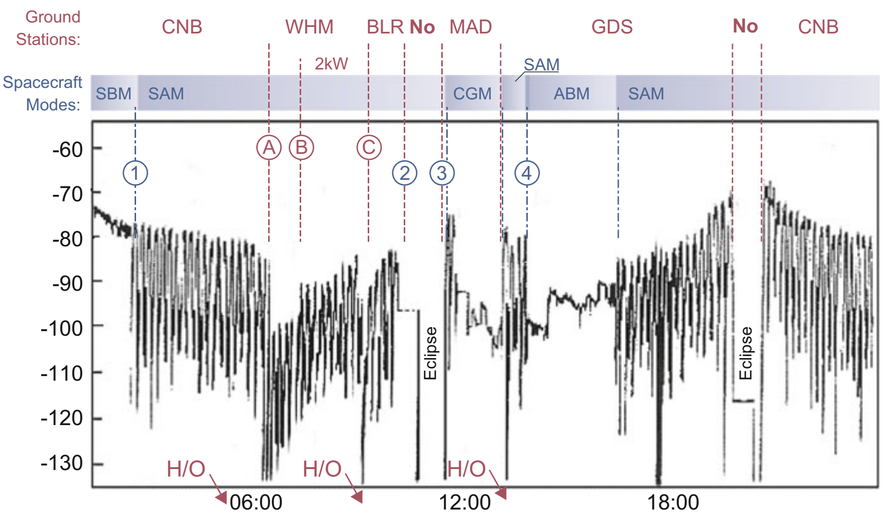

Fig. 2.16 Telemetry data "Automatic Gain Control" (AGC) during the first day of a LEOP

The uplink AGC—a telemetry parameter which is transmitted from the S/C to the ground—indicates the onboard measured signal strength of the telecommand carrier, uplinked from the ground station. The various S/C attitude modes, the evolution of the orbit, as well as the ground station coverage are reflected in this AGC plot of the first 24 h in the transfer orbit.

At first view the graph seems to represent a very erratic behavior of the telemetry value plotted. We see periods of constant or moderate changes, long time spans of extreme oscillations, and gaps without any telemetry. But still this represents actual behavior of the telemetry. How can we interpret this?

First let us take a look at the spacecraft, its configuration, and the activities performed during the first 24 h:

From the start of the plot during the Standby Mode (SBM), we see a rather stable but slightly decreasing level of the automatic gain control level. Here the spacecraft was released by the launcher with a predefined attitude at a certain altitude and distance to the ground station. The decrease in signal strength displays the increase in distance between the ground station and the satellite, as it flies within the GTO towards the apogee. Thus the loss in signal strength is related to the increasing distance to the satellite.

At Point 1 we command the satellite into sun-pointing or sun-acquisition-mode (SAM). The satellite is rotating around the X-axis, which is pointing to the sun. The oscillation with an amplitude of roughly 25 dBm is caused by the fact the there is only one receive antenna. Due to the rotation of the spacecraft, this antenna is sometimes pointing towards the earth, sometimes away or shielded by the satellite bus. This oscillation is also a possible mean to determine the rotation rate of the spacecraft. The graph also shows steady decrease in signal strength up until around 6:00, when it starts increasing again. This is the time when the satellite reaches apogee, the point furthest away from the earth on its orbit, after which the relative distance to the earth becomes smaller again. The overall dip in the AGC level around 06:00 will be explained a little later.

Between point 2 and 3 there is a gap in the telemetry, which indicates a loss of signal. At this time the satellite was passing through perigee and due to the low altitude there was no station available to receive a signal.

At point 3 the signal was acquired again with the spacecraft in SAM, shortly afterwards the satellite was commanded to "Gyro-Calibration-Mode" (GCM) which is a 3-axis stabilized mode. There the antenna is pointing in a fix direction. Therefore there is no fluctuation in the receive signal strength. The telemetry shows a rather constant value that is only decreasing due to the rising distance to the ground station. After completion of the GCM, the spacecraft was returned to SAM which can be seen by the fluctuating telemetry values.

Point 4: The spacecraft is configured for the first Apogee Boost Maneuver (ABM). This is again a 3-axis mode with more or less constant receive strength. After completion it returns to SAM, interrupted by another eclipse during a perigee pass with no ground station contact.

But there is another way to interpret this plot and receive other information. We will focus now on ground activities. From the beginning to point A, we acquired the signal of the spacecraft via the ground station Canberra (CNB), with the AGC level decreasing. At point A the signal drops to a minimum, indicating basically no receive signal from the satellite. At that time there was a ground station handover from Canberra to Weilheim (WHM) with a short interruption in the uplink. The quality up the uplink, i.e., the receive strength, was unsatisfactory, with it dropping basically down to minimum depending on the S/C attitude. So, at point B the uplink power at the ground station was increased from 1 to 2 kW which resulted in a satisfying receive signal strength. Point C marks another station handover with a brief interruption of uplink, this time from Weilheim to Bangalore (BLR). Other stations used during this period were Madrid (MAD), Goldstone (GDS), and again Canberra, in that order with the handovers clearly identifiable.

2.3.3 Failure Probability Vs. Operational Experience

The typical evolution of mission failure probability shows peaks at the beginning of mission, i.e., the LEOP and in the region of planned End of Mission (EOM) (Fig. 2.17, dashed line). An experienced operations team, assisted by the satellite manufacturer is required to cope with the LEOP risks. Operational experience is growing throughout the following routine phase; for long-term mission this experience might get (partially) lost on the operations side as well as at the manufacturer (experts leaving the teams by various reasons).

- Thorough operational documentation required
- Never change a winning team...

First let us take a look at the failure probability during a mission. It starts out with a rather high likelihood of problems at the beginning; one could call it "infant mortality."

The most likely reasons at this time are:

- A launcher failure, which does not really effect the S/C mission control team.
- Failures induced by stress during launch, e.g., vibrations.
- Units or instruments experience space environment for the first time and react differently than expected.
- Operations that are time critical and singular. Time critical event for example are the deployment of solar arrays to charge the batteries or the activation of heaters to keep propellant from freezing. One time executions can be deployment of an antenna or the activation of pyros.
- And finally design or manufacturing errors that were not discovered during testing, like faulty sun sensors.

Once the IOT phase is successfully completed, the likelihood of failures drops significantly. The spacecraft is operated in a stable configuration without many changes. Most problems from that time on originate from single event upsets, like a reboot, equipment failures, or in many cases human/operational errors.

Close to the nominal end of the mission (EOM), the failure rate increases again due to aging effects on the equipment and exhausted resources, which makes resource management an important issue during the course of the operations.

The operational experience of the mission control team is shown in Fig. 2.17 with solid and dotted lines. At the beginning of the mission there will be a team with a high level of experience. This is based on the fact that on the one hand the core team members are chosen from staff that has already supported similar tasks and on

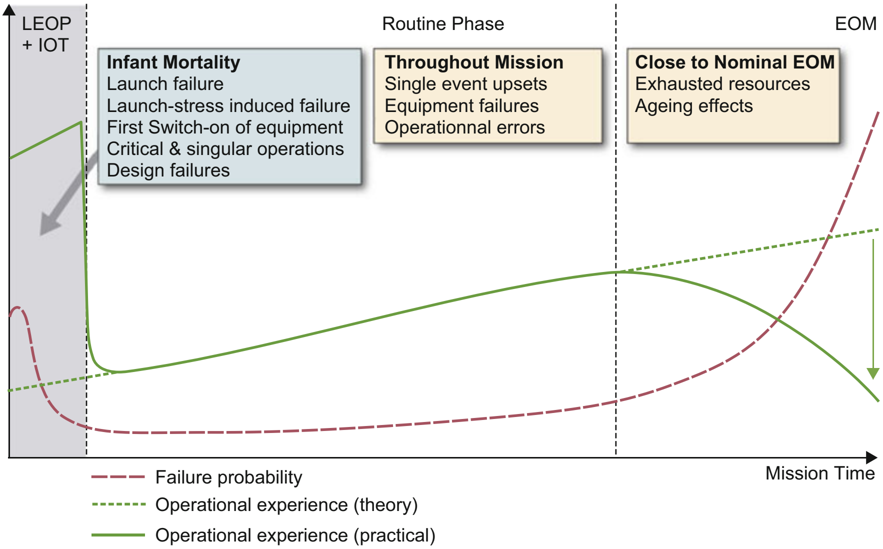

Fig. 2.17 Failure probability vs. operational experience

the other hand that by the time of launch the complete team has participated in the preparation phase, analyzing requirements and databases, defining the mission control system, writing procedures, testing the system, and participating in simulations. The qualification increases during the LEOP, as this is the most demanding phase with all the testing occurring there and typical problems happening in that phase.

Shortly after LEOP is completed the experience level drops significantly (solid line). This is caused by several facts: first of all, during LEOP the team is augmented by staff from the satellite manufacturer, that leave the control center once the spacecraft is checked out, then people are taken off the team to work on new missions and only a core team is left to support the routine operations phase. Once the operational phase is established, the experience increases gradually again in the course of the mission.

Unfortunately, experience shows that once the mission has reached a mature lifetime, the experience level of the team starts to decrease again due to natural attrition, team members leaving for other jobs, or because of budget constraints and finances being cut. More often than not this coincides with the time when failures on the spacecraft start to increase again due to the exhaust of resources and the aging effects.

2.3.4 *Contingency Handling*

This section covers the aspects of contingency handling during operations. The first level of contingency handling should be covered by the onboard FDIR software. The objective of the onboard FDIR, also known sometimes as Redundancy

Management, is the survival of the spacecraft after a single failure for a specified time span. For detailed information on how FDIR is embedded in the Data Handling subsystem please refer to the corresponding Sect. 6.2 about onboard data handling. Communication spacecraft are typically designed to survive up to 48 h without ground intervention. The individual steps are:

- Failure detection by check of exceeded thresholds: The onboard FDIR has limits stored in its software that are periodically checked. Rules can be defined within the FDIR process that further refine the reaction to limit violations, e.g., a limit has to violate a specific number of times with within a fixed number of checks before a reaction is triggered. This is to avoid a premature reaction based on a faulty reading of a telemetry value. In addition to analog values like temperatures, voltages, or angles also status bits, ON or OFF, or complex error words are monitored.
- The next step is the isolation of the problems. This means either taking a faulty piece of equipment out of the loop by switching to a backup unit, if the issue cannot be tracked to a single instrument or processor then a switch to a backup mode is performed or in extreme cases the complete satellite is commanded to a safety mode by turning off nonessential loads and bringing the attitude to a mode that ensures power and thermal balance for survival.
- The spacecraft should be able to survive in the corresponding configuration for a predefined period of time. Depending on how sophisticated the software is in rare cases, the possibility exists that the satellite can recover to its nominal configuration autonomously, e.g., as reaction to brief attitude deviations.

Once the onboard FDIR software has triggered the detailed failure analysis of the cause of the incident, any recovery actions, like returning to normal mode and restoring the mission, if possible, has to be performed by the mission control team on ground. The FDIR functionality can not only be triggered by the malfunction on a unit on board, a single vent upset, which is not reproducible, can also be the cause for a reaction or reconfiguration. So one possible recovery action can simply be to restore the nominal configuration and the issue is solved. If there is actually a defect on board, then there are more consequences to be considered: Limits within the mission control system have to changed, procedures updated, and the FDIR software need to be rewritten to reflect the operations on the backup system, because different telemetry parameters need to be checked and new switching automatisms and reconfigurations have to be defined and implemented.

On the other hand it is always preferred to detect problems before the onboard software takes action. To support this, the ground system has similar functions implemented to the ones that can be found on board. These are more refined and enable detection of issues before they become a problem. For example there are two stages of out-of-limit conditions defined, a warning and an alarm stage. The warning indicates that the situation has to be monitored, but no immediate action is necessary. The alarm stage calls for action; otherwise an instrument or function could be lost.

Other means of detecting satellite issues by ground monitoring is to recognize secondary effects by analyzing telemetry. These can be the unexpected change of telemetry values like temperatures, currents, or sensor values even though they still stay within the range of nominal values. Other effects can be attitude perturbations that are not recognized by the onboard software as problems or the loss of up-or downlink. And finally the long-term analysis of telemetry and consumables. This can give an indication of the remaining lifetime of a unit, instrument, or the mission.

The isolation of problems by ground activities can be of course much more subtle than just switching to backup units. The first step is the identification of the problem or failure. Starting with a systematic approach one first identifies the area of the issue: Is it an operator error, is the cause within the ground system, or is it a malfunction within the space system?

Based on this analysis the corresponding action, best recovery, can be chosen. These can result among other things for example in improved training for operators, new procedures, update of the ground system, databases, or hardware. In case the space system is affected, the goals return to a normal operational configuration, recover the mission, restore payload, and as much as possible activate the nominal equipment again. A closing action can be the necessary update of the onboard FDIR logic in case equipment is permanently damaged.

The baseline for a controlled reaction to system contingencies must be verified and approved processes and procedures. Often the handling of a contingency might be too complex to be handled by a single procedure. In those cases flowcharts are useful. In the following we show how the analysis of a problem can develop into a complex process and flowchart.

In the following we discuss the FDIR process which is activated whenever no telemetry data is received (Fig. 2.18).

If at the expected AOS time no telemetry is received by the FOT, it is first checked whether the receiving ground station sees a downlink (D/L) signal/carrier from the spacecraft (Fig. 2.18, left side). If that is not the case we try to verify if the orbit used for the prediction is correct. The sources of information are the Flight Dynamic (FD), the launch provider, and possible co-passengers. They might be able to tell if they received signal from their spacecraft. At this point the control center should be able to determine if the predictions were made with wrong orbit data, modify the orbit, predict and correct the antenna pointing angels and adjust the antenna. In case the used orbit data prove to be correct, the ground station configuration should be checked and corrected if necessary.

The next possibility could be that a downlink carrier is received and a subcarrier is modulated to it (Fig. 2.18, right side). This would point to a failure in the data handling subsystem of the spacecraft and a manual reconfiguration on the spacecraft has to be attempted. In this case the activities would involve commanding “in the blind,” mean without telemetry verification. If that was the root cause the telemetry should appear in the control room.

But there are many more possible causes for that problem and it would be out of the scope of this chapter to describe them all. Just to summarize, beside the mentioned possibilities of wrong orbit, ground station problems or DH subsystem (of which there could be several), the anomaly could also be within the TCR subsystem, the spacecraft attitude.

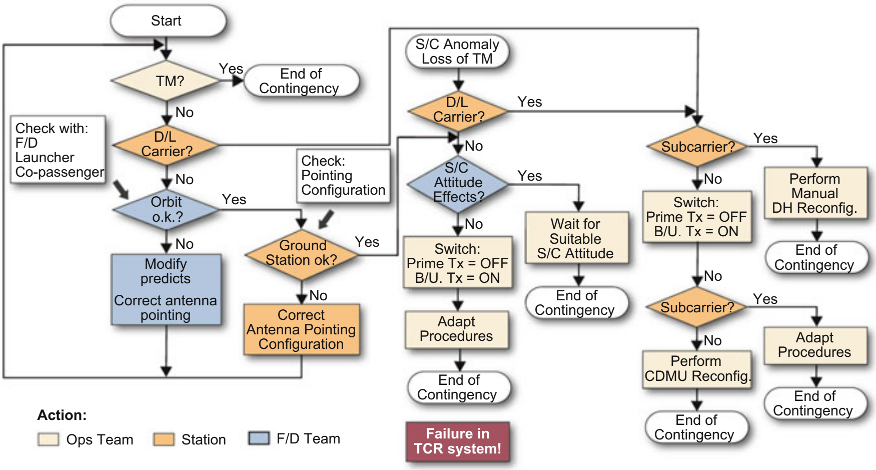

Fig. 2.18 Flowchart for FDIR process in case no telemetry is received

2.3.5 Mission Example TV-SAT 1

TV-SAT 1 was the first commercial German communication satellite, a joint French-German coproduction. It was a small satellite of two metric tons with five transponders on board, designed for direct broadcasting of TV programs (Fig. 2.19).

TV-SAT 1 was a prime example of a mission that went wrong. It provided the mission control teams with unique challenges. The problems GSOC encountered already very early in the mission:

- A partial deployment failure
- Gyro failure
- Thruster temperature problems

In the following we describe the partial deployment failure in more detail including the numerous tests, to determine the exact cause of the failure. We will cover the immediate actions, evaluation of impacts, offline failure analysis, failure investigation, and recovery action attempts. The problem was caused by one of the solar arrays which failed to deploy at the very begin of the mission (Fig. 2.20).

2.3.5.1 Failure Analysis

The partial deployment of the solar arrays was part of the automatic onboard timer function, triggered by S/C separation. The failure was detected by the status of the deployment microswitch at the first contact, when the S/C was still in eclipse. The immediate actions were:

Fig. 2.19 Artist view of TV-SAT1

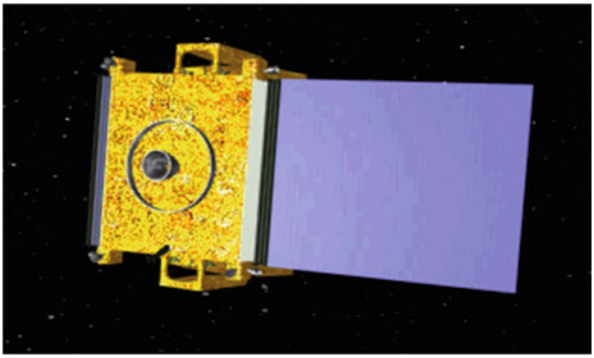

Fig. 2.20 Partial deployment failure of TV-SAT 1

- Checking the ground database, if there was any bit inversion—no failure was detected.
- After eclipse: checking the output power of the affected solar array—the low power levels confirmed the unsuccessful deployment.
- Next the functioning of the onboard timer was checked and there was indication of a failure.
- Following these checks the procedure called for the sending a manual deployment command sequence. Those commands were sent, but there was no change in the status.
- Finally we sent a command sequence to fire the redundant deployment pyros—again with no effect.

Now the operations team knew that the S/C was in deep trouble.

In order to be able to proceed with the operations, the failure impacts on the mission had to be quickly evaluated. It was found that the S/C was generally safe from system side; impacts on the subsystems, mainly power, thermal, and attitude, were reviewed and found to be not mission critical at this mission phase. Apart from some tests it was decided to proceed according to the nominal sequence of events,

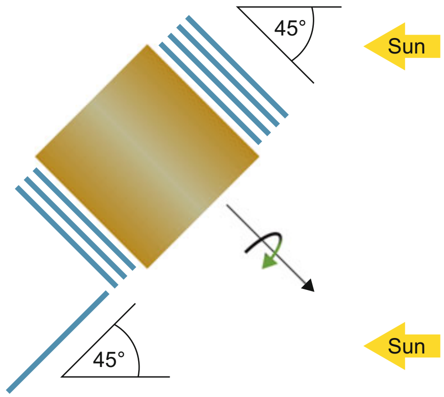

Fig. 2.21 Failure investigation of TV-SAT 1—tilt S/C by 45°

namely to perform all operations including the first apogee boost maneuver. A number of satellite flight procedures as well as elements of the ground system (e.g., alarm flags) had to be modified and adapted.

An offline analysis by the manufacturer identified a number of more than 50 possible causes for the failure, some of them very unlikely. Test strategies and procedures were defined and developed in order to reduce the number of possible causes. In addition recovery strategies and procedures were prepared for the different failure scenarios. A review of the Failure Mode Effects and Analysis (FMECA) revealed the blocking of the receive antenna as a fatal mission impact of the non-deployed solar array.

For the failure investigation new tests had be defined. Once that was completed, the procedures had to be prepared, validated then executed, and the tests evaluated. Here are some of the tests that were performed:

The first test was to tilt the spacecraft by 45° (Fig. 2.21). The solar array currents were measured to give a rough determination of a possible deployment angle. The expected accuracy was ~2°. The result was that if any, the opening angle was less than 2°.

The next test called titled "Shadowing": Principle of this test was to illuminate the panel at low solar incidence angles. Any stirrups holding the panel would thus cast large shadows, which would be measurable from the reduction in the amount of current generated (Fig. 2.22). No conclusions could be deducted from this test; it was not sensitive enough to distinguish between the possible cases: No stirrup closed/1 stirrup closed/2 stirrups closed/3 stirrups closed.

This was followed by "Current Mapping": This test consisted in measuring the power output of the north panel for a variety of solar incidence angles (Fig. 2.23). Power output was expected to vary as the cosine of the angle between the normal to the panel and the solar incidence direction. Any offset in this cosine response could correspond to an opening of the panel. The conclusion from this test was a maximum opening of the panel by 0.85°. This test was repeated after all satellite maneuvers and events to find out if they had any effect on the solar array.

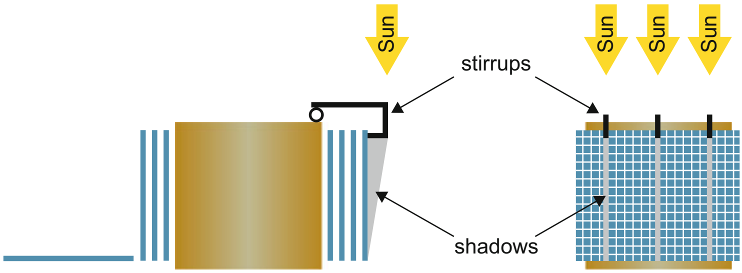

Fig. 2.22 Failure investigation of TV-SAT 1—shadowing test

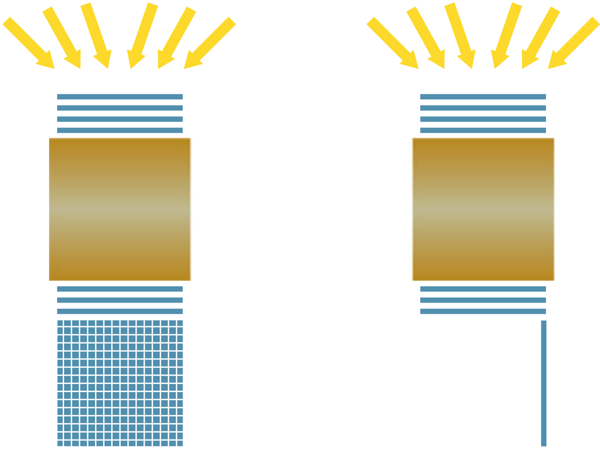

Fig. 2.23 Failure investigation of TV-SAT 1—current mapping

Another test consisted of shaking of the spacecraft angles (Fig. 2.24). Purpose of the shaking tests was to determine the resonant frequencies of the north panel. By shaking the S/C at particular frequencies, oscillations were set up in the panel. After stopping the excitation, the continuing oscillations of the panel were transferred to the S/C body and measured with the gyros. No resonant frequencies in the expected range could be measured, another indication of a fully blocked array.

2.3.5.2 Recovery Attempts

As a result of all the tests the number of possible failure causes could be reduced to 13. A very likely cause for the unsuccessful deployment was a jamming of one or even more stirrups. Different recovery actions were performed, unfortunately all without success:

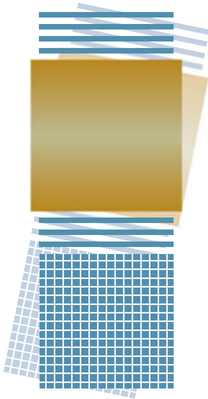

Fig. 2.24 Failure investigation of TV-SAT 1—shaking test

- Fast spin mode around the S/C Y and Z axis in order to exert forces on the stirrups and the panel which could overcome the friction in some failure cases.
- Performing apogee boosts and station keeping boosts in pulsed mode in order to excite proper resonant frequencies with high amplitudes.
- Exposing the panel and stirrups to alternating hot and cold temperatures.
- The solar array full deployment, the BAPTA activation, and the shock of the antenna deployment could also overcome some failure cases.

2.3.5.3 Final Actions

Although it was finally not possible to deploy the solar array, a high amount of operational experience could be gained. For the preparation of the following flight models, preventive actions could be taken for all remaining 13 possible failure modes to make them a success.

In the end it was found out that the actual cause was that it was missed to exchange the transport stirups with the flight stirups. As a consequence payload operations were not possible because the non-deployed solar generator prevented the receive antenna from full deployment. The TV-SAT 1 mission was terminated about 6 months after launch. Therefore the satellite was injected into a 325 km over-synchronous orbit by 2 boost maneuvers. All subsystems were deactivated in order to avoid any risk for other satellites.

The S/C telemetry transmitter was switched on again after 7 years for a short time in order to gain attitude information for the Experimental Servicing Satellite (ESS) study (see Fig. 7.7). The switch on was successful and the satellite signal was acquired at the first attempt.

All following TV-SAT flight models could be operated successfully in orbit.

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Chapter 3
Communication and Infrastructure

3.1 Control Center Design

Marcin Gnat and Michael Schmidhuber

This chapter describes the design aspects of a typical Mission Control Center (MCC). The German Space Operations Center (GSOC) is taken as an example (Fig. 3.1).

The Mission Control Center—as the name implies—is the central ground facility of a space mission. It is the central point where all data and management information concerning the spacecraft are consolidated. These data are received, checked, and processed, decisions are made and—in case of an emergency—the respective procedures are performed in order to restore the nominal conditions of the mission. The way how the MCC operates is defined by its design which specifies its capabilities, flexibility, and robustness. MCC operations are also defined by the people working within, primarily of course by the Flight Operations Team, but also by all personnel responsible for interfaces and infrastructure. Their work in the background is equally important. Finally, the design of the MCC needs to conform to the customer requirements and provide a safe and secure environment for spacecraft operations. This includes not only purely technical solutions but also the respective environment for the people working there.

Within this chapter we focus on several aspects of the design of a control center. At first the necessary infrastructure is analyzed. Then the design of the local control center network is examined followed by the software needed.

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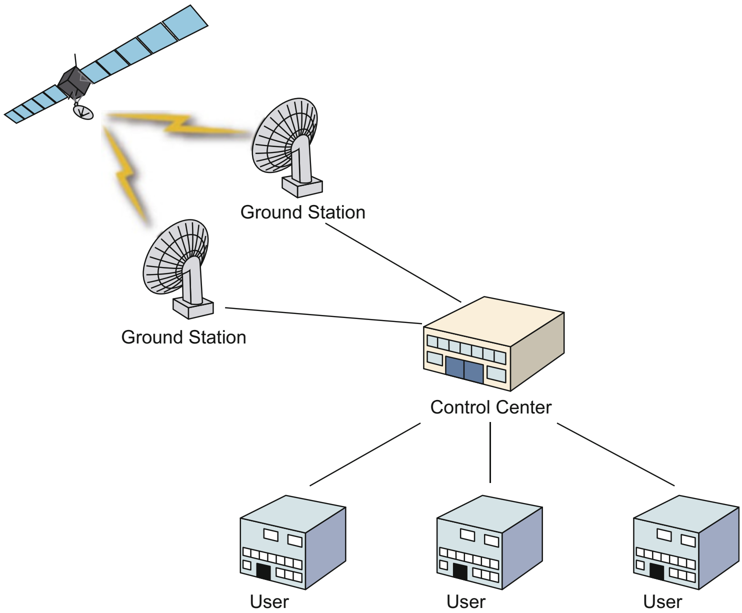

Fig. 3.1 Position of the mission control center in the ground segment

A so-called Multi-Mission operations concept allows greater operational flexibility and an easier phasing of new missions. The decision whether the MCC is laid out as a multi-mission or single MCC should be made early in the design process, as it has far reaching effects on the overall design, especially on the IT infrastructure and the network. The operations concept (Multi-Mission or Single-Mission) also has a large impact on how personnel, especially the mission operation teams, are assigned.

In this chapter we focus on a multi-mission environment, based on the GSOC design. The multi-mission design is typically more complex to design but offers a greater flexibility for the integration of future projects. However, there are also some situations where a multi-mission concept may not be adequate. Simultaneous operations of missions which have no or only minor similarities (e.g., because of different requirements, security aspects etc.) cannot be grouped together easily into a multi-mission environment. It would be difficult, for example, to integrate a scientific mission whose data are more or less publicly accessible with a military mission with very strict security requirements.

In the following we will discuss different aspects of the design for the facility itself (the building), different office and operational subsystems, as well as IT hardware. The construction of the building itself will not be a topic here; however, some important issues will be mentioned.

3.1.1 Infrastructure

Identifying a suitable location for the Control Center building is the first task in planning the infrastructure. Several important aspects have to be considered here. Adverse geological conditions should be taken into account. E.g. geologically insecure zones (earthquakes) as well as areas subject to frequent flooding should be avoided if possible. Nevertheless appropriate measures should be taken to make the Control Center less vulnerable to natural or technical contingencies. This can be achieved, e.g., by using a redundancy concept on different levels. Redundant power supply is essential. Uninterruptible Power Supplies (UPS) can provide constant power to all of the MCC's systems in case of short power breaks, or bridge a time gap until diesel generators run up. The latter can provide power even for several days. A completely independent backup control center will provide the highest level of redundancy.

Concerning redundancy in the area of communication infrastructure, it is advantageous if the control center is located closely to a large city hub, where multiple independent connections to the telecommunication network are available. A dedicated communications antenna for the MCC provides additional independence in case the terrestrial communication lines are cut.

Maintenance aspects should also be given a great deal of attention. Occasionally, it will be necessary to replace part of the equipment (hard drives, switches, workstations, etc.). In most of the cases this can be done without an impact on operations, sometimes however it becomes necessary to really shut down parts of or even the complete MCC. In such a case planning and coordination is vital. Affected projects need to be informed, maintenance tasks need to be scheduled carefully and backup solutions shall be discussed (what happens if the maintenance tasks take longer as planned or are not successful). Additionally it should be considered that some equipment is undergoing considerable deterioration when its power supply is switched on and off repeatedly. Many electronic devices react very sensitive to such power cycling. The damage caused by this power cycling might even be worse than that which caused the initial maintenance action.

The MCC facility has to meet security standards, as applicable by laws, company policy, or project requirements. An access control system includes the basic technical infrastructure (like protective doors, door key management, respective locking policies) as well as more sophisticated elements like access terminals (with key cards) with respective key card management, monitoring cameras in critical rooms and on corridors, as well as alarm systems (intruder alarm). Additionally security personnel shall be available on site all the time. More or less strict visitor control may be implemented, depending on projects located at the facility as well as on facility capabilities. In general, visitors will have no access at all to the network and data processing facilities, whereas conference or display areas with mock ups of satellites may be treated as low security zones).

The facility needs to be equipped and maintained for the safety of personnel. This includes emergency exits and signage, fire and smoke alarm systems (which might be connected to the local fire brigade), and different kind of fire extinguishing facilities. Especially the latter might be essential for larger computing or power (UPS) facilities. A central fire suppression system can be installed (e.g., Argon inert gas extinguishing system or similar sophisticated systems targeted to avoid damage to the equipment). Finally procedures for the case of fire have to be developed and in place, and especially for the spacecraft operations the procedures for the spacecraft operators have to be precise and should clearly define under which circumstances the control room has to be evacuated and when and if running systems should be shut down and how they shall be recovered afterwards.

3.1.1.1 Control Rooms

At the heart of the MCC are the control rooms. Depending on the available resources and needs, there may be several control rooms. They can be of different sizes and might serve different purposes. They can be assigned permanently to certain space missions or might be assigned only for specific phases of a mission (like e.g. for a LEOP—Launch and Early Orbit Phase). Mostly control rooms will be equipped with air condition not only for human comfort but also for the computing hardware. As already mentioned above, the room shall have emergency exits and has to provide enough space for the operational team and additional equipment like printers, voice, and video systems. A photocopier shall be available as well, possibly however outside the control room itself, as it produces a considerable amount of noise, which is not desirable inside the control room.

Space mission may draw a large amount of public attention, which shall however not affect operations. Visitor areas with large glass windows allow a direct view into the control room and give an impression of the operation of a spacecraft mission. It should be possible to use blinds (or a similar solution) to cover the windows for situations or missions that do not want public visibility (Fig. 3.2).

Control rooms should allow changes in the configuration of consoles, as spacecraft missions might have changing requirements to the control room layout during mission lifetime. This requires a forward-looking initial design of elements like cabling for network, telephone, voice, and power.

Control room consoles require typically not only access to the operational systems but also to the office network (for e-mail, documentation) and to the Internet (e.g., to allow representatives of the satellite manufacturer or customers to access their company network over VPN).

Facilities like restrooms or the coffee kitchen should be located close to the control room to minimize the time spacecraft operators need to spend outside the control room.

Fig. 3.2 Control room of German Space Operations Center (GSOC)

Besides the central spacecraft control rooms, there are several other rooms which are typically necessary during operations. The flight dynamics team typically has its own control room for the LEOP phase, to be able to closely coordinate with the flight team and quickly exchange data products. The Network and Systems control room is a communication hub connecting all incoming and outgoing connections to and from the MCC and providing voice communication with the outside world for monitoring and coordination of the Ground Station Network or other important external operational interfaces. Also staff from satellite manufacturers or customers may need dedicated rooms where they can perform important off-line tasks in close vicinity to mission operations. These rooms may need specific access control.

3.1.1.2 Public Space in the Control Center

Finally control centers need enough office space for its own employees as well as for guests. Depending on the overall purpose of the control center facility, different approaches to the layout of the areas for public space, presentation, and catering may be chosen. Many facilities for military or communication purposes will require only relatively small presentation and catering areas. Large national control centers which perform LEOPs and many public relation activities will require large areas for the interested public, along with press and meeting rooms as well as for the catering of visitors. Space missions are still at the cutting edge of technology and hold fascination for many people. A control center offers the unique possibility of fostering the interest in space missions and technology for the public. Therefore it should provide respective means for public information and education.

Before we take a closer look at the computing network in the next chapter, we quickly go through some other subsystems and elements, which are not less important, but not so much in the focus here.

3.1.1.3 Server Rooms and Computer Hardware

The server room is equipped with server, routing, and switching equipment. When designing the system, the servers require special attention, as their reliability, flexibility, and capacity are defined to a large extend by the effort which is needed for maintenance and extension of the system. Until recently, the design was dictated by the use of powerful servers. However, this concept was not very flexible; one defective element of the server caused its total outage and bringing it back to full operation took a considerable amount of time.

As the flexibility and redundancy requirements increased (accompanied by the increase of server count) the focus shifted to so-called blade servers. They can be packed densely and support the increase of applications. At the same time they provide backup capabilities and easy exchange of defective modules. Currently, however, virtualization is the trend. Virtualized application servers are even easier to maintain. They may be seen as a single available space for computing and storage resources which can be used in a very flexible way. In the times when physical servers were used, running 10 applications meant to have 20 physical servers in place (including backup). The same situation in a virtualized environment, however, requires only two physical machines (for prime and backup) hosting ten virtual applications each. The decision which technology to use, should be made on a case-by-case basis. Each technology has its pros and cons, and so for example virtualization with all its advantages is not very well suited for applications with intensive network traffic (because a single physical port of the server hardware has to be shared). The virtualization principle is, however, the prerequisite for a further improvement in maintainability of the control room hardware (operator consoles). This is achieved by using thin client terminals. As these thin client terminals do not contain local hard drives or other moving parts, their reliability and expected lifetime are much higher than that of conventional PCs.

Other advantages are decreased power consumption and less heat buildup, which in turn will result in much less effort for providing adequate air-conditioning for both computer hardware and human operators.

Data storage is also an important topic to consider. For most of the office applications, the hard drives of the office computers and possibly Network Attached Storages (NAS) may be sufficient. Storage of the spacecraft data and documentation requires a different approach, as security, continuity, and collaboration issues have to be addressed. Here again several solutions may be considered—starting from high capacity NAS-like storages, through SAN (Storage Area Network) for short term and working storage, through data vaults and long term archives (in form of magnetic tapes with automated refresh mechanisms).

3.1.2 Control Center Network

In this chapter we will take a closer look at the design and security and maintenance aspects of the control center network or LAN. The computing network is the backbone which connects all of the subsystems within the MCC. It is the linking element and at the same time it protects specific systems from unauthorized access.

3.1.2.1 Network Topologies

Figure 3.3 shows the principle of the network connection between GSOC and the Ground Station in Weilheim. Two separate paths can be identified, each designed to fulfill its special requirements. On the left side there is the so-called Office Path connecting the Office LANs of GSOC and Weilheim together (e.g., to allow teams at both sites to exchange documents). This connection is realized with the help of a VPN over the DLR campus network. On the other side there is a highly reliable, redundant SDH connection for the real-time satellite data, connecting Operational LANs of both sites. The same data router covers also Voice over IP (VoIP) traffic multiplexed on data connection, to allow also highly reliable voice communication for operators at MCC and the ground station.

The above example shows that there are two independent network branches, the Office and Operational LANs. This separation is also realized in the network structure within the control center itself. It reflects the solution which is introduced to fulfill security requirements.

The Operational LAN (called also Ops-LAN) is the high security network area. It is physically separated from other networks with only very limited access from the outside. File transfers are allowed only over specific FTP server located in so-called DMZ (Demilitarized Zone) and real-time connections from ground stations are also only allowed over a firewall and only to trusted locations, having similar high security operational networks.

The mentioned DMZ is the typical separator between different LANs. It consists of another network part with two entry points protected by firewalls. DMZs typically contain only Firewalls and FTP servers, in case of some specific application the outermost DMZ may, however, also include some application (Web) server to provide some MCC services to the outside world.

The Office LAN is a network part used typically in offices of the MCC and is intended for general office work like viewing and working on documentation as well as e-mail access. The Office-LAN has Internet access; however, this functionality is restricted (e.g., the Internet can be accessed from Office-LAN, but the Office-LAN cannot be accessed from the Internet, so it is only a “one-way” access). The Office-LAN is controlled and only registered devices may access the network and the IP addresses are managed centrally by the network administrator of the MCC.

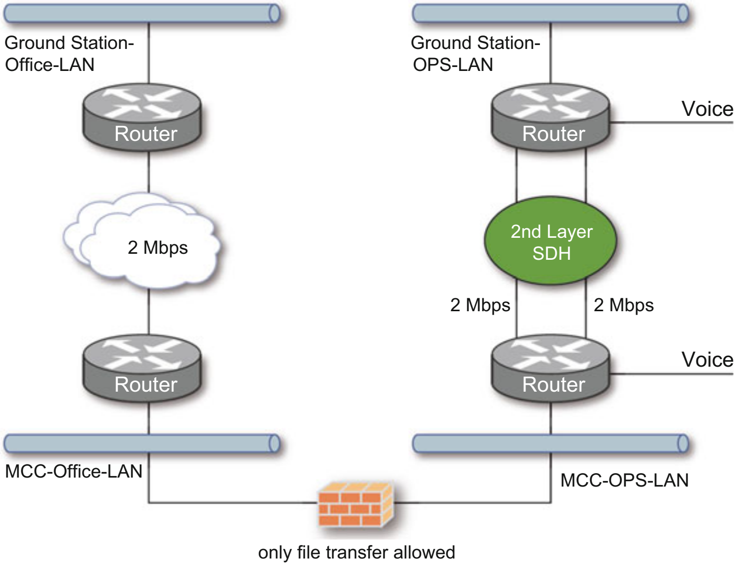

Fig. 3.3 Control center example of link to the ground station

Another network presented in the figures below is the so-called Ops-Support-LAN. This is not necessarily needed for each control center, and rather encompasses some supporting systems, which may need some more access to the outside world, but at the same time are very important for the operative system (i.e., deliver command files). In the example below, the Ops-Support LAN contains Flight Dynamics and Mission Planning systems.

Every area has its own service and security segment, hosting proxies, virus scanners and authentication, name, time, and file servers. Clients are not able to open direct connection to hosts outside their LAN area; the connections can only be established via the proxies in the service segment. These proxies are directly connected to virus scanners in the DMZs, where the in- and outgoing traffic is scanned.

Figure 3.4 shows an example of an MCC network. The real-time TM/TC connections with the ground stations and external partners are shown at the bottom of the figure (OPS-LAN). Also essential operational files are transported on these network parts. It can be seen that to transfer a file from Ops-LAN to some external customer over the Internet, it has to pass a few firewalls and DMZs, before it will be made available on an FTP server at the outermost DMZ.

3.1.2.2 Network Technologies

The MCC network is based on the TCP/IP protocol and underlying Ethernet. The type of cabling depends on the available resources, in principle however fiber optic

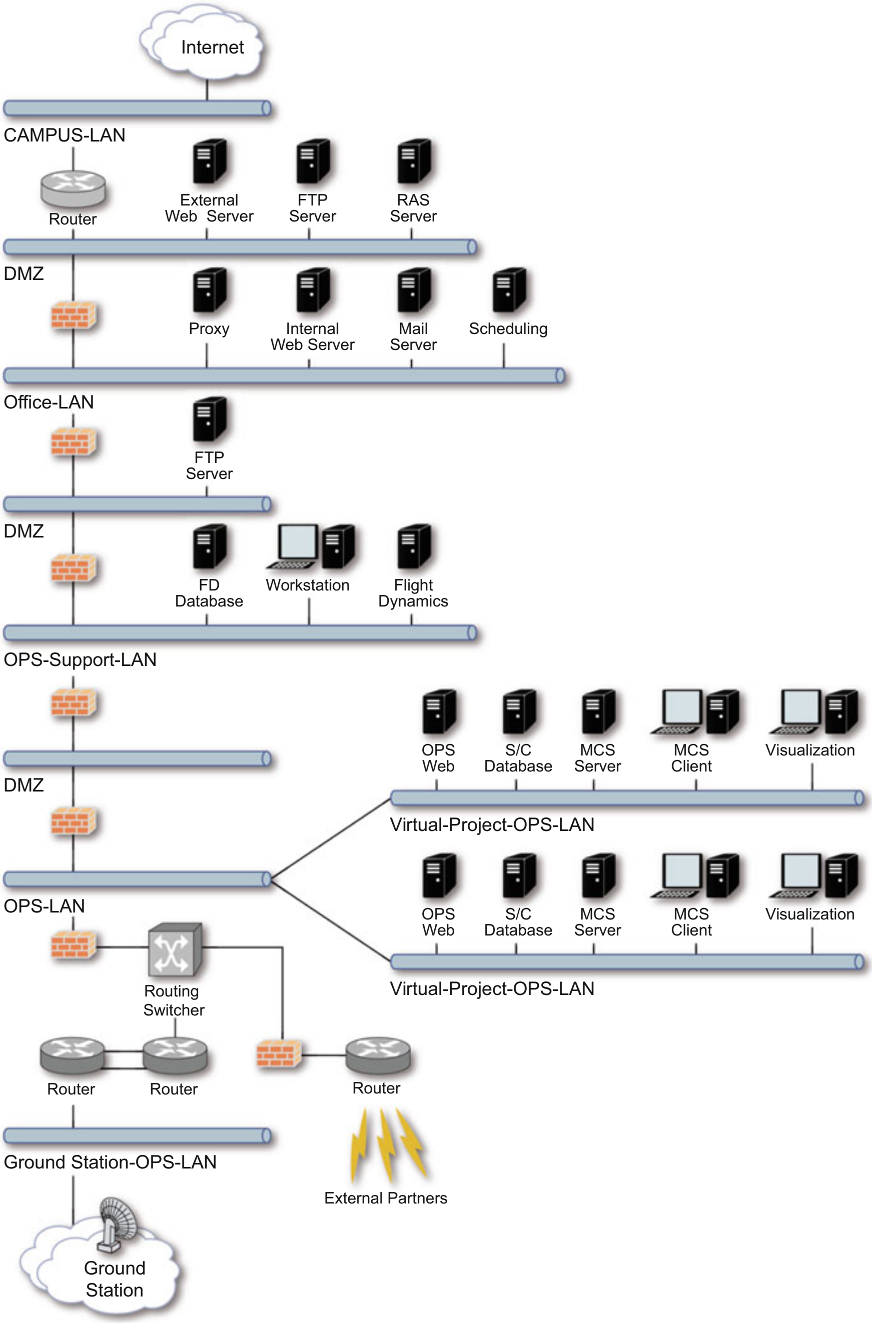

Fig. 3.4 Control center example network

cabling offers a bigger potential for future upgrades and—also an important factor—are tap-proof. Typically it also provides higher bandwidths and thus data rates, so that for future expansions only equipment like routers or switches needs to be exchanged. Exchange of cabling on the other hand is very expensive and may require a high effort. Also respective equipment with interfaces for fiber optic cabling is in general much more expensive than for conventional cables. Therefore it may be reasonable to implement a hybrid solution (fiber optic between big hubs and connections to the single users with means of copper Ethernet cable) for office equipment (PC or laptops).

As already mentioned the control center network constitutes the backbone of all of operational systems and is strictly critical for operations. This requires respective support from skilled personnel. Depending on the size of the control center, and thus also the network as well as projects being supported, it may be required to have the respective network support personnel available permanently, either through shift work or on-call service.

Another aspect of the network is its maintenance. In many cases it will be possible to perform maintenance with minimal or completely no impact on the running operational business. This may be the case if for example equipment needs to be exchanged and this may be performed during the time between two satellite passes. But even then, and especially when there may be a real impact on operations (like an outage of the systems availability for some hours), there is a need for appropriate planning and preparations (i.e., backup regulations, spacecraft on-board autonomy).

3.1.3 Control Center Software

3.1.3.1 General

Within this chapter a few prominent examples of specific software used within a control center are presented. Their generic functionality will be explained using GSOC applications as example. Standard programs and software packages like office software and the operating systems are not discussed here.

The software of a control center is special and often custom-made, although there are a number of commercial software packages on the market which support satellite operations. However, they are typically not cheap (because the customer base is small) and it always has to be evaluated how they fit into the specific environment of the control center and satellite project under discussion. The software not only has to process, e.g., telemetry or perform orbit calculations, but also needs to provide interfaces to other packages or systems. Some of these interfaces may be proprietary, which makes the usage of off-the-shelf products impossible.

Ground station and control center software and especially its interfaces can be divided analogue to the data flow paths.

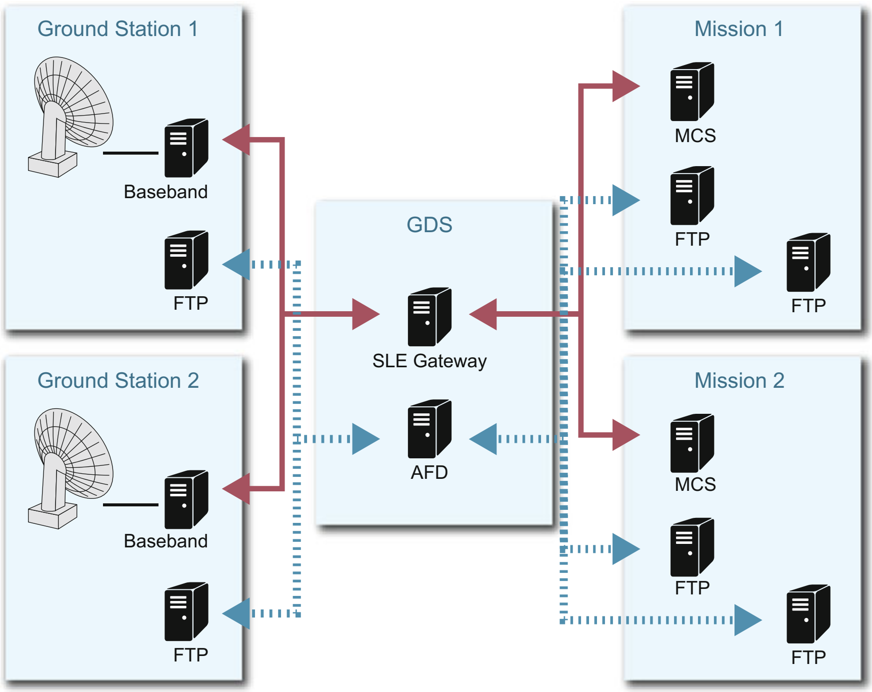

Fig. 3.5 Online (*solid lines*) and off-line (*dotted lines*) data transfers

We differentiate between real-time data stream, which flows from the spacecraft through the ground station to the MCC on one hand. On the other hand a multitude of data has to be transported in the form of files. These are also called "products." They contain data like event predictions, converted telemetry extracts, input from external parties, ground station predictions, etc. They have to be exchanged between internal and external partners. Due to the high number of files, the need for timely delivery as well as reliability, most of the transfers should be done automatically. It happens more or less asynchronously and is not as time critical as the real-time data stream.

The real-time data is often called "online" and respectively file transfers are named "off-line." The diagram in Fig. 3.5 shows these two types of information flow between ground segment subsystems.

For both types of data flows there is dedicated software which generates, processes, transfers, and converts it, as shown in Fig. 3.5.

The online communication is realized with four main elements in a chain. The baseband software is usually installed at the ground station site. It performs basic tasks on the lowest level like frame synchronization, error correction, or time stamping. The service provider delivers the data from the ground station to the corresponding MCC. Currently in most cases the Space Link Extension (SLE) service is used here, which is described in more detail in Sect. 3.1.3.2. Here, the service user acts as the counterpart to the ground station on MCC side. Very common is here again the SLE user software, which receives SLE data and provides them to the Monitoring and Control (M&C) system in a respective format. Finally the spacecraft M&C system (called also TM/TC processor) provides the actual data processing and user interface for the flight controllers.

The off-line communication is similarly built out of four components. The generating and processing systems produce or use files; these systems may also include the M&C system mentioned above. There are dedicated storage systems which provide the required hardware, but also the corresponding data management software. This includes in particular also all databases required for operations. There is automated file transfer software. One implementation example for it, the Automated File Distribution (AFD) software, is described in more detail in Sect. 3.1.3.3. Finally, of course security plays a major role in expensive and sensitive satellite missions; therefore firewalls and virus scanning software are deployed as well.

3.1.3.2 Space Link Extension Gateway System

To perform safe communication with sometimes distant ground stations, specific communication means need to be used. This includes the communication lines which need to be ordered and maintained (ISDN, VSAT, leased line). In most cases commercially available lines are utilized. Of course also protocols tailored to the specifics of space missions need to be employed. A protocol which is frequently used is the SLE. SLE is based on CCSDS standards and is widely accepted by agencies and companies operating ground stations, because SLE ensures interoperability. In contrast to previous systems, cross support can be made available, without the need to readapt interfaces for each mission and each customer. SLE is based on a client-server architecture and allows transfer of telecommands and telemetry which are encapsulated within SLE packets and can be transported that way over the WAN (Wide Area Network) (Fig. 3.6).

As mentioned already, the SLE is server-client based. The role of the server is taken by the so-called SLE service provider. The service provider is located at the ground station, and on request it provides the services related to that station. These services are called Forward Command Link Transfer Unit (FCLTU) for telecommand and Return Channel Frames (RCF) or Return All Frames (RAF) for telemetry. These services are specified in detail in the corresponding CCSDS standards (cf. Table 3.2).

On the opposite site of the network, the SLE user is located at the control center. It manages the abovementioned services, performs all required protocol conversions, and acts as an interface to the Satellite Monitoring and Control (M&C) system. E.g., GSOC uses the SLE Switch Board (SSB), which is able to receive telemetry flows from the different stations in parallel and can send them to different

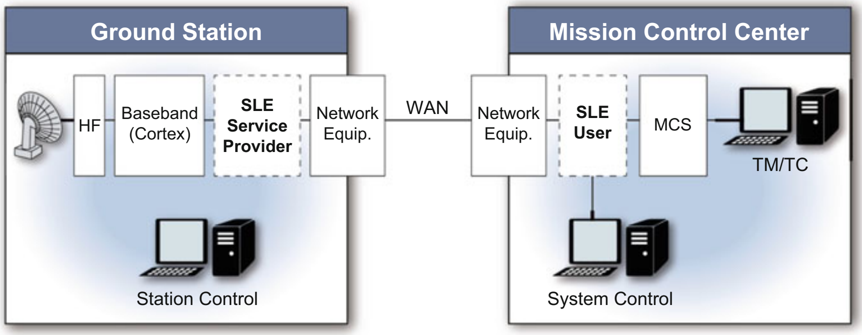

Fig. 3.6 The communication between the ground station and the mission control center via a WAN can be performed using the SLE gateway. For that the ground station equipment and software like the HF and the Baseband interfaces with the SLE Service Provider, which is connected via the network and its equipment to the SLE User on the MCC side, which in turn interfaces with the MCC software and hardware like the Monitoring and Control Software

MCS instances. Alternatively, it receives telecommands from the M&C software, converts them into SLE format, and sends them to the corresponding station.

3.1.3.3 Automated File Distribution Subsystem

The file transfer is a key element in the above described off-line communications.

GSOC is using the AFD, a tool developed as open source by Deutscher Wetterdienst (German Weather Service). This system is used by all GSOC satellite projects. The system is placed in a multi-mission environment, and so each project defines its own file transfer matrix, which acts as an input for AFD configuration. The matrix defines what kind of files shall be transferred, from where to where and how often. As soon the configuration is activated, the AFD system starts the monitoring of the defined directories and performs transfers fully autonomously. AFD is especially useful in case of complex network structures, as it is depicted for the three LANs which are used at GSOC in Fig. 3.7.

3.1.3.4 Spacecraft Monitoring and Control System

As already laid out above, the Spacecraft M&C software package has the tasks to receive and unpack the telemetry, to process the data and display it, to process and encode the telecommands, and to send them out via its interfaces.

It is the central software component used by the Flight Operations Team in the control room of the control center.

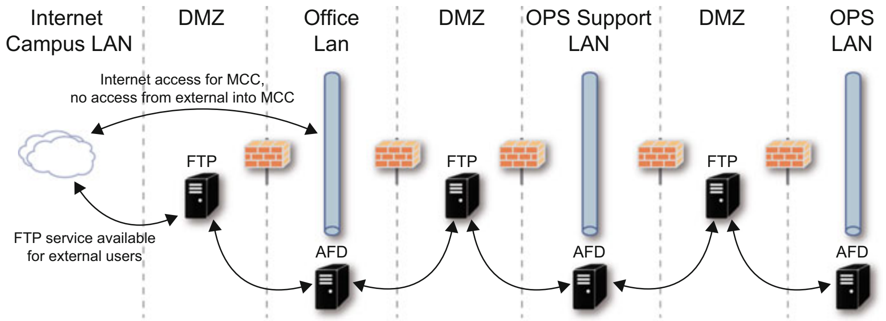

Fig. 3.7 AFD configuration in GSOC network

In many cases this is a monolithic application, but also designs with separated components for different tasks are in use. In the last decades quite a number of different systems have been developed and are commercially available on the market. At GSOC mainly the SCOS 2000 system is used that was developed by the European Space Agency (ESA).

In many control centers and especially for LEOP operations, the mission data needs to be available at various workplaces at the same time. Therefore a server/client functionality is usually included.

As depicted in Fig. 3.8, there are two basic data flow paths in an M&C system. Dedicated interfaces are required on both paths to establish the communication with the data user interface, which in turn ensures the connection with the ground station. Telecommands which are usually grouped in the so-called command stacks need to be processed until they can be broadcasted out via the interface: They need to be encoded and packetized. On the other hand the incoming telemetry also requires processing by the monitoring and control system: The packets need to be "opened" and its content to be attributed to their original parameter. This raw telemetry can still not be efficiently displayed for the flight controllers, since it is so far only a bit pattern, which first needs to be calibrated to appear, e.g., as meaningful physical values like temperatures or currents. Finally there is an option to perform an automatic limit check on selected telemetry parameters: Their values are compared against predefined thresholds, in some cases also more sophisticated mathematical operations are performed. The result of that check can either be just a highlighting of the corresponding telemetry item on the display, an alarm for the flight controller, or even an automated, defined reaction of the system. The central component of the M&C, which allows all the processing described above is the spacecraft database, is also called Mission Information Base (MIB). It contains the definition of the telemetry and the telecommand streams, the calibration information, and the limit definition. Although most M&C systems are developed along standards and are intended for use in all types of missions, it typically requires some effort to adapt the software for each mission. In the end the M&C has to reflect the capabilities of the onboard data handling system of the spacecraft, as described in Sect. 6.2.

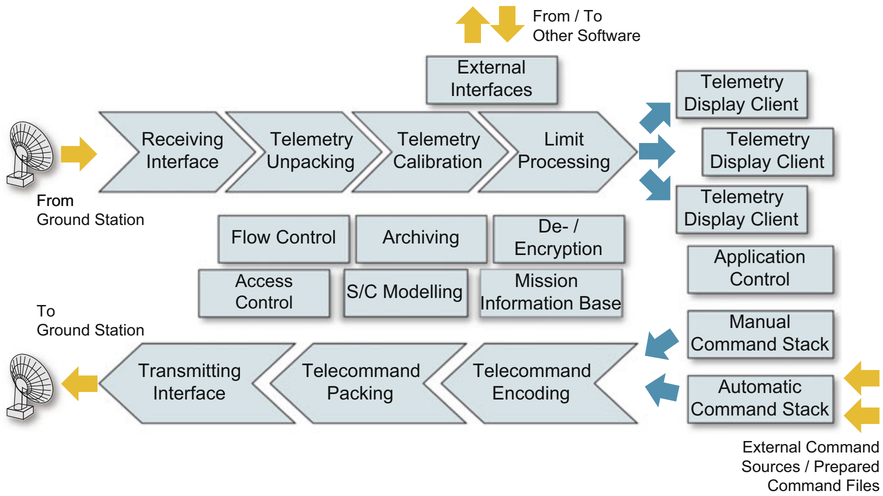

Fig. 3.8 The functional components of a typical Monitoring and Control system. Two paths can be distinguished: Coming from the ground station, some steps of processing are required until it can be displayed on a telemetry client: After being received at the interface of the MCS, the telemetry packets need to be unpacked, the content calibrated, and corresponding limit checks to be performed. On the opposite paths the telecommands of the so-called command stacks need to be encoded, packets are generated, and finally submitted via a defined transmission interface. Further essential components are listed as well

After the acquisition of the spacecraft, the satellite starts to send telemetry and the ground station antenna receives it. The ground station performs demodulation and decoding; first error checks and possibly error corrections to the data stream are applied. All the received data is stored locally in either short term or long term archive (depending on ground station and its applicability). Furthermore, the data stream is being provided on the WAN interface to the MCC. There, the data is received, further decoded, further error corrections are performed, and finally every portion of the actual telemetry is processed, analyzed, stored, and some part of that is displayed at the spacecraft operators console for direct view.

Spacecraft commands are going in the opposite direction. In contrast to the telemetry, which is streaming almost permanently during the contact and often containing redundant or repeating information, commands are being sent only using special operations procedures as described in detail in Sect. 2.2. The telecommand is being sent out of the M&C System at the MCC, packed in respective transfer protocols and transferred over WAN to the ground station. In the mean time, the ground station needs to have established the uplink, which is defined as a stable radio connection with spacecraft. This time the ground antenna is sending and the spacecraft antenna is receiving. Often, to compensate for frequency variations as introduced by the Doppler Effect, a sweep needs to be performed.

The telecommand packets coming from MCC are being modulated on carrier frequency and radiated into direction of the satellite.

3.1.4 Outlook

Although the basic function of control centers is the same and is constant over time, its design is dependent on and evolving with the requirements of the missions it is responsible for. Some design principles are maintained since the beginning and can be found all over the world, while new requirements are emerging with new technologies. In that way each control center will be unique.

3.2 Ground Station Network

Marcin Gnat

The Ground Station Network (GSN) plays a major role in space missions. It has to establish communication with the spacecraft and with other control centers, support specific spacecraft characteristics, and provide operability and safety of the mission. Due to its nature, the GSN takes part in cross support activities between different organizations and agencies.

The GSN covers several functional aspects—the communication path between control center and the ground stations (transporting online data, off-line data, voice), the management of the stations and their antennas, as well as coordination tasks and station scheduling.

Communication with the spacecraft, as an essential part of the spacecraft operations, is mainly represented in reception of telemetry, transmission of telecommands, and tracking. To improve the operability and safety of the mission, there are several approaches. First of all we try to extend the contact time, typically done by introducing additional ground stations to the network and selecting optimally located stations. For some missions (like ISS, LANDSAT, SILEX, and Sentinel) a viable option is the use of a GEO-Relay satellite. This has been used since the TDRS programme to support the Space Shuttle. More relay satellites have become available and the market is expanding (ARTEMIS and in the near future EDRS).

A specific case is the LEOP (Launch and Early Orbit Phase) as contact has to be established assuredly for the first time of the spacecraft's life for several critical tasks. The GSN has the task of reducing the time from separation of the spacecraft from the launcher until the first acquisition. The first acquisition station has to perform the first tracking of the satellite (to allow exact orbit determination), to receive the telemetry to assess spacecraft health after launch. If time allows or it is

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required, the first acquisition station performs also the uplink, to perform time-critical operations like sun-pointing mode or folding out of solar panels.

3.2.1 Station Selection

During design of the GSN, several technical properties, parameters, and requirements need to be considered. They are mostly provided in form of the Space to Ground Interface Control Document, while some others may be found in the Spacecraft Design Document and the requirement documents (starting from Mission Requirements, through Customer Requirements, and finally in Ground Segment Requirements).

The analysis starts with the main mission characteristic, which is the orbit. Based on the knowledge of the location and speed of the satellite during the mission phases, we can decide which ground stations may potentially be used, depending on their geographical location. See also Sect. 4.1.3.1. For earth missions, the orbit type can change during the LEOP, but remains rather stable during the routine phase. Orbit types are categorized by the height (or other words from the distance from the Earth), the inclination of the orbit plane against the earth equator, and the shape of the orbit path.

The majority of satellites are on circular orbits. Here we differentiate between Low Earth Orbit (LEO) with heights of up to 1,000 km, Geostationary Earth Orbit (GEO) with approximately 36,000 km, and everything in between, called Medium Earth Orbits (MEO). Orbits may have different inclination, for GEO mostly being at zero degrees, for polar-orbit LEOs close to 90°, and for many other LEOs somewhere in between. Spacecraft flying in LEO with inclination of let's say 55° may be contacted only by stations which are at maximum such latitude.

Polar stations are of high importance for the polar-orbit earth-observation missions, as they allow having contact practically every single orbit. For missions with low inclination like GEO they are not usable, however.

Other earth satellites are on elliptical (= eccentric) orbits. This results in varying orbit height and velocity, from very low ones (around 100 km only) up to 60,000 km (way beyond GEO). The reason for this can be either a transfer phase between different orbits (e.g., GTO = geostationary transfer orbit), an improved observability from ground (e.g., Molniya type orbits) or scientific requirements.

The GSN needs to be carefully fitted to the mission. In case of highly elliptical orbits, a spacecraft at apogee is visible for many ground stations for long times (hours); at the same time the signal strength is significantly weaker. At the perigee on the other hand, the spacecraft will be at a very low height with extreme speed. The resulting antenna tracking speed is very high and excludes most antennas.

Missions to the Moon, Mars, or further in space are called Deep Space Missions. For such missions the spacecraft do not orbit anymore around the Earth. Due to the long distances, the required antennas are larger in size to reach the required

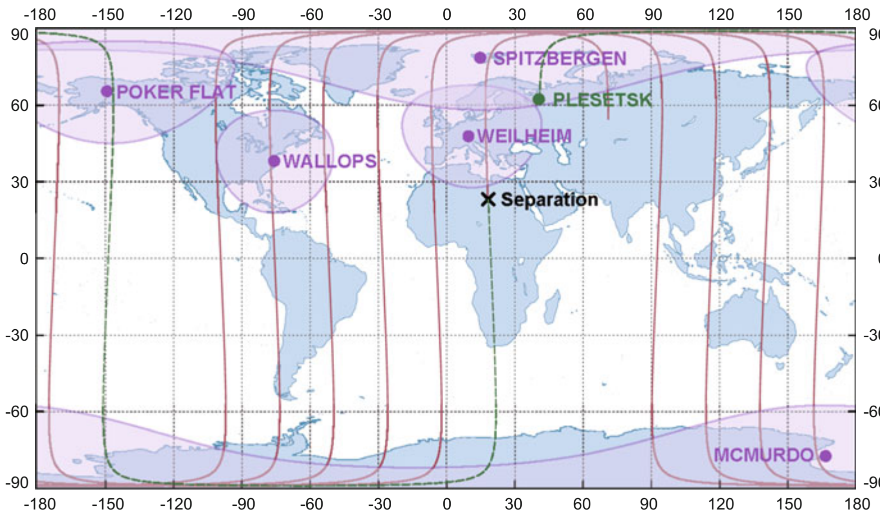

Fig. 3.9 Typical LEOP Ground Station Network for LEO Spacecraft

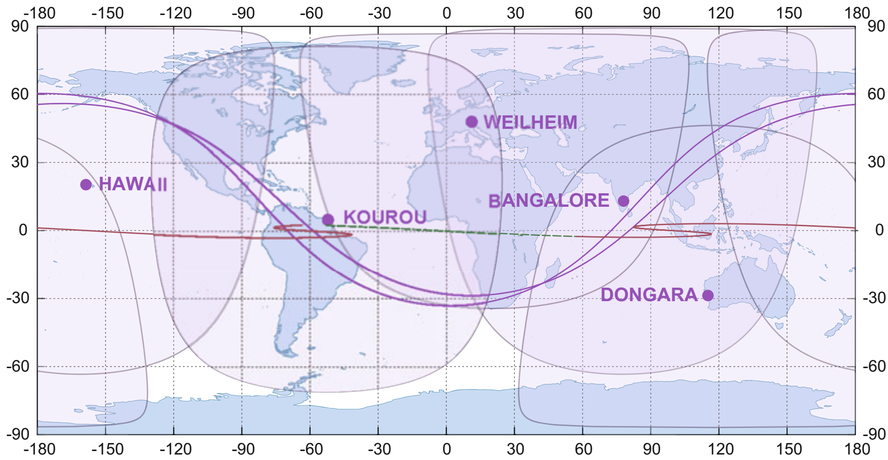

Fig. 3.10 Typical LEOP Ground Station Network for GEO Spacecraft in GTO

sensitivity. They can be built with a slower maximum tracking speed as the target motion is dominated by the earth rotation speed (Figs. 3.9 and 3.10).

Let's look shortly at other parameters which influence the choice of ground stations or antennas. We mentioned already the signal strength. Two main parameters which influence the space link quality are the transmitting power (Equivalent Isotropically Radiated Power—EIRP) of the spacecraft and the ground station, as well as the reception sensitivity (antenna gain, called also G/T—gain-to-noise

Table 3.1 Frequency band assignments as used in space operations

Band	Range (MHz)
L-Band	1,215–1,850
S-Band	2,025–2,400
C-Band	3,400–6,725
X-Band	7,025–8,500
Ku-Band	10,700–14,500
Ka-Band	18,000–35,000
V-Band	37,500–50,200

temperature). The space link quality (= link budget) calculation (see Sect. 1.3.3.2) shows how much margin is left in different conditions during the mission.

Other parameters which influence the choice of the stations are the downlink and uplink frequencies, in general called also frequency bands as shown in Table 3.1. Not all stations have antennas supporting all possible frequency bands; there is (at least up to now) specific specialization for stations, depending on their purpose. And so the stations supporting GEO missions typically have antennas with Ku- and Ka-Band capabilities, whereas LEO stations have S- and X-Band capabilities. Deep space antennas typically support also S- and X-Band frequencies, however with much larger diameter of the dish (to provide better EIRP and G/T).

They make the situation even more complex, during the LEOP of the geostationary satellites. S-Band is traditionally used before switching over to Ku- or Ka-Band for Payload-IOT (in-orbit test) and routine operations.

More and more there is an interest to support LEO satellites with Ka-Band due to higher bandwidth available. The trend is also apparent for LEOP support of GEO satellites as S-Band is under increasing pressure and interference from other spectrum users like mobile Internet access. The necessary ground station infrastructure is slowly being built up.

Another aspect of station selection is the bandwidth. Mostly ground stations support the fully available downlink data rate in the specific frequency band. For the uplink sometimes limitations apply. Up to now, relatively low data rates (between 4 and 20 kbps—kilobits per second) have been used for uplink. The capabilities and equipment of existing stations were designed accordingly. A tendency to increased uplink data rates can be observed to satisfy trends like more frequent software uploads. Not all ground stations can support this.

Other parameters, which we will not deliberate here too much, just to mention are modulation type, coding, randomization, space link data format, and finally specific tracking requirements—for ranging and Doppler. All of these things are luckily standardized, and the respective CCSDS (Consultative Committee for Space Data Systems) and ECSS (European Cooperation for Space Standardization) standards shown in Table 3.2 describe it in detail.

The technical compatibility between spacecraft and the station is essential; thus one should not rely only on standards. Before the launch of the spacecraft, a Radio-Frequency Compatibility Test (often referred to as RF-Comptest) is usually arranged. This test assures the radio interface compatibility and is used to prepare the station configuration for use during LEOP. The RF-Comptest is performed

Table 3.2 The most important CCSDS and ECSS standards for space mission communication. They are available from the Web sites of the organizations

Document	Title
CCSDS 401.0-B	Radio frequency and modulation systems—Part 1 Earth stations and spacecraft
CCSDS 132.0-B-1	TM space data link protocol
CCSDS 131.0-B-2	TM synchronization and channel coding
CCSDS 232.0-B-2	TC space data link protocol
CCSDS 231.0-B-2	TC synchronization and channel coding
CCSDS 232.1-B-2	Communications operation procedure-1
CCSDS 301.0-B-4	Time code formats
CCSDS 320.0-B-5	Ccsds global spacecraft identification field: code assignment control procedures
CCSDS 910.4-B-2	Cross support reference model—Part 1: Space link extension services
CCSDS 911.1-B-3	Space link extension—Return all frames service specification
CCSDS 911.2-B-2	Space link extension—Return channel frames service specification
CCSDS 912.1-B-3	Space link extension—Forward Cltu service specification
CCSDS 133.0-B-1	Space packet protocol
ECSS-E-70-41A	Ground systems and operations—Telemetry and telecommand packet utilization
ECSS-E-HB-50A	Communications guidelines
ECSS-E-ST-50-01C	Space data links—Telemetry synchronization and channel coding
ECSS-E-ST-50-02C	Ranging and Doppler tracking
ECSS-E-ST-50-03C	Space data links—Telemetry transfer frame protocol
ECSS-E-ST-50-04C	Space data links—Telecommand protocols synchronization and channel coding
ECSS-E-ST-50-05C	Radio frequency and modulation
ECSS-E-ST-50C	Communications

typically at the latest six months before launch, as soon as the so-called "RF-Suitcase" is available. The RF-Suitcase typically contains the Flight Model of the RF equipment with some parts of the On-Board Computer (OBC). In some cases even the whole spacecraft is transported to the ground station to perform testing. In the latter case a clean room needs to be available, however. In most cases only the relatively small RF-Suitcase is provided.

There are a few more aspects one should not forget, when designing the GSN. The accuracy of the carrier separation between a spacecraft's multiple antennas or multiple spacecraft is essential, as interferences may occur, rendering one or both space links unusable. This may lead to exclusion of up- or downlink over specific geographic regions or during specific times.

Another thing is the spacecraft autonomy, depending on it we may use less stations on ground, being sure the satellite may survive several hours on its own in case of some outage in the ground station. On the other hand, in case of low autonomy or critical application (e.g., precise orbit keeping), the requirements on GSN increase dramatically. Finally onboard constraints may require more frequent contact times (e.g., limited onboard data storage).

3.2.2 *Station Communication*

The selected ground stations are connected to the control center with a communication network. Different levels of network communication characteristics have to be considered and used. The choices we make will be based on some baseline requirements like bandwidth needed for supporting the mission, availability at specific locations, and total cost over the mission lifetime.

3.2.2.1 Communication Paths

There are several ways to connect to ground stations.

Most often leased lines are used. The backbone technology of these lines is in the hands of the telecom provider, and the control center may or may not have some disadvantages due to that. But even though we cannot influence the selection, we need to know it, to judge the quality of the link. Keywords here are SDH (Synchronous Digital Hierarchy), ATM (Asynchronous Transfer Mode), or MPLS (Multi-Protocol Label Switching) (Fig. 3.11).

Another technology, which is still operationally meaningful, is ISDN (Integrated Services Digital Network). It does not provide too much bandwidth (64 kbps per line), but is very reliable and provides the option of dial-up connection. This is analogous to the phone system at home; you pay as you use it. This may be especially interesting for small missions with limited budgets. Unfortunately the ISDN technology is getting older now, and many telecom operators are phasing it out, because it requires a separated infrastructure, is utilized less frequently, and provides relatively low flexibility in comparison to modern IP networks.

A specific variant of communication is the so-called roof-to-roof communication, which is typically realized by VSAT (very small aperture terminal). This is nothing else than satellite dishes installed at the premises of the MCC and its partner center or station, communicating over a geostationary communication satellite. The advantage is an extremely high flexibility (you can virtually connect your MCC with every point on the Earth) at decent bandwidths. The solution is, however, typically pretty expensive (rental of the GEO transponder) and considerable delays to your data streaming are being introduced.

Finally, let's say some words about the Virtual Private Network (VPN) over Internet. This option may seem very attractive, as the Internet "costs nothing," virtually provides "unlimited bandwidth," and "is everywhere." However, one should consider if the disadvantages are not too large. It may not cost much, but you actually share it with unspecified number of other users, and there is no "support for Internet" available. The bandwidth varies permanently and not necessarily you have really good access from everywhere. Many operational systems are not allowed to be attached to a network that has Internet access. In general, it is suggested not to use the Internet as a transport technology for real-time TM/TC or other critical applications, where security and data integrity play a

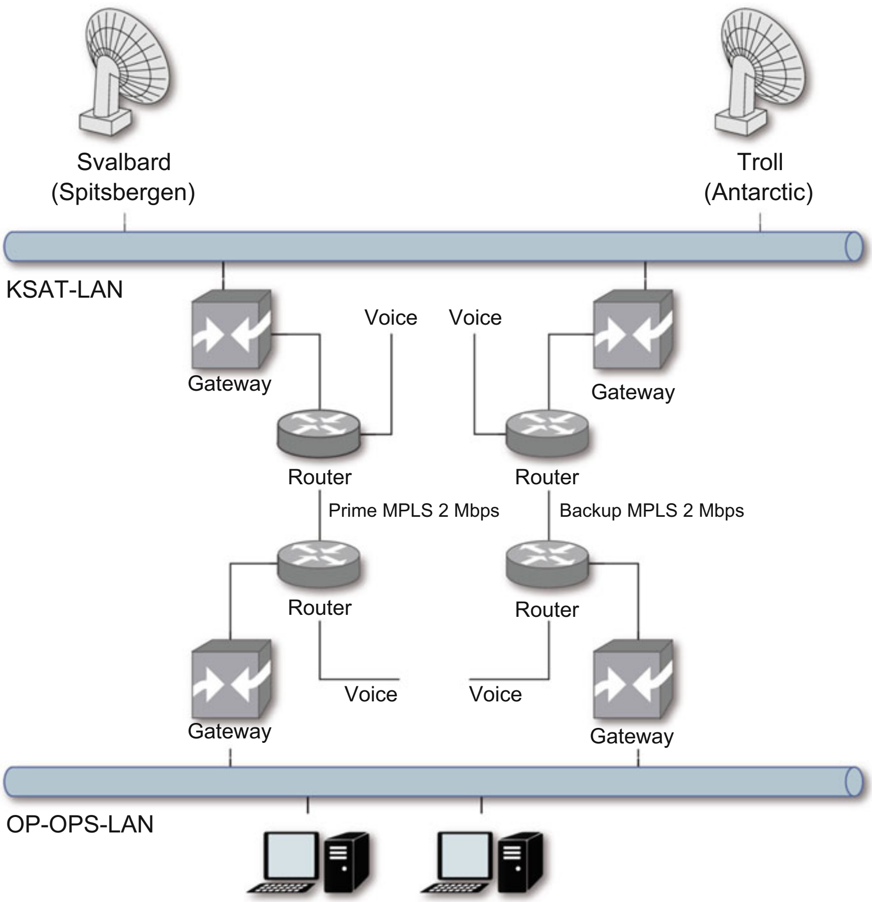

Fig. 3.11 Example of redundant ground station connection

role. Still it is a viable solution for off-line data transfers and for connection between MCC and the manufacturer site for simulation and testing with Central Checkout System (CCS).

3.2.2.2 Data Transfer Methods

Consideration is required for the data types which shall be exchanged over the communication lines. For telemetry and telecommand as well as for voice interfaces, good, reliable real-time connection is needed but not necessarily with high bandwidth, whereas management information, scheduling, tracking, pointing may be organized in a more cost friendly manner. Nowadays practically all traffic is realized based on TCP/IP protocol, whereas in past some proprietary transport protocols were used (DECnet, X25, NASCOM).

At the level of application, the CCSDS Space Link Extension (SLE) standard is extensively used for real-time communication, whereas the FTP is the main file transfer mechanism. In the future it is expected—especially for file transfer and management information exchange—some changes, and respective standardization work is being performed right now. More online services as well as mission operation services will be made available.

Addressing operational aspects of the GSN, one should keep in mind that all of the connections need to be tested prior to LEOP, and so respectively the arrangement and integration of communication lines and ground stations need to be performed at the right time. The personnel need to be trained to operate the network, and procedures for different operational scenarios need to be prepared and validated.

3.2.2.3 GSN Examples

As we know now all aspects of the GSN, we may look at the example presented in Fig. 3.12.

This is an example of the GSN and communication infrastructure for the LEOP of a geostationary satellite. At the bottom, the MCC (in this case GSOC) is located. Solid lines constitute voice communication, whereas dotted ones the data connections (in terms of real time TM/TC).

The MCC has a voice connection either over voice system or telephone with the launch site, ground station Weilheim, and the respective Network Management Centers of external partners (PrioraNet, CNES and ISRO) (Fig. 3.13). Data links are implemented fully redundant using different routes. Weilheim is integrated with two SDH 2 Mbps links, whereas connection to PrioraNet stations is available over two levels of NMCs with VSAT terminal and ISDN. Similarly the CNES and ISRO networks have been integrated with a single NMC each.

One can clearly see that the partnerships with external partners and agencies allow extending the GSN resulting in a sophisticated spacecraft support network, raising the total reliability to very high levels. On the other hand, it results in a complex network, which needs to be controlled, planned, and maintained. Not to be neglected are contractual and financial aspects, with questions like: do we always get the highest priority at the specific station? How much does it cost in long term? Is it possible to get some discounts if we consolidate our requirements to only one provider? What would be the trade-offs if resigning of some specific station?

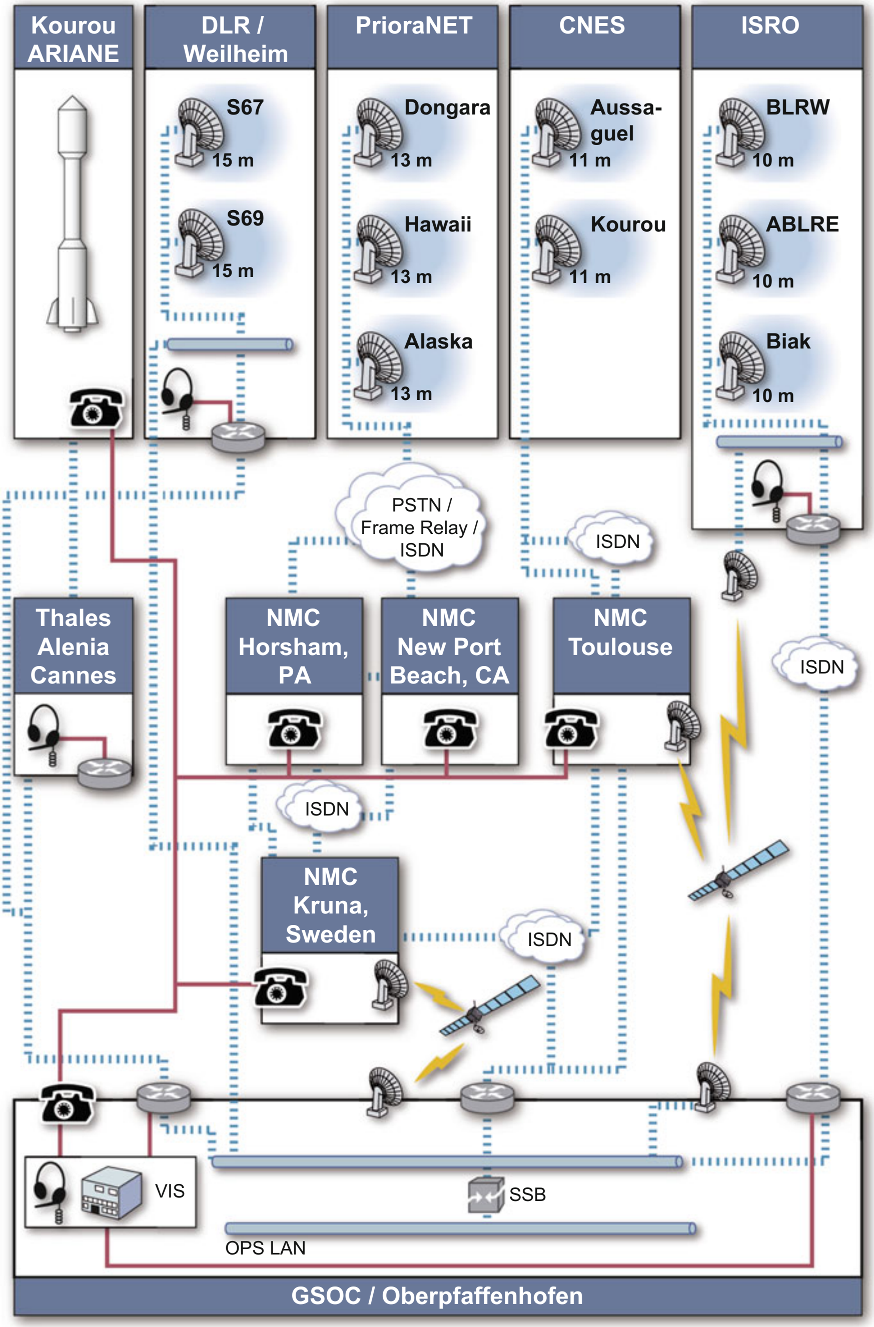

Fig. 3.12 Communication within GSN for the LEOP of a typical GEO mission

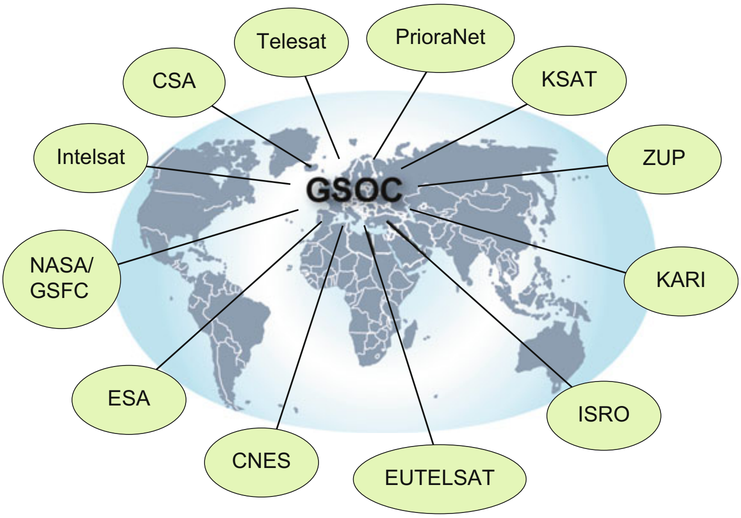

Fig. 3.13 International GSN cooperation from the perspective of an MCC-like GSOC

3.2.3 LEOP and Routine Operations

Within this chapter, we will look a bit more at the operational aspects of the control center infrastructure, network, and GSN. The main operational work is divided equally between Ground Data System, network, and hardware groups, whereas GDS performs most of coordination work as well as GSN operations.

The GDS team acts as an interface between the satellite operations teams and the other communication and ground station support groups. It manages the GSN for the satellite missions, and is responsible for the interfaces with external partners supporting the missions. In other words the GDS manages operational internal and external interfaces of MCC.

The GDS accompanies each satellite project within MCC from the very early phases (mission studies, Phase A) until the very end (Phase F, decommissioning). Within the projects, the GDS is one of the project's subsystems, where it works on fulfilling the GDS part of the project requirements, and acts as an expert for GSN and control center infrastructure for the whole MCC for the missions. This construction allows project managers to bundle all communication and infrastructure questions and requests to one person (the designated project responsible from GDS). That person takes care of distribution and coordination within Communication and Ground Stations department or with external partners. This allows high synergy, a high reuse of resources, and an optimal work distribution.

Other tasks and areas of responsibility of GDS are listed below:

- Participate in meetings and project reviews (PDR, CDR, TAR, etc...)
- Take responsibility for the project's Ground Station Network

- Perform management of interfaces to the external partners, including contractual and technical agreements
- Providing first level expertise for all network, communications, and infrastructure questions and issues
- Coordination of all operations related activities within Communication and Ground Stations department
- Preparation of work packages, work package description
- Preparation of cost calculations related to communication and infrastructure
- Preparation of relevant project documentation [requirements, design, test plans, reports, ICDs (Interface Control Documents), DMRs (Detailed Mission Requirements)]
- Assessment and implementation of new operational solutions for communications and infrastructure
- Participation in international standardization organizations with operational communication topics (CCSDS, IOAG)

In the example of GSOC, the GDS group includes three specific subgroups, which have their own tasks.

3.2.3.1 GDS Engineering Team (NOPE)

The GDS NOPE (Network Operations Engineer) team is a group of engineers taking care of the technical and organizational tasks related to specific satellite missions. Typically each mission has a designated GDS engineer who accompanies the project from phase C (cf. Sect. 2.1), participates in project meetings, and plays the role of contact person in the absence of the GDS manager. The most important tasks of the GDS engineers are:

- Mission preparation
- LEOP (Launch and Early Orbit Phase) preparation (configurations, coordination)
- Active mission support (NOPE) during LEOP (also on shift)
- Performing of tests (Data Flow Tests (DFT), connection tests, configuration tests)
- Preparation of the configuration for all data connections for the mission
- Configuration coordination with external partners
- Preparation of reports
- Troubleshooting and failure analysis

3.2.3.2 Systems (Network and Systems Control)

The Network and Systems Control (sometimes also called Network Management Center) is—at least from the communications point of view—in charge of MCC operations. The team consists of a number of operators, who work on shift to cover

24/7 operations, and support engineers who coordinate the shift team and manage the work and operational processes within the Network Control Room (Systems Room). “Systems” can be compared to a central phone switch board, where all connections (operational and technical) from all MCC control rooms are routed (switched) to the ground stations worldwide. Systems also plays the role of a voice center, as it is permanently staffed and has contact (either via telephone or a special voice system) with all projects and all stations, allowing quick reaction in case of contingencies or emergencies. This function is for example used to coordinate extraordinary contract requests on holidays or at night, when the scheduling office is not working.

3.2.3.3 “Systems” (Network and Systems Control Team)

- Network control during routine operations (establishing of connections on project request and along the schedule)
- Support for NOPE during LEOP
- Monitoring of the connections and network within GSOC and to external partners
- Support of contingency and emergency scheduling and operations

3.2.3.4 Scheduling Office

A dedicated scheduling office is a functionality which becomes necessary with increasing numbers of missions and available antennas of a control center. Coordination between these two elements becomes essential to avoid conflicts and to increase synergy between missions. The scheduling office tasks can be performed by one person and needs to be operated only during office hours. The tasks of scheduling are:

- Receive ground station support requests from projects and coordinate allocation at the organization’s own and at external ground stations
- Perform contact planning according to mission requirements applying mission priority rules
- Publish the weekly contact plan (schedule) for all MCC missions and resources
- Provide support and propose solutions in case of conflicts

When we look at the operations work aligned to the mission lifetime, most of the work may be divided into three phases: preliminary preparation (design), detailed preparation (design), and mission execution, which contains specific events like LEOP. Interestingly, most of these tasks may be equally used for all systems and subsystems, as they perform them more or less in the same way, just at different levels.

In the preliminary preparation phase the main work focuses mainly on the analysis of the customer requirements. This consists of checking if the existing system fulfills the requirements or if changes or upgrades have to be considered. This is important, because it will drive the costs. Based on that, detailed requirements for the subsystem are defined.

Another task is to prepare the general design (concept), which includes interface specification. This part is continued in the detailed design definition phase, where also test and verification plans need to be created.

The implementation phase is typically very busy for everybody in a space mission project, and it is not different for the Control Center infrastructure, network, GDS, and GSN. In detail it is necessary to implement and integrate all subsystems, which encompasses hardware, software, and service procurement (like communication lines), installation, and testing. Sometimes hard- or software has to be delivered to cooperating partners, which means that necessary export licenses have to be issued on time.

Aside from that, the frequencies have to be coordinated and licensed. This is typically done at national level for ground stations. The spacecraft owner needs to apply at the International Telecommunication Union (ITU) for the allocation of the communication frequencies.

Also a specification for external partners needs to be prepared and issued in form of Detailed Mission Requirements (DMR). This, however, may be done only as soon as respective contracts with these partners are in place. So, as one can see, there is a lot of paper work which needs to be taken care of in advance.

Finally in that phase, the complex set of technical and operational tests and validations is performed. They are based on previously prepared test plans, and include all subsystems, from data processing, through communication, and ending with end to end tests (including all components) and simulations. Especially the latter ones are important for validation of previously prepared operational procedures (e.g., emergency procedures). Not to be neglected is staff planning and training, both on technical and operational side.

The LEOP marks the border between preparation phases and the routine operations. All systems need to be handed over to the operational team before LEOP, typically performed formally during the Operational Readiness Review (ORR).

During operations (including LEOP) there are a number of tasks performed repeatedly, like scheduling of ground contacts, preparations and execution of passes, reporting, and accounting. At the same time, the whole GSN and MCC infrastructure needs to be monitored and controlled; maintenance needs to be performed. The interfaces to external partners, to all ground stations, and of course the internal interfaces need to be handled. In case of any anomalies and failures, the actions need to be performed according to procedures, the error reports need to be generated, and any anomaly and failure shall be tracked with respective Discrepancy Report, to avoid such cases in future.

Chapter 4
Flight Dynamic Operations

4.1 Orbital Dynamics

Michael Kirschner

4.1.1 Introduction

This chapter talks about the orbital aspects of flight dynamics (FD) and is divided into two main sections. In the first one, theoretical aspects are addressed in Sect. 4.1.2, where an overview is given about the description of a satellite orbit, the orbit perturbations, orbit maneuvers, and orbit maintenance. These topics are important for the understanding and successful performing of FD operations, which is presented in the second main Sect. 4.1.3 flight dynamics tasks.

It is important to note in reading this chapter that only resulting formulas of satellite orbit theories are presented and not their derivations. For those, who are interested in such details, books about the orbit dynamics of a satellite should serve as reference like “Satellite Orbits” (Montenbruck and Gill 2000) or “Fundamentals of Astrodynamics and Applications” (Vallado 2001).

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4.1.2 Theoretical Aspects

4.1.2.1 Description of a Satellite Orbit

Each point or position in space is clearly identified by three parameters like the Cartesian coordinates x, y, and z. If we use a coordinate system, a point or position can be expressed by the radius vector $\boldsymbol{r}$ with its components r_x, r_y, and r_z.

An object in space close to a central body is moving all the time, meaning it is changing its position continuously. Therefore we can describe an orbit as a sequence of positions, where the positions are time depending. Consequently we need more than three parameters for the description of an orbit. In total six parameters fully define an orbit. As the position is changing, the time derivative of the position vector being the velocity vector can be used with its components v_x, v_y, and v_z. These six parameters are also called state vector.

For the visualization of an orbit, another set of six orbital parameters is commonly used, the so-called Keplerian elements. In the following this set of elements is explained addressing the geometry of an orbit and its orientation in space. It has to be noted that only closed-loop orbits are addressed.

Geometry

The general form of a closed-loop orbit is an ellipse, which has two axes, the major and the minor one. Half of the major one is our 1st parameter and is called semi-major axis a (c.f. Fig. 4.1). The cut of both axes (major and minor ones) is the geometric center of the ellipse. But it is very important to know that the gravitational body is not located in this center, it is located in one of the two focal points of the ellipse, the location of the Earth Center EC (c.f. Fig. 4.1).

Therefore, an elliptical orbit has two special points, the apogee (A) and the perigee (P), where the distance is a maximum w.r.t. the EC or a minimum, respectively.

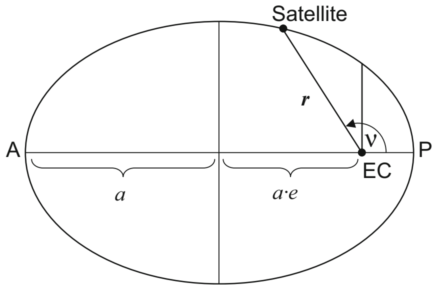
The second parameter is the numerical eccentricity e, which is a measure for the slimness of the ellipse. The smaller it is, the more circular it looks like, and the larger it is, the smaller the relation is between the minor and major axis is. The product of a and e determines the distance between the focal point and the center of the ellipse. For closed-loop orbits it has a range of

$$0 \leq e < 1.$$

In the special case $e = 0$, the ellipse is changed to a circle, where the two focal points are now identical with the center of the circle.

The third parameter is the true anomaly v, which is the angle between the radius vectors to the perigee and the location of the satellite.

Fig. 4.1 Ellipse geometry



Orientation in Space

The remaining three parameters are described with a look to the orientation of an orbit in space. We have to use a coordinate system, which is inertial fixed and where the origin is identical with EC. The z-axis points to the North (N) and is identical with the rotational axis of the Earth. The other two axis x and y are perpendicular to the z-axis and define the equatorial plane. The x-axis points to the so-called Vernal equinox (♈), which is defined as the direction from EC to the center of the Sun at the time (March 20), where the Sun crosses the equatorial plane from the South to the North. The x-axis is therefore identical with the cut of the equatorial plane and the ecliptic.

The point, where the satellite crosses the equatorial plane from the South to the North, is called Ascending Node (A.N.), the opposite one Descending Node, and the connection of both points is called nodal line. The ascending node is used to define the right ascension of the ascending node Ω (RAAN), which is the angle between the direction of the ascending node and the x-axis. At the ascending node, the inclination i can be defined, which is the angle between the orbital and the equatorial planes. The last of the six parameters is the argument of perigee ω defined as the angle between the directions to A.N. and the perigee (P). For all definitions please refer also to Fig. 4.2.

We have now found a set of six orbital parameters, which are commonly used as Keplerian Elements. They are summarized in Fig. 4.3 and divided into three classes. The first class defines the shape of the orbit (semi-major axis and eccentricity), the second class the orientation in space (inclination, RAAN, and argument of perigee), and the third class the position inside the orbit measured from the perigee (true or mean anomaly).

This set of orbital elements is convenient for the visualization of an orbit. But for orbit propagation, this set has some disadvantages. Singularities have to be handled in two cases. If we have a circular orbit, there is no apogee or perigee. Therefore the argument of perigee is not defined. In the case, where the orbit is inside the equatorial plane like for GEO (*Ge*ostationary E*a*rth Orbit) missions, we have no node crossing, and therefore the RAAN is not defined. Due to these limitations, another set of elements is used for the numerical orbit propagation, which is free of singularities, the so-called state vector. It contains the position and the velocity vectors, in total six parameters.

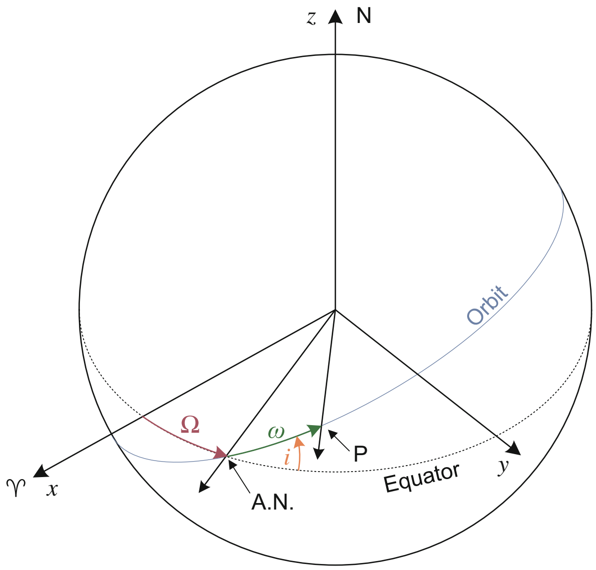


Fig. 4.2 Orientation of an orbit in space and definition of Keplerian elements

Class	Parameter	Name
Shape	a	semi-major axis
	e	eccentricity
Orientation in space	i	inclination
	Ω	right ascension of the ascending node
	ω	argument of perigee
Location	ν/M	true/mean anomaly

Fig. 4.3 Keplerian elements classification

4.1.2.2 Satellite Velocity

One of the most important parameters used in many equations is the velocity of a satellite. The vis viva theorem or energy law can be used to derive the equation for the satellite velocity. For closed-loop orbits, the energy equation

$$E_{\text{tot}} = E_{\text{kin}} + E_{\text{pot}}$$

can be derived like in the following:

$$-\frac{1}{2}\frac{\mu}{a} = \frac{1}{2}v^2 - \frac{\mu}{r}, \tag{4.1}$$

where μ is the gravitational parameter of the Earth ($\mu = \text{GM}_{\oplus} = 398{,}600.4415\ \text{km}^3/\text{s}^2$), a the semi-major axis, v the velocity, and r the radius to the satellite location.

Elliptic Orbit

Equation (4.1) can be converted to

$$v_{\mathrm{ell}} = \sqrt{\mu\left(\frac{2}{r} - \frac{1}{a}\right)} \tag{4.2}$$

which describes the velocity on an elliptical orbit. For example, around the perigee of the high-elliptic GTO (*G*eostationary-*T*ransfer *O*rbit) in an altitude of about 250 km the speed is about 10 km/s, and in the apogee in the geostationary altitude of about 36,000 km the speed is about 1.6 km/s.

Circular Orbit

The equation for the velocity on a circular orbit can be easily derived from Eq. (4.2), because a is equal to r

$$v_{\mathrm{circ}} = \sqrt{\frac{\mu}{a}}. \tag{4.3}$$

Examples: For LEO (low Earth orbits) in a few hundred kilometers altitude the velocity is about 7.5 km/s, in the altitude of the geostationary orbit (GEO) the velocity is about 3 km/s, and in the distance of the Moon (about 380,000 km) the speed is about 1 km/s.

First Cosmic Velocity

The first cosmic velocity is more an academic parameter, as it describes the circular velocity exactly above ground:

$$v_{\mathrm{Cosmic1}} = \sqrt{\frac{\mu}{R_{\mathrm{E}}}} = 7.905\,\mathrm{km/s}, \tag{4.4}$$

where R_{E} is the Earth radius with a size of 6,378.137 km at the equator. Of course, the atmosphere had to be removed that a satellite can orbit the Earth without touching the ground. But this velocity has another important significance: it determines the absolute minimum performance of a launcher system, which is normally expressed in a velocity increment $\Delta v = v - v_0$, where v_0 is zero at the moment of launch. With respect to the energy law, we would change only the kinetic energy, as we stay at the launch altitude (R_{E}) all the time. But in reality, the rocket has to bring a satellite to higher altitudes (above 200 km) to stay in an orbit around the Earth due to the high air drag in lower altitudes, which would bring the satellite back to ground very quickly. Going to higher altitudes increases the potential energy. In addition, the rocket has also to act against the air drag during the ascent. Due to these two effects also called gravity and drag losses a launcher system needs a performance between 9.2 and 9.6 km/s to bring a satellite to LEO.

Second Cosmic Velocity

The total energy for an escape trajectory in the form of a parabola is zero. Using Eq. (4.1), the second cosmic velocity can be derived for an escape from ground:

$$v_{\mathrm{Cosmic2}} = \sqrt{\frac{2\mu}{R_{\mathrm{E}}}} = \sqrt{2}v_{\mathrm{Cosmic1}} = 11.180\,\mathrm{km/s}. \tag{4.5}$$

In the more general form

$$v_{\mathrm{escape}} = \sqrt{\frac{2\mu}{a}} = \sqrt{2}v_{\mathrm{circ}} \tag{4.6}$$

Eq. (4.6) describes the escape velocity from any circular orbit around the Earth.

4.1.2.3 Orbital Period

The time for one orbit revolution called orbital period T can be easily derived from a circular orbit.

In general we can use the well-known equation of the path, which depends on the velocity and time

$$s = v \cdot t.$$

For one revolution s will be $2\pi \cdot a$ and using Eq. (4.3) for v we get

$$T = 2\pi\sqrt{\frac{a^3}{\mu}}. \tag{4.7}$$

4.1.2.4 Orbit Perturbations

The orbit of a satellite is influenced by different sources of perturbations, which change the behavior of a pure Keplerian orbit. The main sources are

- The gravitational field of the Earth
- The air drag
- The solar radiation
- The influence of the gravity of Sun and Moon and
- Thruster activity

The first two items are addressed more in details in the following.

Earth Gravitational Field

The shape of the Earth is not an ideal sphere; it can be approximated by an ellipsoid of rotation, where the radius to the poles is about 20 km less compared to the radius in the equatorial plane. Due to this oblateness, the nodal line performs a rotation around the pole axis. This rotation can be described by the time derivative of the RAAN for circular orbits

$$\dot{\Omega} = -\frac{3}{2} J_2 \sqrt{\mu} \frac{R_E^2}{a^{3.5}} \cos i, \tag{4.8}$$

where the zonal parameter J_2 of the gravitational potential for our Earth describes the form of the ellipsoid. It can be described as a function of the difference between the equatorial and polar radii of the equipotential surface of the Earth's gravity field. More accurately expressed, J_2 is a function of the difference in principal moments of inertia (for more information, please refer to the reference). Its size is $1.083 \cdot 10^{-3}$. Applied to the orbit of the International Space Station (ISS) with an inclination of about 51° and an altitude between 300 and 400 km, the node drift would be about 3 degrees per day to the west.

We can use this effect of node drift to describe a very special orbit, the so-called Sun-synchronous orbit. The requirement for such an orbit is that the node drift has to be equal to the mean motion of the Earth around the Sun meaning the difference between the RAAN and the right ascension of the Sun is constant. As the Earth is moving eastwards around the Sun, which is defined with the positive sign, the mean node drift has to be +2π per year or +360° per year. Using Eq. (4.8) and the constant variable C, we can now write

$$\dot{\Omega} = -C \frac{\cos i}{a^{3.5}} = +\frac{2\pi}{\text{year}}, \tag{4.9}$$

The negative sign on the left side of Eq. (4.9) can be turned to plus (+), if the cos$i$ is negative. This is possible for inclinations larger than 90°.

Figure 4.4 shows all theoretical possibilities, where the altitude is plotted versus the inclination. In reality, most of the satellite missions flying on a Sun-synchronous orbit have altitudes between 500 and 1,000 km. The corresponding inclinations are between 97° and 100°.

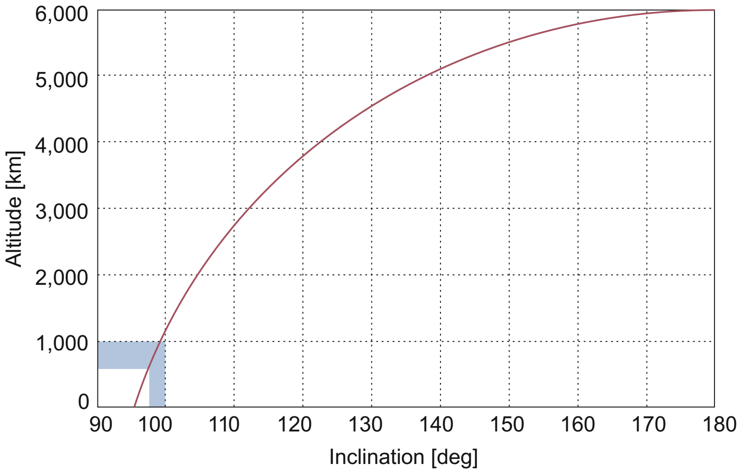


Fig. 4.4 Sun-synchronous orbits

The oblateness of the Earth has also an effect on the lines of apsides, which is the connection of perigee and apogee and therefore identical with the major axis of an ellipse. This line of apsides rotates inside the orbital plane and its drift can be approximated by

$$\dot{\omega} = -\frac{3}{4} J_2 \sqrt{\mu} \frac{R_{\mathrm{E}}^2}{a^{3.5} \cdot \left(1 - e^2\right)^2} \left(1 - 5\cos^2 i\right). \tag{4.10}$$

The drift vanishes, if the expression $(1-5\cos^2 i)$ is zero, which is for an inclination of 63.43°. This effect is used amongst others by the Russian Molniya satellites, which fly on a high-elliptic orbit with an apogee placed over Russia. Due to the low speed around the apogee, the contact to the satellite is guaranteed for a long time w.r.t. the revolution time.

Air Drag

Even if the atmosphere of the Earth is defined up to an altitude of 2,000 km, a clear effect can be seen for circular orbits with an altitude below 800 km and for high-elliptic orbits with a perigee altitude below 250 km.

In general, the air drag reduces the orbital energy due to heating up the environment, which results in a decrease of the semi-major axis [refer also to Eq. (4.1)]. In the case of the high elliptic orbit with the perigee inside the "dens" atmosphere, only the kinetic energy is changed with the consequence that only the altitude of the apogee is reduced (refer also to Fig. 4.5). Once the apogee is also inside the dense atmosphere and the orbit has been circularized, the orbital height and period is decreased due to the air drag, but the satellite velocity is increased.

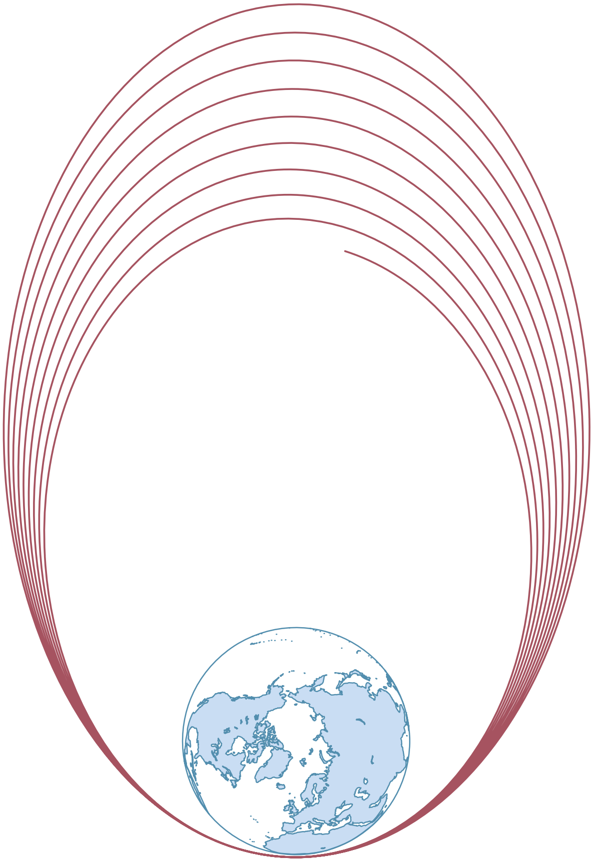


Fig. 4.5 Influence of air drag on the orbit

This looks like a paradox: the air drag force acts in the opposite direction of the velocity and the velocity is increased. We have to look to the energy law: the reduction of the potential energy is larger than the reduction of the total energy. Therefore the kinetic energy and the corresponding velocity have to be increased!

The rate of the decay depends on the density and the density itself on the solar activity, which has a cycle length between to maxima of about 11 years (refer also to Fig. 4.6). In general, the higher the sun activity is the higher the air density for the same altitude.

4.1.2.5 Maneuvers

If the orbit has to be changed and the natural perturbation forces are neglected, so-called orbit maneuvers have to be performed. A maneuver will always change the velocity vector only, its length and/or its orientation, and not the position at the time of maneuver execution. In general, two types of maneuvers can be defined,

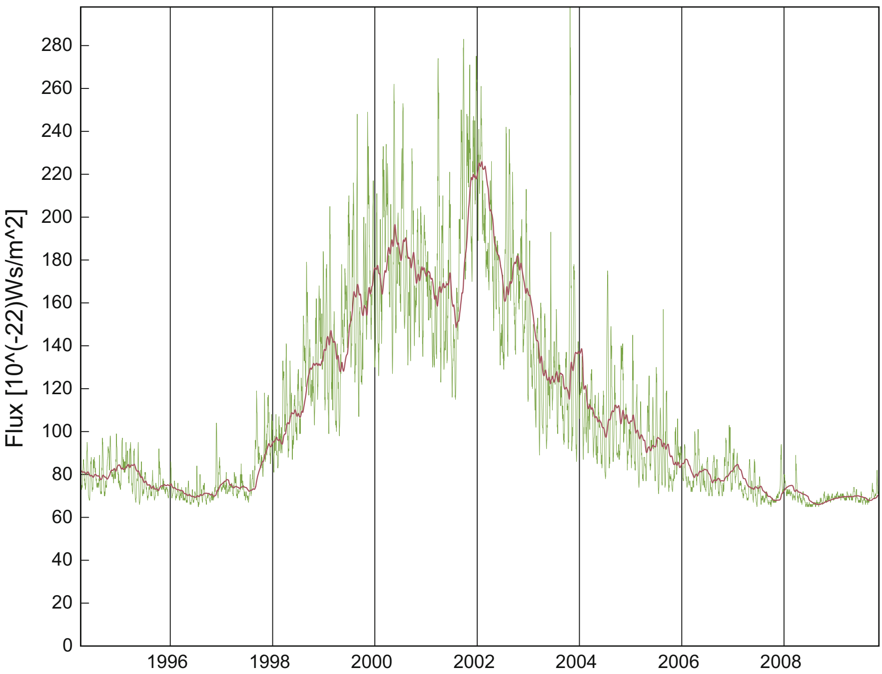


Fig. 4.6 The averaged observed and estimated profile of sunspot numbers

the in-plane and the out-of-plane maneuvers; the in-plane maneuver changes the shape of the orbit and the out-of-plane maneuver changes the orientation of an orbit in space.

Two examples are used to explain both types of orbit maneuvers.

In-Plane Maneuver

If an in-plane maneuver is performed on a circular orbit, the resulting orbit will be always an elliptical orbit due to the change in the shape. A consequence is that a perigee and apogee will be generated. If we consider only impulsive maneuvers or only small extended maneuvers, where the burn duration is much smaller than the orbital period, only the kinetic energy is changed at the location of the maneuver, but for all other points along the orbit the altitude is changed.

Usually we change the altitude of the perigee or apogee. In the following we want to lift the perigee of a GTO to GEO altitude (please refer also to Fig. 4.7). In this example the satellite was injected into the high-elliptical GTO with an apogee exactly in the required geostationary altitude like for an ARIANE launch. Once the satellite reaches the apogee, it has to increase its velocity. The reason is that the velocity in the apogee of an elliptical orbit inside the circular target orbit is always smaller compared to the required circular velocity.

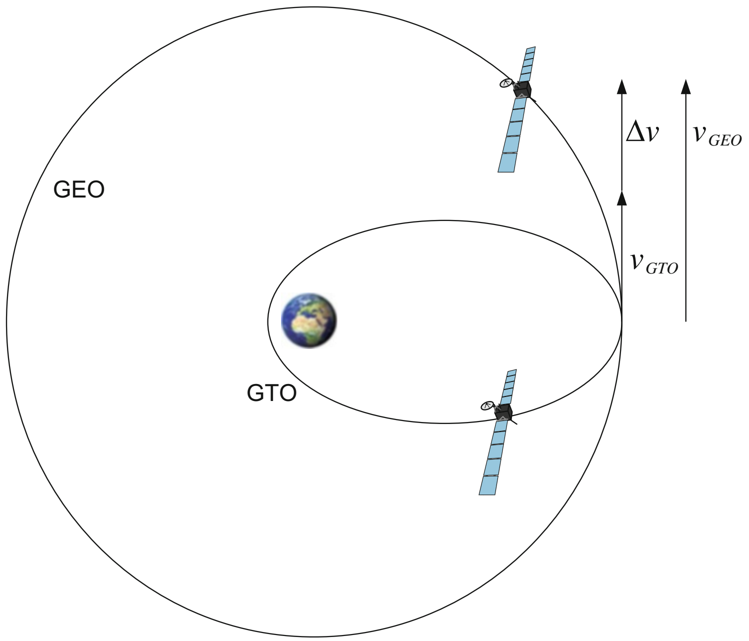


Fig. 4.7 An In-plane maneuver; lift of transfer orbit (GTO) perigee to geostationary (GEO) altitude

Using the following geometric parameters

r_{apo} = 42,164 km (GTO apogee radius)
a_{GTO} = 24,400 km (GTO semi-major axis)
a_{GEO} = 42,164 km (GEO semi-major axis)

and Eqs. (4.2) and (4.3), we can calculate for the velocity in the apogee $v_{\text{TO}} = 1.603$ km/s and for the required circular velocity $v_{\text{GEO}} = 3.075$ km/s. The necessary in-plane velocity increment is

$$\Delta v_{\text{IPL}} = v_{\text{GEO}} - v_{\text{GTO}} = 3.075 - 1.603 = 1.472 \ \text{km/s}.$$

Out-of-Plane Maneuver

This class of maneuvers changes only the orientation of an orbit. Consequently, the velocity vector is rotated only and its length is kept constantly. The out-of-plane velocity increment Δv_{OPL} can be calculated by the geometry of an isosceles triangle (refer also to Fig. 4.8)

$$\Delta v_{\text{OPL}} = 2 \cdot v \cdot \sin(\Delta i/2), \tag{4.11}$$

with $v = |v_1| = |v_2|$, which can be calculated with Eq. (4.2), and Δi is the required inclination change.

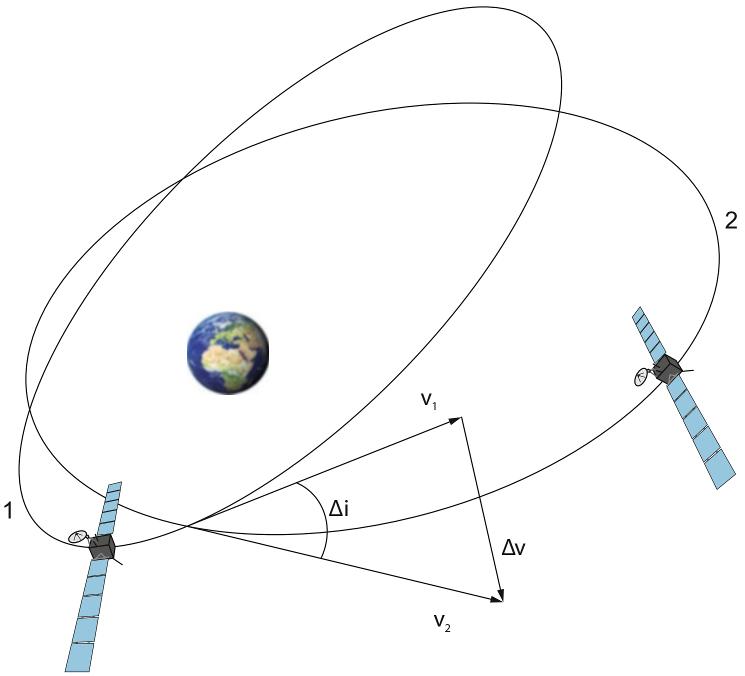


Fig. 4.8 Out-of-plane maneuver; change of the inclination

Performing maneuvers in space is always connected with optimization. The reason is that the fuel consumption is directly proportional to Δv. In the current case, the velocity along an elliptical orbit is different. The minimum velocity increment for a rotation of an orbit can be achieved in the apogee. Of course, on a circular orbit the velocity is constant and therefore the velocity increment cannot be minimized.

Using again the GTO of the previous chapter as example and due to the location of the launch site an initial inclination, e.g. 7°, has be balanced. Using the value of the apogee velocity of the previous example, we can calculate a velocity increment of

$$\Delta v_{\mathrm{OPL}} = 2 \cdot v \cdot \sin(\Delta i/2) = 2 \cdot 1.603 \cdot \sin(3.5^\circ) = 0.195 \ \mathrm{km/s}.$$

Combined Maneuvers

For a mission to GEO it is always necessary to lift the perigee to the geostationary altitude *and* to rotate the initially inclined injection orbit into the equatorial plane. The two previous chapters showed how we can calculate the necessary change in the velocity vector. In reality we will not inject the satellite by two subsequent maneuvers into GEO because of the vectorial nature of the maneuver (Fig. 4.9). The advantage of vector geometry is used to optimized the resulting Δv, as

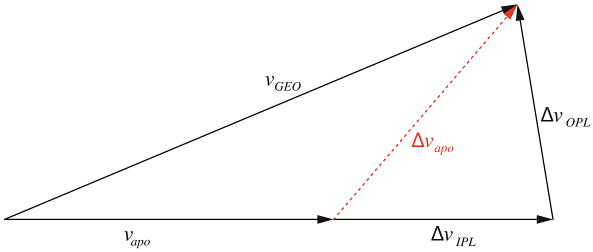


Fig. 4.9 Combined maneuvers

$$\left|\Delta v\right| = \left|\Delta v_{\text{IPL}}\right| + \left|\Delta v_{\text{OPL}}\right| \geq \left|\Delta v\right| = \left|\Delta v_{\text{IPL}} + \Delta v_{\text{OPL}}\right|.$$

Therefore both types of maneuvers are combined in the apogee. For the optimized maneuver a Δv of only 1.497 km/s is needed compared to a Δv of 1.667 km/s, which extends the mission lifetime by about 3 years, as about 1 m/s per week are needed for keeping the satellite inside its box.

4.1.2.6 Orbit Maintenance

Maneuvers are necessary to maintain an orbit of a satellite. As the requirements for orbit maintenance are very mission specific, the TerraSAR-X mission is used as example. Here a short list of general requirements is given, which had to be fulfilled for the TerraSAR-X mission:

1. Compensate the orbit decay
2. Keep the ground track stable
3. Keep the time relation of the orbit stable
4. Keep the orbit form stable
5. Achieve the mission target orbit

These general requirements were derived from mission requirements for TerraSAR-X like:

- Sun-synchronous orbit (1)
- Repeat cycle of 11 days (1, 2, 3)
- Frozen eccentricity (4)
- Control of the satellite along a reference orbit within a tube with a diameter of 500 m (1, 2, 3, 4)

A node crossing requirement of $|\Delta s| = \Delta\lambda \cdot R_{\text{E}}) \leq 200$ m could be derived to fulfill the overall and very challenging tube requirement, where $\Delta\lambda$ is the longitude difference of two subsequent ascending node crossings. When we start in the reference altitude, the air drag is responsible for the orbit decay, a decrease of the altitude. As a consequence the orbital period is a little bit smaller compared to the reference one and we reach the next node crossing a bit earlier. Therefore the longitude of the next node crossing is in the eastern direction w.r.t. reference

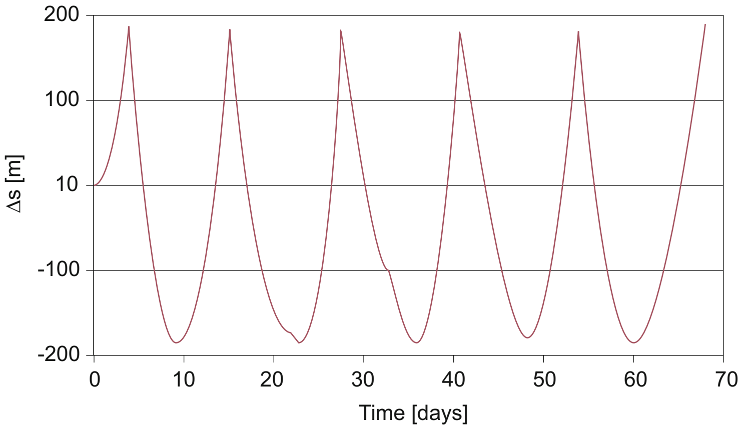


Fig. 4.10 A node crossing simulation of the first 70 days of the TerraSAR-X mission

node. We now let the satellite drift, until it reaches the upper limit of +200 m, where we have to perform an altitude correction maneuver. This maneuver overshoots the reference altitude in order to maximize the time between two orbit maintenance maneuvers. Due to the higher altitude the satellite has a higher orbital period and reaches the next node crossing a bit later, which is the opposite behavior compared to lower altitudes. The node crossing is now drifting to the west. We calculate the overshoot so that the reference altitude is reached at the western limit of −200 m w.r.t. the reference node crossing. This strategy was simulated for the first 70 days of the TerraSAR-X mission and is shown in Fig. 4.10. With this strategy the time between two orbit maintenance maneuvers could be maximized to about 2 weeks for the start of the mission.

As for each mission an estimation of the fuel consumption for orbit maintenance has to be done, a simulation of the whole mission lifetime was performed a few months before launch. Figure 4.11 shows the expected altitude changes at that time, which can be converted into Δv by the following equation derived from Eq. (4.3):

$$\Delta v_t = -\frac{1}{2} \cdot \Delta a \cdot \frac{v}{a}. \tag{4.12}$$

At the start of the mission the altitude had to be corrected by about 40 m, which had to be increased by a factor of 4 to the end of the mission correlated with orbit raise maneuvers each 1–2 days. The reason is that for the end of the mission a solar activity maximum had to be expected. But the amplitude of the Solar maximum was finally much smaller compared to the prediction in 2007, where the simulation was performed. A consequence is that fuel could be saved, which can be used to extend the mission lifetime of TerraSAR-X.

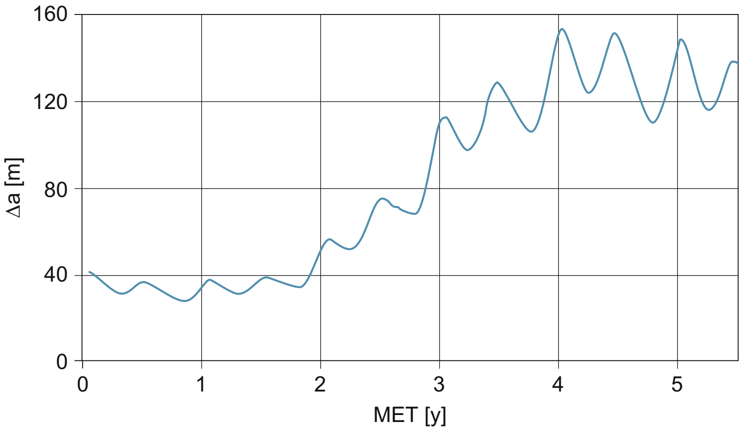


Fig. 4.11 Simulation of altitude change (Δa) maneuvers over mission elapsed time (MET) for TerraSAR-X

4.1.3 Flight Dynamics Tasks

The involvement of FD in a satellite mission begins years before launch with the mission preparation. In general, the FD team supports mission operations and the ground systems network during the preparation and, if the satellite is launched successfully, the execution of a satellite mission.

4.1.3.1 Mission Preparation

The main tasks before launch are the performance of a mission analysis comprising the orbit selection, a ground station visibility analysis, launch window calculation, first acquisition analysis, and maneuver strategies as well as the implementation of an operational flight dynamics system (FDS). The major topics of these tasks are addressed a bit more detailed in the following.

Orbit Selection and Ground Station Visibility Analysis

At the very beginning of a mission it has to be discussed with the customer, which orbit can fulfill the mission requirements. Once the orbit is selected, a ground stations network has to be searched, which can support the mission. In general, during the *L*aunch and *E*arly *O*rbit *P*hase (LEOP) and possibly during the commissioning phase more ground stations are needed compared to the routine phase. Which ground stations are usable depend on the launch site and the orbit type. As example Fig. 4.12 shows the ground track of an LEO satellite mission for the first day after injection. The satellite was injected close to the Hawaiian Islands into a

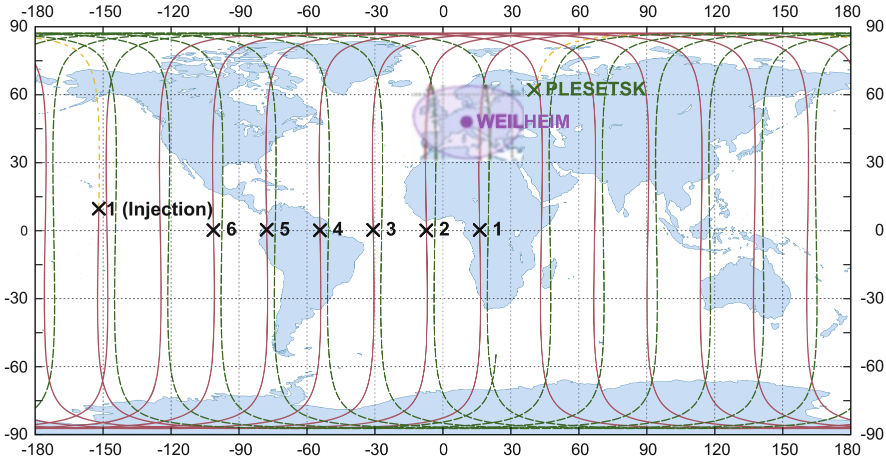


Fig. 4.12 Ground track of the first 24 h for a LEO mission (example CHAMP)

close-polar orbit with an altitude of about 450 km. The first contact over the Weilheim ground station, which has been used as the main antenna supporting all mission phases, occurred about half an orbit after the injection and another orbit revolution later the second one with a visibility time of about 8 min for each. It is typical for locations like the Weilheim one that due to the Earth rotation to the East the next contact possibility is about 10 h later, when the Earth has performed roughly half a revolution and the descending orbits come into the visibility area of Weilheim. This is one of the reasons why more than one ground stations are used during LEOP.

Figure 4.13 shows the additional ground stations from the NASA polar network, Spitzbergen, Pokerflat, and Wallops used for the CHAMP mission. A fourth one, McMurdo, is not visible in this figure, as it is located close to the South Pole. The first visibility after injection took place over the South Pole station McMurdo (MCMUR) and the second one was a relatively long and combined contact including Weilheim (WHM), Spitzbergen (SPZBG), and Pokerflat (PKF). This triple pass had a total visibility time of about 25 min and shows clearly the advantage of the geographical locations of the selected ground stations for the early phase of this mission. All theoretical visibilities of the involved ground stations are shown in Fig. 4.14, where the elevation is plotted over the time. At least each half orbit a contact is possible over the close-polar stations, but only the contacts over 10° elevation are used. In addition, it can be seen that a full coverage by ground stations is not possible for LEO missions.

Launch Window

Another important task is the calculation of the necessary launch window. Two aspects have to be considered, the launch date and time as well as the length of the

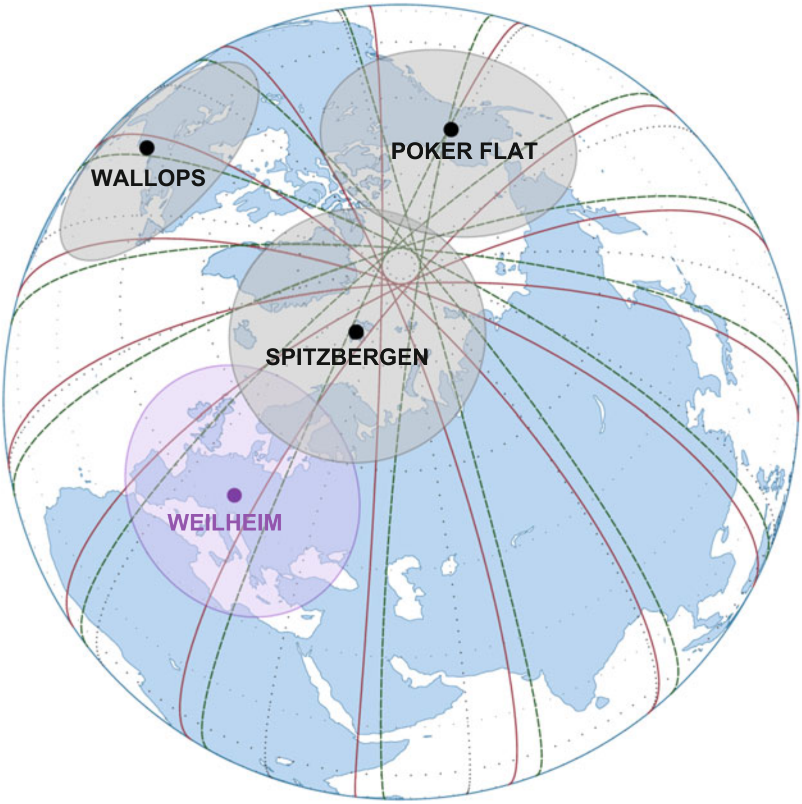


Fig. 4.13 Northern hemisphere with ground track of the first 24 h of an LEO mission (example CHAMP)

window. As both depend on the mission requirements, two examples are used to explain some of the constraints. In case of the TerraSAR-X mission, the satellite has to be operated along a reference orbit with a repeat time of 11 days and a local time of the ascending node of 18 h. In order to optimize the number of maneuvers and the fuel consumption, the injection time had to be exactly at the same time for each launch date and the length of the window was only a few seconds. For the injection into a GTO the launch window is completely different. The start of the launch window can vary within 1 h and the length of it can be between 1 and 2 h. The main constraints are the Sun location as well as the visibility of the ground stations during the apogee boost maneuvers. In the latter case, problems occurring during the countdown can possibly be solved during the launch window. This would not be possible in the first example.

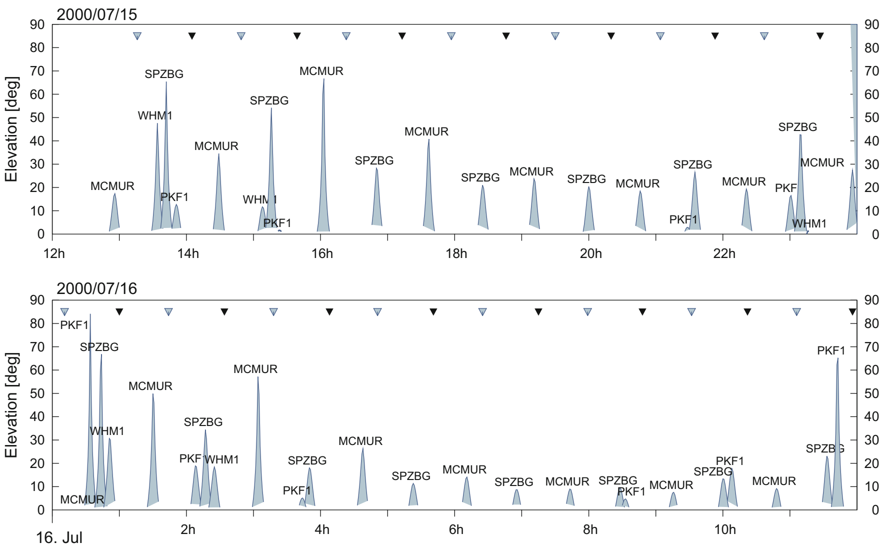


Fig. 4.14 Station visibility of the first 24 h of an LEO mission (example CHAMP)

a	i	MCMUR	WHM	SPZBG	PKF	MCMUR	WHM
0 km (N)	0° (N)	12:52:02	13:29:49	13:37:36	13:47:58	14:24:40	15:05:15
-10 km (L)	0° (N)	+2 s	−7 s	−9 s	−9 s	−18 s	−26 s
+10 km (H)	0° (N)	2 s	+7 s	+9 s	+9 s	+18 s	+27 s
0 km (N)	-0.02° (L)	+1 s	0 s	0 s	+1 s	0 s	−1 s
0 km (N)	+0.02° (H)	1 s	0 s	0 s	−1 s	0 s	+1 s

Fig. 4.15 Time offsets due to injection dispersion for an LEO mission (example CHAMP)

First Acquisition Analysis

During mission preparation, a so-called first acquisition analysis has to be performed for all missions. The reason is the possible injection dispersion of the launcher rocket. Even the launcher company releases nominal injection elements a few weeks before launch; dispersion values are also given mainly for the altitude and the inclination. In case of the CHAMP mission, the uncertainties were ±10 km for the altitude and ±0.02° for the inclination. Both values change the geometric conditions of a ground station contact, but in addition the first one influences the orbital period and therefore also the AOS and LOS times of a ground station contact. Two offsets can thus be derived, the time and the azimuth offsets at AOS. Both offsets are listed in Fig. 4.15 (time) and Fig. 4.16 (azimuth) for the first six contacts, where the second row shows the nominal (N) values for the AOS times and the azimuth. In rows 3–6 the offsets are shown for low (L) and high (H) dispersions for the semi-major axis (a) and the inclination (i), respectively.

a	i	MCMUR	WHM	SPZBG	PKF	MCMUR	WHM
N	N	70.71°	166.00°	177.17°	336.73°	31.20°	240.95°
L	N	+0.83°	−0.39°	−0.20°	−1.12°	+0.53°	+0.75°
H	N	−0.79°	+0.37°	+0.19°	+1.03°	−0.51°	−0.68°
N	L	0.22°	−0.11°	−0.18°	−0.29°	+0.18°	−0.19°
N	H	−0.23°	+0.11°	+0.18°	+0.28°	−0.19°	+0.18°

Fig. 4.16 Azimuth offsets due to injection dispersion for an LEO mission (example CHAMP)

Some of the cells are highlighted indicating possible problems. In general, a ground station will point to the nominal azimuth before the nominal AOS time. In the case it receives no signal at the nominal AOS time, it will wait a maximum time of 10 s, until it starts to search the satellite. Besides the time offset due to an altitude dispersion, the maximum azimuth offset, where a signal can be received, is defined by the beam width of the antenna. In the current example, a beam width of $\pm 0.3°$ is used for an acquisition with an S-Band antenna. If the azimuth offset is larger than the beam width, the ground station will never be able to contact the satellite. The ground station will wait the given time offset, before it starts to search the satellite. Consequently it is important to provide the ground stations with both offset tables that they know, what they have to expect and when they have to start the search.

Implementation of an Operational Flight Dynamics System

In the case, mission dedicated software has to be developed or existing software has to be modified depending on the analysis of specific FD mission requirements; this software has to be tested and validated. Together with the software packages in the FD multi-mission environment, which have to be used to fulfill all required tasks for the mission, the whole FDS has to be tested and validated together with all involved subsystems of the satellite control center, especially the interfaces as well as the in- and output data of the FDS.

4.1.3.2 Mission Execution

Once the satellite has been launched and separated from the upper stage successfully, the mission execution phase with FD operations begins comprising tasks like orbit determination including Δv estimation, orbit prediction, generation and delivery of orbit-related products, maneuver planning if thrusters are on board, monitoring and handling of critical conjunction events with other space objects, as well as lifetime and reentry predictions. Some of these tasks are addressed more detailed in the following.

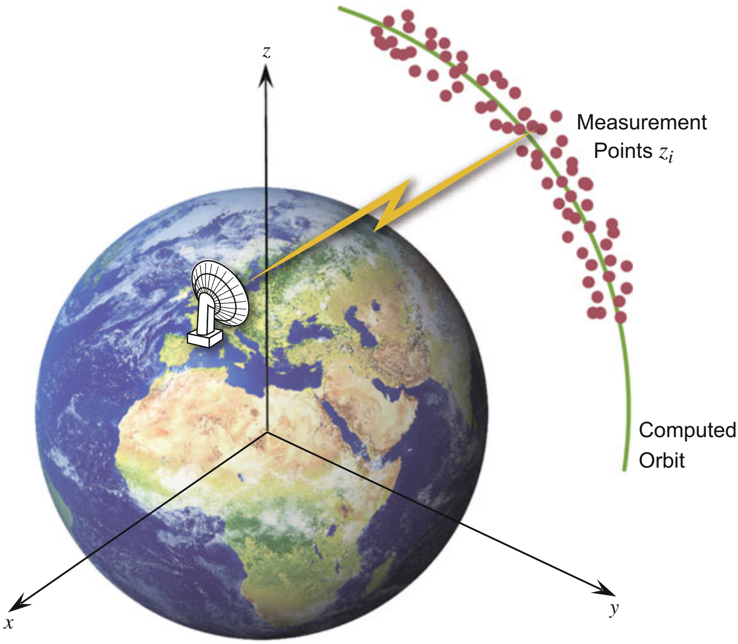


Fig. 4.17 Cloud of measurement points around the real orbit

Orbit Determination and Prediction

After the separation of the satellite from the upper stage it is very important to refine the nominal injection elements due to possible injection dispersions (please refer also to section "First Acquisition Analysis"). If the ground stations are able to lock to the signal of the satellite, they can generate tracking data, which are used in an orbit determination (OD) software to calculate a new set of orbital elements. As described in Sect. 4.1.2.1, a satellite orbit is clearly defined by six parameters and therefore a set of six equations have to be solved. Theoretically six measured parameters would be enough to solve the equations. But in reality much more are needed, as the measurements are not exactly located on the real orbit, they describe more or less a cloud around the orbit (see also Fig. 4.17).

Measurement data generated by ground stations are called tracking data (see also Fig. 4.18) and can be angle data (azimuth and elevation), range (distance between spacecraft and antenna), and Doppler data (relative velocity of the spacecraft towards the ground station). A fourth measurement type is the GPS navigation data, which is generated and stored on board of a satellite and dumped to the control center via a ground station contact.

An orbit determination software searches an orbit characterized by r and v, which minimizes the sum of the differences between the measured and calculated values according to Eq. (4.13).

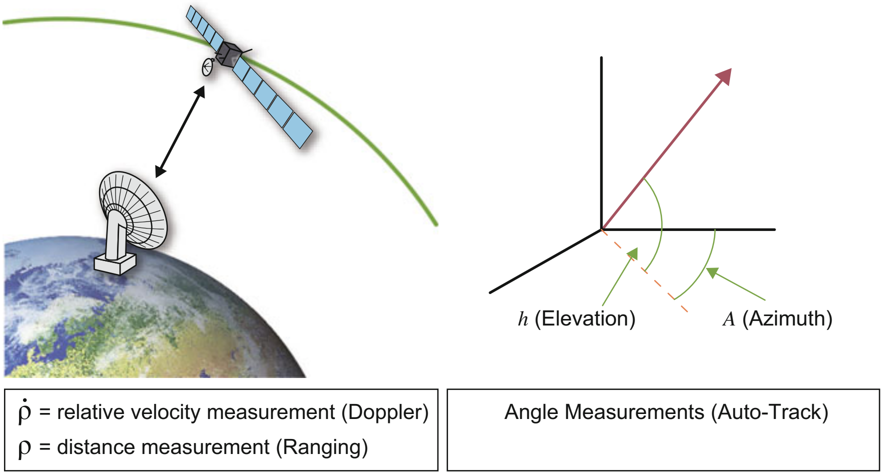


Fig. 4.18 Tracking data types

$$\sum_i \left[z_i - f_i(\boldsymbol{r}, \boldsymbol{v})\right]^2 = \min, \tag{4.13}$$

where z_i is the measured parameter and $f_i(\boldsymbol{r},\boldsymbol{v})$ is the computed value of the measured parameter based on the current orbital elements ($\boldsymbol{r},\boldsymbol{v}$). Two main techniques are used for orbit determination, the batch least squares fit and the Kalman filtering. If the OD process is successful and calculates a set of refined orbital elements, these orbit parameter sets can be used to predict the orbit and to generate required orbit-related products like the AOS and LOS times for the ground station network.

Maneuver Planning

If orbit thrusters are on board a spacecraft, maneuvers have to be planned according to the mission requirements. The planning for an orbit maintenance in LEO was discussed in Sect. 4.1.2.6; here we focus on the maneuver planning for the transfer from GTO to GEO.

As previously shown in Sect. 4.1.2.5, in-plane and out-of-plane maneuvers have to be performed to bring the spacecraft into its required target box in the geostationary ring. There are at least two main reasons why such an injection into GEO cannot be done by only one apogee maneuver.

1. The geometry of the high-elliptical transfer orbit is directly linked to the launch site and therefore all subsequent apogees have a dedicated longitude. There is a high probability that these apogee longitudes will never hit the

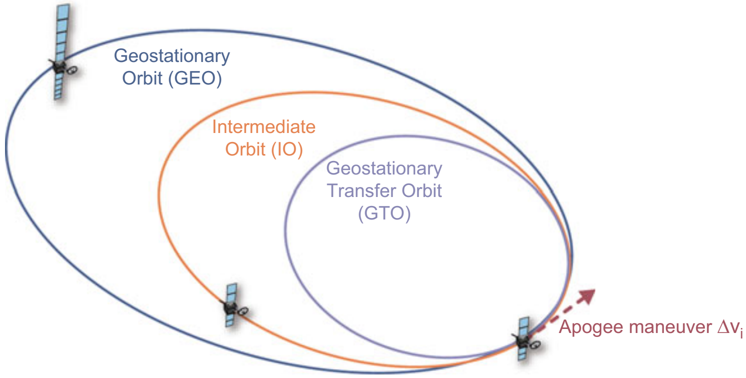


Fig. 4.19 GTO mission profile

longitude of the target box. Consequently, the perigee of the GTO has to be increased step by step in order to come into a drift orbit to the target box.

2. The performance uncertainty of the thrusters on board has an initial uncertainty of about 2 %. In case of an over-performance of a single injection burn with a corresponding Δv of about 30 m/s, the apogee will be exceeded, which has to be reduced again with the same amount of Δv. Such a “fuel waste” will reduce the lifetime of the satellite by about 60 weeks due to the necessary North/ South correction maneuvers with a Δv of about 1 m/s per week.

   In general, at least three apogee maneuvers will be performed to lift the perigee to the geostationary altitude. With the first one the thruster performance can be calibrated and the uncertainty be reduced to less than 1 % and with a relatively small last maneuver compared to the first ones the absolute over-performance can be reduced to a few centimeters per second.

So-called intermediate orbits (IO) are therefore used for the positioning of a satellite in GEO (refer also to Fig. 4.19). How many apogee maneuvers have to be used for the positioning depend on the mission requirements like the number of involved ground stations, dual-station visibility during an apogee maneuver, and backup strategies for the nominal maneuver sequence.

As example, Fig. 4.20 shows the fine station acquisition of a typical communications satellite in GEO, where the altitude offset w.r.t. the GEO altitude is plotted versus the East longitude. After the last apogee maneuver as well as the complete solar array and antenna deployments, the apogee shows a higher altitude as required. But this situation could be used to perform a safe flyby of the boxes of other satellites placed between drift start and the target box.

Once the satellite is placed in its target box, the orbit is furthermore influenced by perturbations of the gravity fields of Earth, Sun, and Moon. As a consequence, the satellite starts to drift inside the box in North/South and East/West direction (refer also to Fig. 4.21). Before the satellite exits its box, the drift has to be reversed or corrected. Consequently, the routine control center has to perform orbit determination each day to monitor the drift and depending on the requirements it has to plan and execute so-called station-keeping maneuvers.

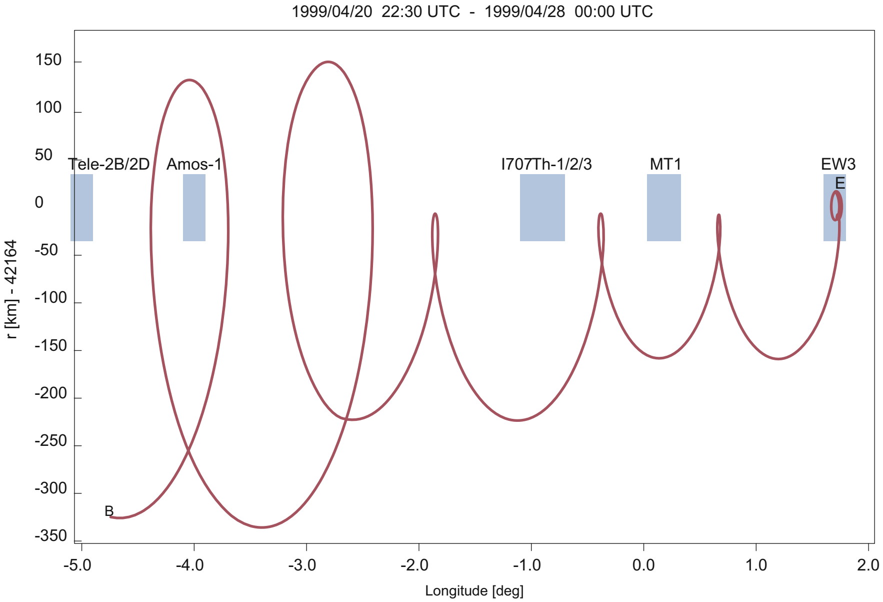


Fig. 4.20 Fine station acquisition of a GEO mission (example Eutelsat EW3 in 1999)

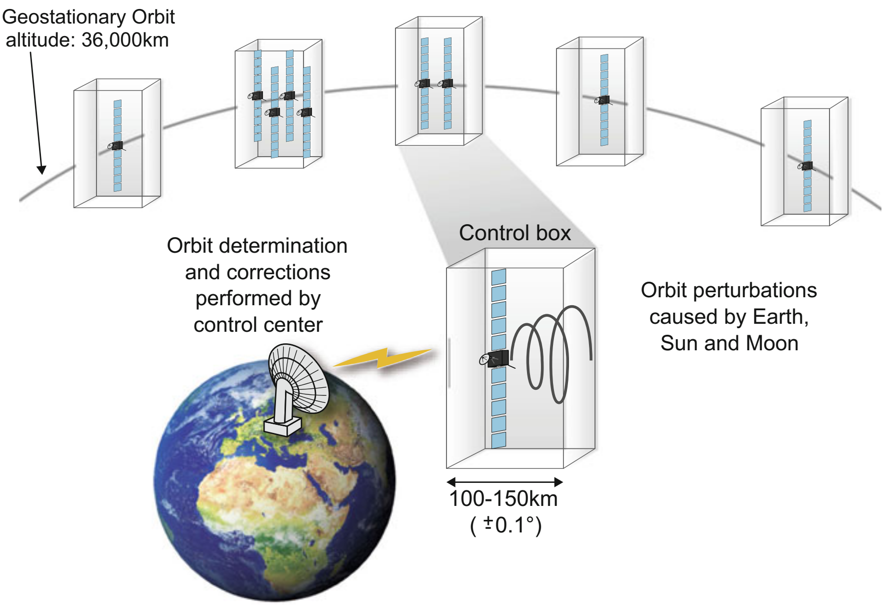


Fig. 4.21 Fine station acquisition of a GEO mission (example Eutelsat EW3 in 1999)

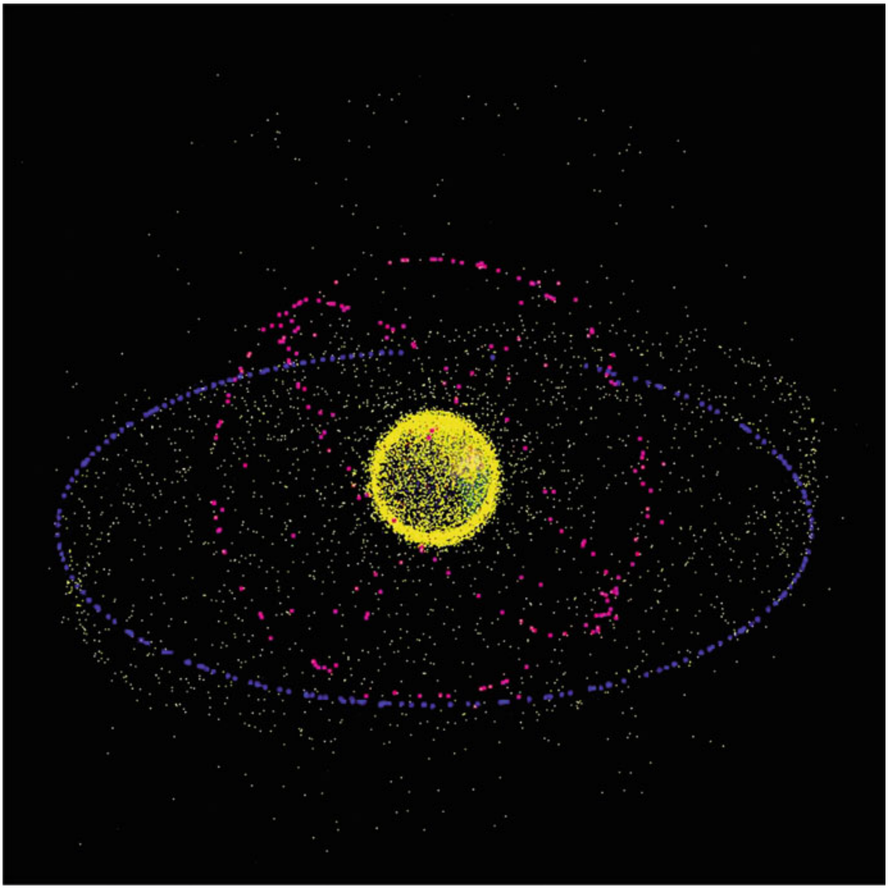


Fig. 4.22 Man-made space objects orbiting the Earth (*Source*: DLR)

As Fig. 4.21 shows as well, more than 1 satellite can be placed and have to be controlled in one box. In such a case, not only the natural drift has to be monitored, the control center has also to prevent a collision between the satellites inside one box.

Collision Avoidance Operations

In January 2009 a collision occurred between two artificial satellites, IRIDIUM 33 and COSMOS 2251, which created roughly 1,500 tracked debris and other small fragments still orbiting in the wide range of the LEO. Since this event, the monitoring and handling of critical conjunction events between active satellites and space debris got an important role for flight dynamics operations. Currently more than 15,000 objects are orbiting the Earth, where the orbital data with a reduced precision is publicly available (see also Fig. 4.22).

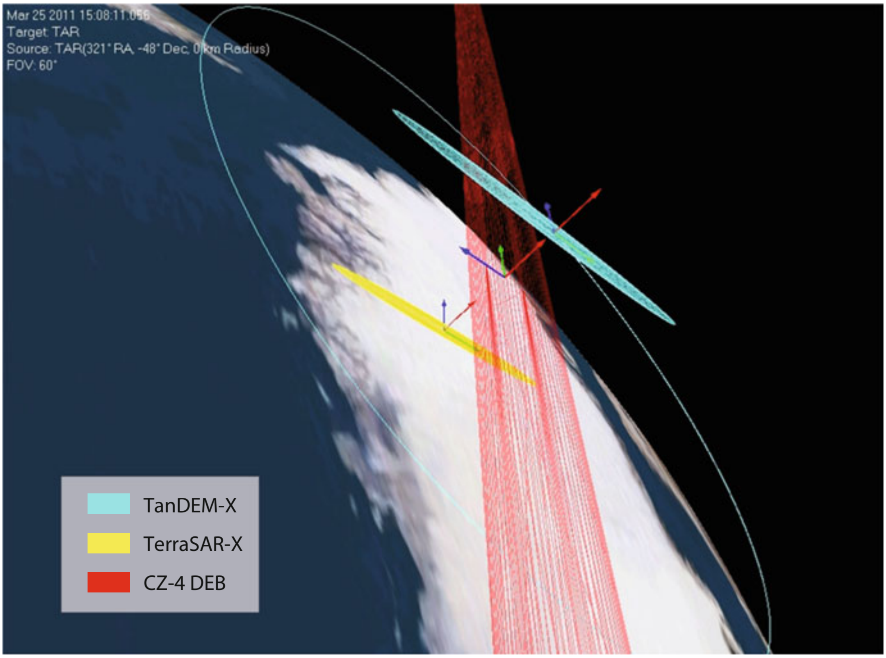


Fig. 4.23 Close approach between a CZ-4 debris (part of the error ellipsoid in *red*) and the TerraSAR-X (*yellow ellipsoid*)/TanDEM-X (*cyan ellipsoid*) formation (*Source*: DLR)

In general, the Joint Space Operations Center (JSpOC) in the USA tracks objects with a size larger than 10 cm by radar antennas in LEO and optical sensors in high orbits like GEO. Based on the tracking data of these sensors and an orbit determination for each space object, JSpOC performs critical conjunction calculations for all operated spacecrafts. Once they detect a close approximation, they contact the control center and provide data about the event including precise orbital data of both objects. As JSpOC is not aware of any executed or planned maneuver, the FD team can recalculate the conjunction event based on the information of JSpOC and the precise orbit data of its own satellite. Depending on the event geometry, the collision probability, and the error ellipsoids around the space objects, the FD team has to plan a collision avoidance maneuver taking into account dedicated mission requirements for the effected satellite.

As example of such an event, Fig. 4.23 shows a critical conjunction between a space debris particle and a close formation of two satellites in LEO. The conjunction occurred close to the North Pole, where the formation has its maximum radial separation of a few 100 m. The space debris passed the formation between the two satellites with a critical distance to one of the satellites. As a consequence, the FD team had to execute a collision avoidance maneuver half an orbit before the event and a formation reacquisition maneuver half an orbit after the event.

4.2 Attitude Dynamics

Jacobus Herman and Ralph Kahle

4.2.1 Introduction

The attitude of a spacecraft is always defined in relation to an external reference system. A few of the more obvious examples of such reference systems are a coordinate system based upon the satellite's trajectory, one with the Earth, the Sun, or another celestial body at the center, or an inertial coordinate system defined by the "fixed" stars (see also Sect. 4.1.2.1). The three angles between the spacecraft's body-fixed axes (for example denoted by a unit vector $\boldsymbol{x}_{\mathbf{sat}}, \boldsymbol{y}_{\mathbf{sat}}, \boldsymbol{z}_{\mathbf{sat}}$) and the chosen reference coordinate system (e.g., the unit vector $\boldsymbol{x}_{\mathbf{ref}}, \boldsymbol{y}_{\mathbf{ref}}, \boldsymbol{z}_{\mathbf{ref}}$) uniquely define the attitude. The transformation matrix from this reference frame into spacecraft body coordinates is often referred to as the attitude, or direction cosine matrix:

$$\boldsymbol{A} = \begin{pmatrix} \boldsymbol{x}_{\mathrm{sat}} \cdot \boldsymbol{x}_{\mathrm{ref}} & \boldsymbol{x}_{\mathrm{sat}} \cdot \boldsymbol{y}_{\mathrm{ref}} & \boldsymbol{x}_{\mathrm{sat}} \cdot \boldsymbol{z}_{\mathrm{ref}} \\ \boldsymbol{y}_{\mathrm{sat}} \cdot \boldsymbol{x}_{\mathrm{ref}} & \boldsymbol{y}_{\mathrm{sat}} \cdot \boldsymbol{y}_{\mathrm{ref}} & \boldsymbol{y}_{\mathrm{sat}} \cdot \boldsymbol{z}_{\mathrm{ref}} \\ \boldsymbol{z}_{\mathrm{sat}} \cdot \boldsymbol{x}_{\mathrm{ref}} & \boldsymbol{z}_{\mathrm{sat}} \cdot \boldsymbol{y}_{\mathrm{ref}} & \boldsymbol{z}_{\mathrm{sat}} \cdot \boldsymbol{z}_{\mathrm{ref}} \end{pmatrix} \tag{4.14}$$

whereby each dot product is the cosine of the angle between the two axes referred to (see for example Wertz 1978).

Intuitively easiest to grasp are the roll, pitch, and yaw angles which are also widely used in the navigation of ships and airplanes (see Fig. 4.24 and also Arbinger and Luebke-Ossenbeck 2006). Roll is the movement around the direction of motion (the length axis in case of ships and planes). Pitch denotes the angle around the axis perpendicular to the orbital plane—the stamping motion of a ship—and yaw the angle around the local vertical axis.

It is always possible to choose an orthogonal reference system, for example, denoted by $\boldsymbol{r}_x, \boldsymbol{r}_y, \boldsymbol{r}_z$, such that the instantaneous attitude of the satellite is defined by the rotation around a single axis and thus by a single angle θ.[1] This leads to the

[1] The single rotation axis is an eigenvector of the direction cosine matrix $\boldsymbol{A}$; for example, $\boldsymbol{A} \cdot \boldsymbol{r}_x = \boldsymbol{r}_x$, when $\boldsymbol{r}_x$ is the rotation axis.

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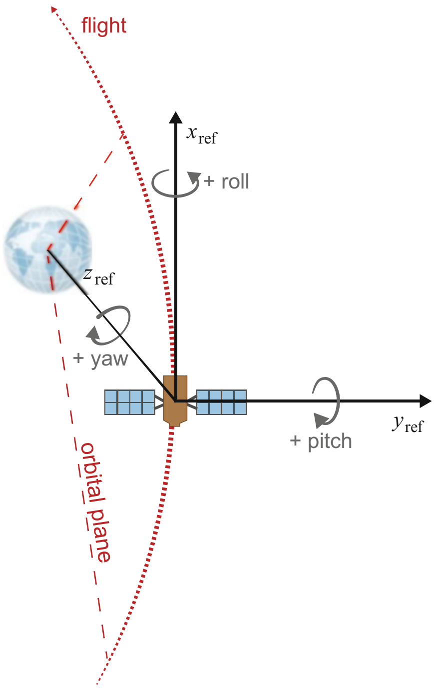

Fig. 4.24 Example of the roll, pitch, and yaw angles. Reference is normally the orbit frame. Roll is the movement around the direction of flight, yaw the movement around the local vertical axis, and pitch around the direction perpendicular to the orbital plane

definition of quaternions, or so-called Euler symmetric parameters. More than one representation is possible, but a possible choice is (see e.g., Wertz 1978):

$$\begin{aligned} \boldsymbol{q}_1 &= \boldsymbol{r}_x \sin\frac{1}{2}\theta \\ \boldsymbol{q}_2 &= \boldsymbol{r}_y \sin\frac{1}{2}\theta \\ \boldsymbol{q}_3 &= \boldsymbol{r}_z \sin\frac{1}{2}\theta \\ q_4 &= \cos\frac{1}{2}\theta \end{aligned} \tag{4.15}$$

whereby $q_1{}^2 + q_2{}^2 + q_3{}^2 + q_4{}^2 = 1$ and q_4 is the scalar component of the quaternion. Quaternions were designed especially for three-dimensional space and thus offer from computational point of view the most efficient way to handle all necessary attitude operations, such as propagation, vector and matrix multiplications, rotations, transformations, etc. Note that an attitude quaternion will normally be time dependent and its interpretation not straightforward anymore. Therefore, results are normally re-translated into more easily understood quantities such as the deviations from a default attitude, or into roll, pitch, and yaw angles.

The attitude will in general not be constant in time. Changes are introduced by internal disturbances, e.g., caused by the firing of a thruster or a change in the speed of a reaction wheel (see Larson and Wertz 1992 and references cited therein for descriptions of all sensors and actuators that are commonly used), and also by external influences such as radiation pressure or gravity gradients. The deviations from a prescribed attitude which are still acceptable depend upon design and environment, but most of all on the task in hand. Clearly, providing Sun on the panels or pointing the nadir-antenna towards the Earth has less stringent requirements than imaging a star or mapping part of the Earth's surface with a resolution of one meter. Most missions have therefore three distinct regimes of attitude control:

1. Rate damping.
2. Coarse control in a basic safe mode (e.g., designed to survive a situation with low power or with thermal problems).
3. Fine control used for payload operations and orbit control.

The exact definitions vary from mission to mission, but a rough indication is provided by the following numbers:

1. Rates > 0.2–0.3°/s must be damped before operations can start; high rates can for example occur shortly after launch when the satellite is separated from the upper rocket stage.
2. Coarse attitude control points the satellite with a typical accuracy of a few degrees, which is good enough to guarantee for example power on the solar panels or to establish a link to a ground station.
3. Fine pointing generally is of the order of arc-minutes or better.

Determination of the spacecraft's attitude and its variability depends upon the desired accuracy and the available environment. It might for example be feasible to use the magnetic field or radiation for attitude determination when flying close to a central body, but at some distance the signal strength might become too weak to be usable. Widely implemented is a configuration like the one discussed below, but each mission may and will have individual solutions.

1. Large rates are often measured with an optical gyroscope (IMU = inertial measurement unit which can handle up to ~15°/s; Larson and Wertz 1992). Smaller rates can also be obtained by determining the time derivative of subsequent attitude measurements.
2. A simple (and hence robust) bolometer can yield the directions of the Earth (or another central body) and the Sun with an accuracy of a couple of degrees. Close to the Earth a magnetometer (Larson and Wertz 1992) may deliver similar results. Reaction times of such devices may be of the order of several seconds (CESS = coarse Earth Sun sensor. Bolometric Sun sensors with higher precision have also been developed but are less frequently used).
3. A star tracker (Larson and Wertz 1992) typically provides information with an accuracy <1 arc-min in all three axes. Computations can be done at least once per second, thus also enabling rates to be determined accurately.

Several general principles must be taken into account before using such measurements for attitude control. First of all, the on-board computer normally calculates the desired corrections at a fixed frequency, so that the measurements have to be propagated to the next grid point. Secondly, all measurements exhibit noise which is normally suppressed by applying filters. Spurious or missing points must be flagged as unusable and some kind of damping mechanism must be included in order to avoid “bang-bang” control.

Actuators are then used to actively control the attitude. Once again, the exact choice will depend upon environment and mission goals. Commonly used are thrusters, reaction wheels, and, when a sufficiently strong magnetic field is present, magnetic torque rods (Larson and Wertz 1992). Other, more exotic solutions have also been tried, but will not be covered here. A more exhaustive enumeration of equipment used for attitude control can be found in Sect. 6.5 on AOCS operations.

4.2.2 Disturbances

The attitude of each satellite is influenced by several intrinsic and external factors, the relative importance of which depends once more upon build and environment. Some of these influencing factors may be actively used to control the attitude.

4.2.2.1 Satellite Intrinsic

Reaction wheels provide a powerful and accurate way to control the attitude in a fine pointing mode. Disturbances which are several orders of magnitude larger than the allowed tolerances may occur in case of malfunctioning, or when one of the wheels is switched off completely. Turning off all wheels simultaneously will also lead to severe attitude deviations due to the unavoidable asymmetrical run-down.

Thrusters used for orbit correction maneuvers are designed to act through the center of gravity. However, the CoG may shift during the mission for example due to the emptying of the fuel tanks. Originally equal thrusters may develop dissimilarities with time, which also leads to attitude disturbances if maneuvers are made with several thrusters at the time.

Movable or flexible parts of the satellite may influence its attitude. A camera, antenna, or other instrument might be movable and lock into a prescribed slot at the end of a slew. A boom or another flexible appendix leads to disturbances as does the sloshing of fuel in an emptying tank.

4.2.2.2 External Influences

In the vicinity of the Earth, or another massive central body, the major disturbance of the attitude stems from the differential gravitational force acting on the satellite.

The gravity gradient tends to point the length axis, or more precisely the axis with the smallest moment of inertia, towards the central body.

The influence of solar radiation pressure and to a far lesser extent the pressure of the reradiated IR radiation of a central body depends upon distance, the area-to-mass ratio, and properties of the satellite such as reflectivity. The geometry of the orbit also plays a role. The disturbance of radiation pressure will be more or less constant when flying in Sun-synchronous orbit, but will be variable in other orbits.

Ram pressure or air drag can become important for an orbit close to a central body. It is a differential effect depending on asymmetric properties of the spacecraft. It is in general several orders of magnitude smaller than the effects mentioned above. Satellites flying around the Earth at altitudes $\leq$400 km will be affected, whereby the height of the remnant atmosphere depends upon solar activity.

The smallest effect of all is caused by the interaction with the magnetic field. Satellites which use magnetic torque rods for attitude control are confined to Low Earth Orbits $\leq$500 km where the field strength is of the order of 40 nT. Disturbances are typically an order of magnitude smaller than the effects of air drag. However, if the satellite possesses a significant magnetic dipole of its own (occurs for example in "SAR" missions that use a Synthetic Aperture Radar which sends out a strong electromagnetic signal) the torques can be as large as the disturbance caused by the gravity gradient.

A quantitative example is included here for a satellite with a mass of 480 kg and a cross section in flight direction of 1 m^2, flying in a polar orbit at 490 km altitude.[2] Here the gravity gradient is of order 10^{-4} Nm, aerodynamic drag 10^{-5} Nm, the disturbance by solar radiation 10^{-6} Nm, and the magnetic torque 10^{-7} Nm (see also d'Amico 2002).

4.2.3 Attitude Determination

Two external reference points suffice in principle to determine the attitude. The practical implementation is considerably more complicated, though. This will be illustrated by looking at the steps necessary to evaluate star tracker data.

1. A CCD (charge coupled device) image of a portion of the sky is made each duty cycle of the attitude control system. The field-of-view is typically ~100 deg^2 comprising some 20–100 bright stars (Fig. 4.25). The limiting magnitude and minimum signal-to-noise ratio might be autonomously adjusted or be set per ground-command. A baffle provides protection against stray light from, e.g., the Sun or the Moon.
2. Comparison with a star catalogue, which is stored on board, yields the pointing direction of the star tracker's bore sight and the orientation of the FOV. The

[2] Properties of the two GRACE satellites (Gravity Recovery And Climate Experiment) launched in 2002.

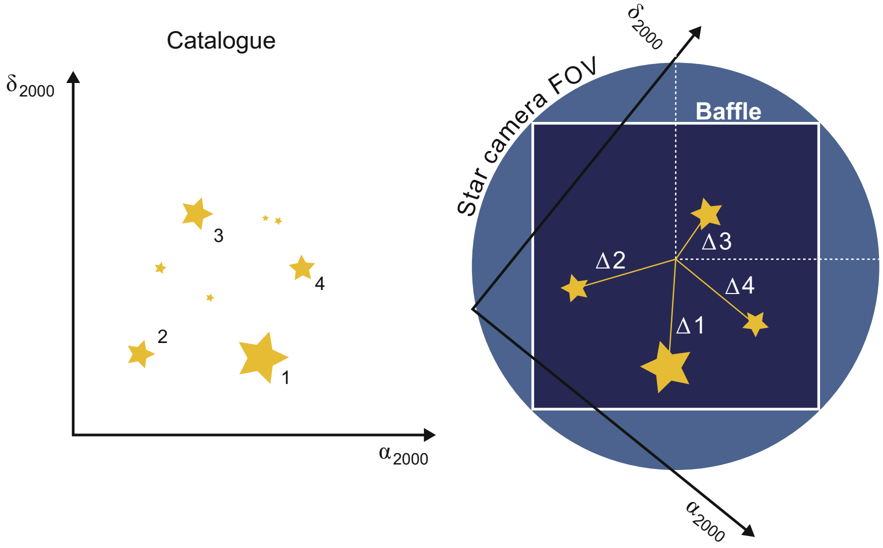

Fig. 4.25 The bore sight and orientation of the star tracker is determined in Y2000 coordinates by comparison with a catalogue. Magnitude information is also used to guard against spurious detections. The high redundancy allows an estimate of the quality of the solution to be made

apparent stellar positions must be corrected for the effect of aberration.[3] Very precise measurements further require the inclusion of a correction for the proper motion of the stars.

It is enough in principle to have two identified stars (position on the sky and magnitude) to fix the camera's looking direction and orientation. The surplus information from 20–100 is used to get an estimate of the accuracy of the determination and will also guard against spurious identifications.[4]

3. The attitude of the star tracker is now known in the coordinate system of the catalogue, usually right ascension and declination (α,δ) at epoch 2000.0.[5]
4. The location and built-in angles of the star trackers on the satellite are known from prelaunch measurements, so that it is easy to transform the result into the attitude for the three spacecraft's body axes (see Fig. 4.26). This might already be sufficient if for example an instrument has to be pointed towards a galaxy or a star. Note that the built-in angle of the star tracker and the motion of the satellite

[3] Aberration is the apparent displacement of a star due to the motion of the observer; for a satellite moving at a velocity v with respect to the fixed stars the displacement $\approx v/c \sin\varphi$, where c is the velocity of light and φ denotes the angle between motion and star direction.

[4] Another satellite, a comet, an asteroid or meteoroid could transitorily be identified as a star. Blinding could lead to afterglow on one or more pixels with the same result.

[5] Right ascension is measured along the Earth's equator towards the East; zero point is the direction of the vernal equinox at the specified epoch. Declination is measured perpendicular to the equator and is positive towards the North.

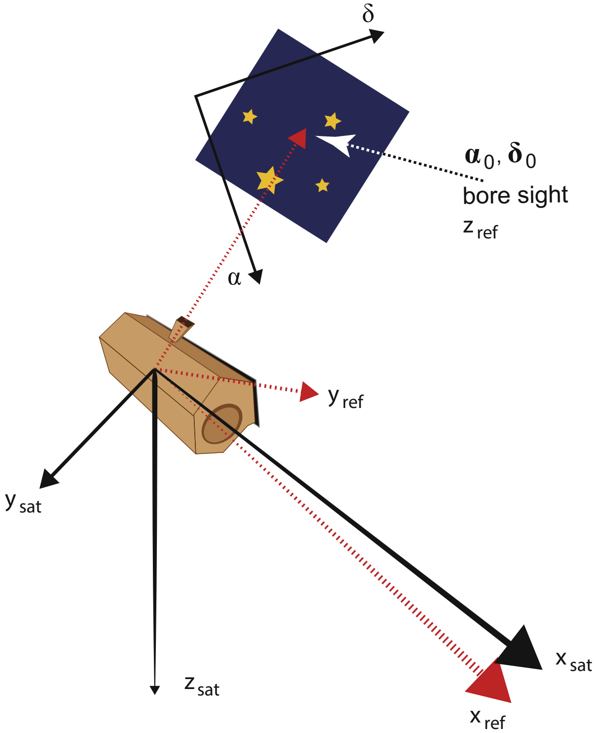

Fig. 4.26 Transformation of the measured star tracker's attitude (here indicated by the system with suffix "ref" which has for illustration purposes one of its axes aligned with the bore sight) into the body frame yields the attitude of the satellite in Y2000 coordinates (suffix "sat"). The relation with the measured star field is indicated

along its orbit imply that the accuracy of the attitude determination will in general not be identical in the three body axes.

5. If, however, the purpose is to point an instrument at a certain location on the Earth's surface, further transformations are necessary. The first step is to insert the precession and nutation from the epoch of the catalogue until the exact moment of the determination. Use of the instantaneous direction of the vernal equinox and the current value for the obliquity of the ecliptic (the angle between the equator and the plane of the ecliptic) leads to the denomination "true-of-date" (Fig. 4.27).
6. Additional information on the current position of the satellite in its orbit allows the transformation from inertial to Earth-fixed coordinates. Position information can be sent to the spacecraft in the form of an ephemeris, or as a single state vector at a given epoch which is then propagated with an on-board model. The current generation of satellites often uses GPS receivers to determine this information autonomously on board (only feasible in orbits around the Earth with an altitude $<$10,000 km). After this step the attitude can be expressed in the intuitively understood roll, pitch, and yaw angles.
7. Further corrections in order to relate the attitude to a target on the surface might need to take into account the oblateness of the Earth or the random polar motion. The latter can be implemented by correcting the time from UTC to UT1 (Seidelmann 1992).
8. Payload operations require a final transformation into an instrument frame.

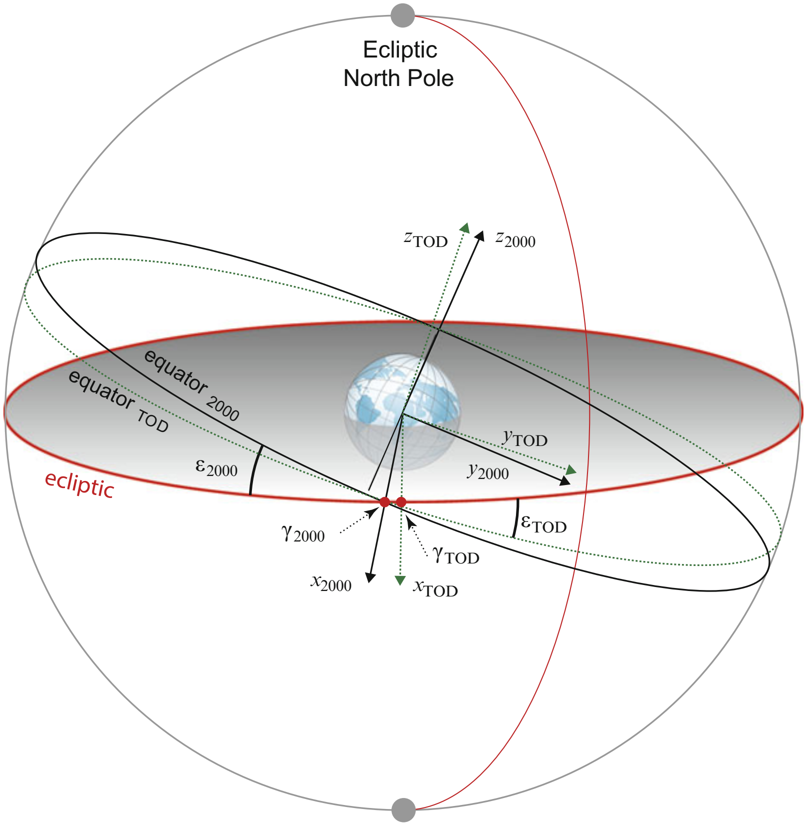

Fig. 4.27 The relation is shown between the inertial earth-centered coordinate systems Y2000 and "true-of-date" (TOD; see text for further details). Targeting of a point on the surface requires time information and knowledge of the current position in orbit also. Thereafter the attitude can be expressed in the familiar roll, pitch, and yaw angles. High precision applications might require additional corrections such as for the UT1-UTC time difference or for the oblateness of the Earth

The above is by no means unique, but different missions, different payloads, different sensors, or different central body might all lead to adaptations of this scheme. Several operational aspects have to be considered always though.

- The sensor used for attitude determination in fine pointing mode will normally have at least one backup. The mutual alignments of prime and backup(s) have to be matched to such a degree that payload operations are not disturbed if a switch-over is necessary. The same holds true for redundant sensors in other modes.
- Such mounting matrices are of course measured when building the satellite but might change during the mission, especially at launch. Also, characteristics

in orbit might differ from those in the laboratory. Note that in case of misalignments detected in flight, one sensor has to be chosen as reference and the orientation of all others will be related to this one. A good choice would for example be a star tracker which serves as cold backup, i.e., it is not operationally used at the time.

- Similarly, results from different types of sensors have to match within their respective accuracies.
- Computations might use measurements from different types of sensors which requires their relative weights to be set. The direction to the center of the Earth can for example be derived from CESS and magnetometer data with a relative weight of 2:1 (numbers are arbitrarily chosen in this example).
- Additional information is usually required, before the sensor information can be evaluated. The below would be exemplary for a satellite in Low Earth Orbit.

(a) Star tracker:	Catalogue (e.g., in Y2000 coordinates) containing all stars brighter than a given magnitude
	Current epoch
	Precession and nutation matrices as a function of time
	Mounting matrix with respect to the satellite
	Orbit position
(b) GPS receiver:	Mounting matrix
	GPS SV (also called PRN) usability
	GPS SV ephemerides
	GPS clock correction
	Model for orbit propagation
(c) CESS:	Position of heads and thermistors
	Calibration of temperatures
	Model of terrestrial (or central body) albedo
	Position of the Sun as a function of time (a simple sinus function normally suffices)
(d) Magnetometer:	Mounting matrix
	Model of the terrestrial magnetic field
	Orbit position
(e) IMU:	Mounting matrix

- Propagation to next grid point (model and or rates)
- Rates from time differentiation or from separate measurement
- Noise may require use of filters; obviously wrong measurements should be discarded. This requires quality control, e.g., by monitoring the point-to-point variation or signal-to-noise ratio and flagging of "invalid" points.
- Gaps require special handling

(a) Expected outages E.g., the star tracker used will be blinded by the Sun (provided the orbit continues as is)

(b) Unexpected	Star tracker blinded by another satellite Long stretch of invalid measurements e.g. due to sensor malfunctioning or wrong parameter adjustment Software problem or any other problem causing sensor outage

Mitigation measures are designed to minimize the effect on payload operations. A switch to the redundant sensor, either autonomously done on board or by a priori commanding is the easiest solution.

It might be possible to wait until the sensor delivers valid data again and in the meantime to bridge the gap by propagating the last valid attitude measurement; this usually requires a model of the disturbances acting on the spacecraft and/or the availability of independent rate measurements. However, even with sophisticated means a propagation of attitude information usually doesn't remain within the limits necessary for payload operations for a long time.

Cruising through the outage, i.e., disable active attitude control, is another possibility, but obviously deviations will grow even quicker than in case of propagation.

Finally, there could be an option to switch to another spacecraft mode that uses different sensors which are not affected.

4.2.4 Attitude Propagation

A prediction over a certain time interval in the future can be made as soon as the current attitude and angular rates have been determined. The simplest propagation would be a linear extension of the measurements, a so-called kinematic model. The combination of an attitude matrix $\boldsymbol{A}$ [e.g., the one from Eq. (4.14)] with the skew matrix for rotation $\boldsymbol{\Omega}$ leads to the former's time derivative:

$$\frac{d\boldsymbol{A}}{dt} = \boldsymbol{\Omega} \cdot \boldsymbol{A} \tag{4.16}$$

$\boldsymbol{\Omega}$ is defined in terms of the angular rates around the three principal axes of the reference coordinate system ω_x, ω_y, and ω_z (denoted by $\boldsymbol{\omega}$ in the following):

$$\boldsymbol{\Omega} = \begin{pmatrix} 0 & \omega_z & -\omega_y \\ -\omega_z & 0 & \omega_x \\ \omega_y & -\omega_x & 0 \end{pmatrix} \tag{4.17}$$

The equivalent when Euler symmetric parameters are used [see Eq. (4.15)] becomes $\mathrm{d}\boldsymbol{q}/\mathrm{d}t = \frac{1}{2}\boldsymbol{\Omega} \cdot \boldsymbol{q}$ with the skew matrix defined in this instance as

$$\boldsymbol{\Omega} = \begin{pmatrix} 0 & \omega_z & -\omega_y & \omega_x \\ -\omega_z & 0 & \omega_x & \omega_y \\ \omega_y & -\omega_x & 0 & \omega_z \\ -\omega_x & -\omega_y & -\omega_z & 0 \end{pmatrix} \tag{4.18}$$

A kinematic model can in practice be used only in the absence of disturbance torques (e.g., interplanetary flight without active control) or in order to bridge short time intervals (for example propagation to the next second grid point in the on-board computer, normally less than one second).

A more elaborate and better prediction can be attained by adding a dynamic model, which takes all internal and external disturbances into account (see Sect. 4.2.2 for an overview). The complexity increases with the accuracy to be obtained. Each of the disturbance torques has to be modeled and projected onto the chosen reference system, for example, body-fixed coordinates. All terms then have to be included in the right-hand side of Eq. (4.16). The number of disturbance torques taken into account will depend upon the actuators used, the build of the satellite, and the accuracy that has to be reached. Euler's equation of dynamic motion is mostly written in the form of

$$\boldsymbol{I}\frac{\mathrm{d}\boldsymbol{\omega}}{\mathrm{d}t} = \boldsymbol{T}_d + \boldsymbol{T}_c - \boldsymbol{\omega} \times (\boldsymbol{I} \cdot \boldsymbol{\omega}), \tag{4.19}$$

where $\boldsymbol{T_d}$ and $\boldsymbol{T_c}$ denote the overall disturbance and control torques, respectively, $\boldsymbol{I}$ is the tensor for the satellite's moments of inertia, and the vector $\boldsymbol{\omega} = (\omega_x, \omega_y, \omega_z)$. $\boldsymbol{T_c}$ will normally include the torques by the wheels and/or the thrusters and maybe also the magnetic torque rods when present, whereas $\boldsymbol{T_d}$ will comprise the torques caused by the gravity gradient, by the air drag and possibly by the solar radiation pressure and by the magnetic field also.

The vector product $\boldsymbol{I}\cdot\boldsymbol{\omega}$ is the total angular momentum of the satellite, often denoted as $\boldsymbol{L}$. Note that the last term in Eq. (4.19) is a vector cross-product. It can be seen from Eq. (4.19) that the relative influence of disturbance torques diminishes if either $\boldsymbol{\omega}$ or $\boldsymbol{I}$ becomes large. The attitude of fast rotating satellites or of large structures such as the international space station ISS is not that sensitive to disturbances.

Finally note that Eq. (4.19) is valid for a rigid body only. Inclusion of the dynamics of flexible or irregularly moving structures is normally too complicated to be considered. A more elaborate and detailed treatment falls outside the scope of this chapter, but examples can be found in Wertz (1978), Sidi (1997) and Wiesel (1997).

4.2.5 Attitude Control

Attitude control can be either passive or active. Passive control can for example be achieved by stabilization due to gravity gradients, by the alignment of an on-board magnet with an external field, or by a high spin rate sustained over a long time which could for example be applicable during the cruise phase in interplanetary flight. Far more common is active control though.

Once the spacecraft's attitude and rates are determined and a model for all disturbing influences implemented, future trends can be predicted and thus counteracted by applying one or more of the satellite's actuators. Commonly used are

- *Thrusters*: a pressurized propellant in a central reservoir is led to several thrusters that singly, or in combination can control one of the spacecraft's axes. The torque is achieved by expelling a small amount of mass and is described well by the Tsiolkovsky rocket equation in the low-mass approximation (see e.g. Wiesel 1997). Thrusters can be used in any space environment, but have the obvious disadvantage that the amount of propellant is limited.
- *Reaction wheels*: a change in the rotation velocity of one—in case of a three-wheel configuration—or several—in case of the more often used four-wheel configuration—wheels leads due to conservation of angular momentum to an opposite rate change in one of the satellite's axes. The transfer of momentum will in practice be less than 100 % due to friction. Reaction wheels can also be used universally, but require a significant amount of power which must either be provided by an internal source (e.g. a battery) or by the Sun. Also, the amount of lubricant carried is limited meaning that at some time the friction will become too large for further operation.
- *Magnetic torque rods*: these are mounted parallel to each of the spacecraft's axes. A current sent through the rod will produce a Lorentz force in the presence of a magnetic field. This requires a power source and a strong enough field. The torque is perpendicular to the current and to the field, which implies that an axis aligned with the field can't be controlled at that particular position in orbit.

Attitude control is normally implemented as a fully autonomous process on the satellite (see Sect. 6.5). Manual commanding is impracticable due to the short duty cycle (≤ 1 s in general), but the possibility to intervene in the control loop in case of problems is still present on most satellites.

A change of the default attitude will normally be commanded manually. An example of this would be the commanding of a 90° yaw slew that will be treated below, but it might also be the pointing of the instrument towards a target or preventing irradiation of certain parts of the spacecraft by turning it away.

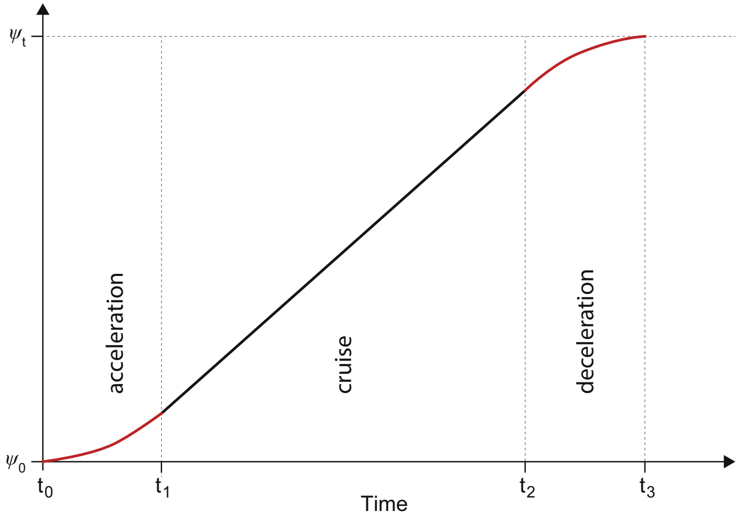

Fig. 4.28 Profile for a yaw slew from ψ_0 (default) to ψ_t (target, e.g., 90°). Acceleration starts at t_0 and ends at t_1, where the cruise phase at constant rate starts. The deceleration phase starting at t_2 lasts just as long as the acceleration and ends at t_3

Consider a satellite flying around the Earth with its orbit control thrusters all located on the backside.[6] Then, in order to change the inclination of the orbit an "out-of-plane" maneuver is necessary, preferably near one of the nodes. This requires a ±90° yaw slew with the offset reached at a very precise moment, namely just before the start of the burn. It is implemented by defining a profile connecting start and target attitude (ψ_0 and ψ_t respectively) with user-defined rate $\dot{\psi}_0$ and acceleration $\ddot{\psi}$.

The acceleration and deceleration phases are equally long and determined by the time it takes to reach the cruising rate.

$$t_1 - t_0 = t_3 - t_2 = \frac{\dot{\psi}_0}{\ddot{\psi}}. \tag{4.20}$$

The angle traveled in the acceleration phase is

$$\Delta\psi = \frac{1}{2}\ddot{\psi}(t_1 - t_0)^2 \tag{4.21}$$

[6] TerraSAR-X, TanDEM-X, and PAZ are all build like that.

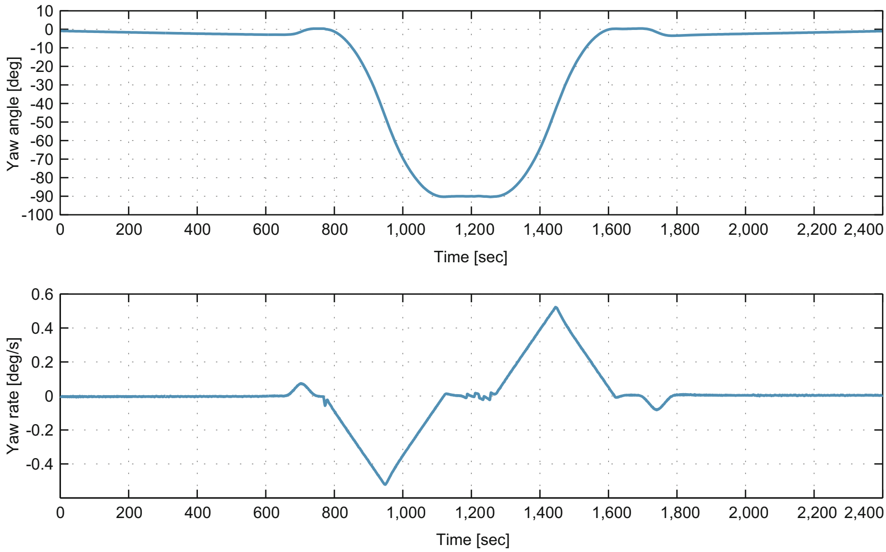

Fig. 4.29 The *top panel* shows the actually achieved profile for a yaw slew from ψ_0 (default, here 0°) to ψ_t (target, here −90°) preparatory to an inclination maneuver on TerraSAR-X. The corresponding rate measurements are shown in the *bottom panel*

so that $(\psi_t - \psi_0) - 2\Delta\psi$ is the angle remaining to be traversed in the cruise phase (Fig. 4.28). This of course takes

$$t_2 - t_1 = \frac{(\psi_t - \psi_0) - 2\Delta\psi}{\dot{\psi}_0}. \tag{4.22}$$

The maximum cruise rate and acceleration that can be chosen depend of course upon the power of the actuators, but more critical on the limits set by the sensors that are required to deliver attitude and rate information during the slew. Most star trackers for example will not function above a rate of ~0.5°/s. Finally note that the dwelling time at a non-nominal attitude might be limited e.g. by power constraints.

An example from the praxis can be found in Fig. 4.29, where the measured yaw angles and rates are shown for a −90° yaw slew. The prescribed profile is followed closely, but some slight deviations occur. For $t < 700$ s and for $t > 1{,}800$ s the yaw angle is not 0°; this is due to the fact that the TerraSAR-X satellite performs so-called yaw-steering, a roughly sinusoidal motion superimposed on the orbit with an amplitude of ~3° to compensate for the Earth's rotation during data takes. Yaw-steering is switched off in preparation of the slew ($\psi = 0$ at $t = 750$ s, rates > 0 from $t = 700$ to 750 s). Acceleration starts at $t = 750$ s and ends at ~920 s

[Eq. (4.21)]. The choice of parameters in this case is such that the cruise phase [Eq. (4.22)] that now begins is very short and ends at $t = 925$ s already with the onset of the deceleration phase. The yaw angle shows a small overshoot at $-90°$ and some rate disturbances are visible around $t = 1{,}200$ s due to the maneuver. The return slew to $\psi = 0°$ starts at $t = {\sim}1{,}250$ s. The operation is almost symmetrical, the only difference being that the re-enabling of the yaw-steering has a delay of 1–2 min (allows different maneuver sizes to be made with the same procedure).

Autonomous attitude control on board is based upon the attitude determination (4.16) and propagation (4.17). The latter will normally include disturbances (see Sect. 4.2.2). The deviation from the desired attitude, which can either be a default attitude or a profile as defined above, and the rate of change in this deviation yield the direction and magnitude of the torque to be applied to the spacecraft in order to correct these errors.

The desired torque might then be distributed over several types of actuators, if that is wanted. One could, in order to save fuel for example, send as much as possible to the magnetic torque rods and only if that is not sufficient invoke the thrusters for the remainder. The computed torques must finally be transformed into the coordinate system of the actuators; in case of a four-wheel configuration this will be a 3×4 matrix.

The calculation of the desired torques from the measurements can take several layers of complexity. The principle will be illustrated here for a one-dimensional controller. Denote the predicted deviation in roll by δ_{roll}. The time of the computation could for example be the next grid point of the on-board processor. The prediction itself might be complex too (see Sect. 4.2.4). The easiest controller to build is a proportional one that reacts directly on the predicted deviation. The desired torque T_{des} can be written as:

$$T_{\mathrm{des}} \sim -G\,\delta_{\mathrm{roll}}. \tag{4.23}$$

The gain parameter G determines the strength of the reaction and is normally configurable. The result will be a rather crude bouncing between the upper- and lower- dead bands. A better way therefore is to include the rate of change as well as the predicted deviation. The result is a so-called PD controller (proportional, differential) which takes the form

$$T_{\mathrm{des}} \sim -G_1\delta_{\mathrm{roll}} - G_2\frac{\mathrm{d}\delta_{\mathrm{roll}}}{\mathrm{d}t}. \tag{4.24}$$

The gain parameters G_1 and G_2 are once more configurable and can themselves be complex functions of actuator properties. A final refinement not only takes the predicted values, but also includes the time interval before. The length of the interval ($t_1 - t_0$) can be chosen and it is also possible of course to include a function that gives less weight to older values. This yields a so-called PID controller (proportional, integral, differential) which is widely used nowadays.

$$T_{\mathrm{des}} \sim -G_1 \delta_{\mathrm{roll}} - G_2 \frac{\mathrm{d}\delta_{\mathrm{roll}}}{\mathrm{d}t} - G_3 \int_{t_0}^{t_1} \delta_{\mathrm{roll}} \, \mathrm{d}t. \tag{4.25}$$

The performance of the attitude control system can and normally will be evaluated by high precision attitude determination *a posteriori*. The on-board restriction that requires a full new computational cycle for each grid point (≤ 1 s) vanishes and long stretches of data can be used. Such a high precision *a posteriori* attitude might also be required by the users of the payload, for example, for very accurate image processing or in order to combine images taken at different times.

4.2.6 Tasks of AOCS (Attitude and Orbit Control System)

Each mission normally has one or more engineers dedicated to each particular subsystem. It normally comprises such areas as instrument or payload operations, power/thermal, data and on-board computer, as well as AOCS. Tasks of AOCS are manifold and generally it is one of the busiest subsystems during LEOP (launch and early orbit phase) and also during the further mission.

- In LEOP the performance and alignments of all sensors and actuators used for attitude determination and control must be verified (including redundant ones). A reconfiguration might be required in case of defunct components (more likely to occur later in the mission).
- AOCS has a close interaction with the users of the instrument. Control parameters must be adjusted to optimize payload output; sometimes this can continue throughout the mission [a prime example is the GRACE mission where the satellites themselves are the instrument (d'Amico 2002)].
- There is also a close interaction with FD (flight dynamics). The performance of maneuvers must be evaluated and taken into account in future planning. The direction of thrust might be different from the a priori calibration and might shift during the mission. Single thrusters can under- or over-perform and might even become defunct.
- Non-standard attitude maneuvers can be required, especially during LEOP, which need manual computation and commanding (also with FD).

- Precise attitude determination might be required for payload checkout and instrument calibration (again in co-operation with FD).
- Close interaction with P/T (power/thermal subsystem); the temperature and power range of each sensor and actuator must be monitored, especially when switched on or off. The performance in terms of current (cost function) can become important when in the later stages the batteries start degrading.
- The performance of all actuators must be monitored during the entire mission (e.g., to detect increasing friction or leakage).
- Resources must be administered (fuel expenditure, book-keeping of the number of thruster cycles).
- Sensor gaps (e.g., intrusions) must be avoided by switching to a redundant sensor or by a configuration change.
- Special wishes (orbit correction maneuvers, imaging at special attitude, new targets) must be accommodated.
- A complete re-design might be required in case of serious changes in the mission's design. Examples of these are the new magnetorquer safe mode on the TerraSAR-X and TanDEM-X satellites that is not using the thrusters anymore (became necessary to safely fly in a formation with mutual distances <150 m), or the magnetic yaw-steering for GRACE, which was designed to align the satellite with the magnetic field (necessitated by the failure of several CESS thermistors).

The above also requires offline support from the flight dynamics group and for most missions also from the users of payload instruments. An in-depth overview of the AOCS subsystem is given in Sect. 6.5. Here only two examples will be given of the interaction of AOCS and FD with the on-board attitude control system.

4.2.6.1 Example 1

Interaction with the satellite is normally not per single command but with prepared procedures which not only contain the several commands required for a certain action, but also the timing, pre- and post-conditions that must be met, possibly the setting of the correct attitude (which for example in case of a 90° yaw offset will take a certain amount of time to achieve and might need to be carried out at a certain orbit position), possible changes in FDIR settings, timely interruption of payload operations, mode changes, switch on of additional telemetry packets, and — in case the action is done in real time — telemetry checks to be performed at the console.

The procedure shown in Fig. 4.30 can only be used time-tagged, i.e., the complete sequence is prepared beforehand, uploaded to the satellite and then stored in the on-board buffer. It will be carried out autonomously at the specified time.

Procedure for an inclination maneuver with +90° yaw slew on TerraSAR-X (Note that the only parameter input is the start time of the maneuver and its duration)		
1	Pre-conditions	Satellite must be in the correct AOCS mode The thruster branch must be available and active Time-tagged commanding only Payload and other AOCS operations are not possible from 9 m 40 s before until 14 m 50 s after the start of the maneuver The maximum burn time is 180 s
2		The reference time, $\mathbf{t_0}$, is the start time of the maneuver
3	**t_0-00:59:52**	Additional heaters near the thrusters are switched on
4	**t_0-00:11:00**	Additional telemetry packets containing parameters pertaining to the maneuver are switched on
5	**t_0-00:09:50**	The autonomous reaction which triggers if deviations larger than a specified limit occur on the Sun vector are disabled
6	**t_0-00:09:50**	The table showing the settings of the on-board surveillance is dumped
7	**t_0-00:09:45**	Payload operations on TerraSAR-X require continuous correction of the yaw angle in order to image the desired swash on the Earth's surface. This so-called "yaw-steering" is disabled
8	**t_0-00:09:44**	The autonomous recovery from "attitude hold mode" into the mode for payload operations is disabled
9	**t_0-00:09:40**	The satellite is commanded to attitude hold mode; this mode uses the same sensors and actuators as the mode for payload operations, but has larger tolerances on attitude and rate deviations. It also allows one to command any orientation with respect to the orbit frame. A margin of three minutes is included to let the spacecraft stabilise
10	**t_0-00:09:35**	The commands in step 10, …, 14 set and activate an on-board
11	**t_0-00:09:34**	safety mechanism that would stop the thrusters in case they
12	**t_0-00:09:31**	would still be active after the specified end of the burn. Terra-
13	**t_0-00:09:28**	SAR-X uses off-modulation to control the attitude during the
14	**t_0-00:09:26**	burn, so a 10% margin of the maneuver duration is added
15	**t_0-00:06:40**	A yaw slew to +1.574 mrad is commanded. Note that the 90° maneuver is not made with exactly 1.571 mrad, but with 1.574 to account for the difference in performance of the thrusters Slew and stabilization last exactly 400 s
16	**t_0**	Start of the maneuver with the input burn time. As soon as the accumulated thruster on-time reaches the specified duration the thrusters close automatically.
17	**t_0+1.1*(burntime)**	Additional, normally superfluous, maneuver stop command; safeguards e.g. against a zero to many in the input duration.
18	**t_0+00:00:01**	The yaw-steering (see #7) is re-enabled. Commanded here, but takes effect only once the satellite is back in attitude hold mode

Fig. 4.30 (continued)

19		Autonomous start of the slew back to the nominal yaw attitude. Slew and stabilization last 400 s
20		Autonomous transition into attitude hold mode at the end of the slew
21	**t_0+00:13:00**	Well after the end of maneuver and slew the additional heaters switched on in step #3 are turned off again
22	**t_0+00:13:51**	The on-board surveillance of deviations of the Sun-vector is re-enabled
23	**t_0+00:13:52**	The table showing the settings of the on-board surveillance is dumped
24	**t_0+00:14:00**	The autonomous transition from attitude hold mode into the mode where payload operations are performed is re-enabled. The transition will occur as soon as attitude deviations are small enough; statistics over a large number of maneuvers shows this normally occurs between four to eight minutes
25	**t_0+00:14:54**	The additional telemetry switched on in step #4 is switched off again
26	Post-condition	Next maneuver can't be made within 60 minutes (unless step #21 is deleted and the timing adapted)

Fig. 4.30 Example of a flight procedure for an "out-of-plane" orbit maneuver, i.e., a burn with a 90° yaw offset

In case a flight procedure can be carried out in real time, i.e., during a contact, every single command will be accompanied by several telemetry checks. Parameters are checked before sending the command and after on-board processing the effects should become visible. Expected values, parameter names, and pages are embedded in the procedure. An example is shown in Fig. 4.31.

The flight procedures must be developed well in advance and be tested either on the satellite itself, on a real-time test bed, or on a software simulator.

4.2.6.2 Example 2

The second example deals with part of the evaluation of star tracker data as collected during the LEOP of GRACE (d'Amico 2002, but see also footnote 2 in Sect. 4.2.2). The measured attitude quaternions showed erratic behavior soon after the star tracker had been switched on with a one-orbit periodicity and each time lasting about 5 min (see Fig. 4.32).

…	Previous part	see Fig. 4.2.1
		Verify telemetry ASD00367 PSI_FROM_QUAT = 0.0 rad Page = AOC5505A
15	If TLM OK then	Slew to -90° is commanded. Slew and stabilization last exactly 400 s
	Wait 400 s then	Verify telemetry ASD00367 PSI_FROM_QUAT = -1.574 rad Page = AOC5505A
…	Following part	see Fig. 4.2.1

Fig. 4.31 Example of telemetry verification during a station contact. Such checks may be part of a procedure as shown in Fig. 4.30, when it is completely or partly carried out in real time

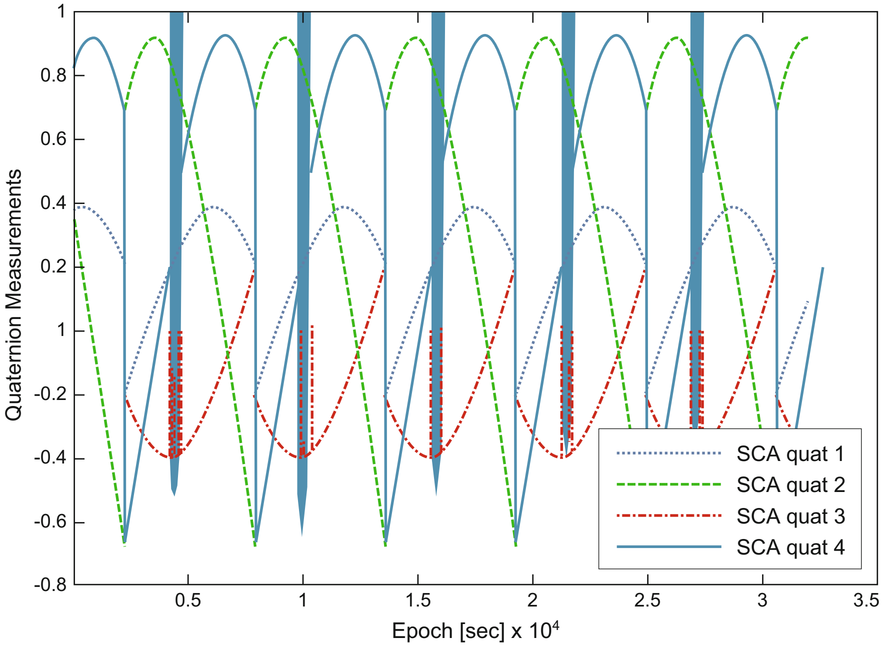

Fig. 4.32 [© (d'Amico, 2002)] The measured star tracker quaternions on GRACE displayed strong disturbances during the first days of LEOP. These recurred each orbit and lasted about 5 min each. Note that the discontinuities (e.g., around epoch 0.25, 0.75 etc.) are no error, but caused by a parity switch; changing the sign of all four components does not change a quaternion

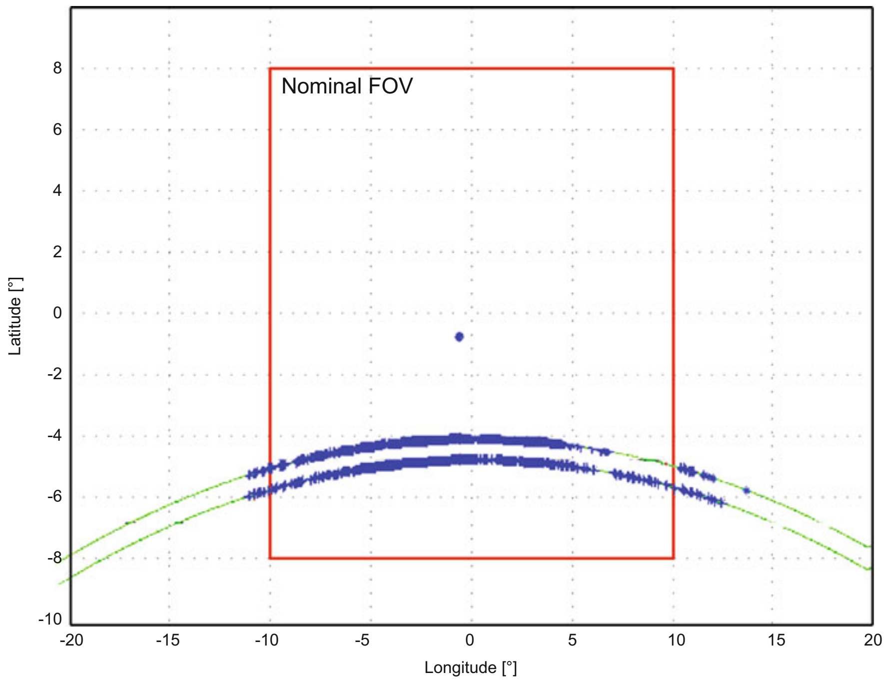

Fig. 4.33 [© (d'Amico, 2002)] The path of the Moon through the star tracker's FOV is shown for two consecutive orbits (*green line*). The *blue marks* denote the points where the solution became invalid. The nominal FOV is shown by the *red lines*; it is *rectangular* due to the baffle in front of the camera

The activity of the attitude thrusters and hence fuel expenditure increased considerably during and immediately after these disturbance periods. The cause could soon be identified. The star tracker started to deliver invalid quaternions as soon as the Moon was in, or close to its FOV (see Fig. 4.33).

The conclusion was twofold: in the first place it was seen that it is necessary to switch to the redundant star tracker as soon as the Moon enters the FOV of the prime and in the second place it was clear that the sensitivity for stray light extended over an area ~30 % larger than specified (see Fig. 4.34).

This strategy was implemented and then used for the entire mission. In the late phases of the mission the FOV had to be enlarged even further, because the sensitivity of the star trackers to stray light worsened over the years. Finally, it should be noted that a switch to the redundant star tracker is only possible if that is not blinded by the Sun.

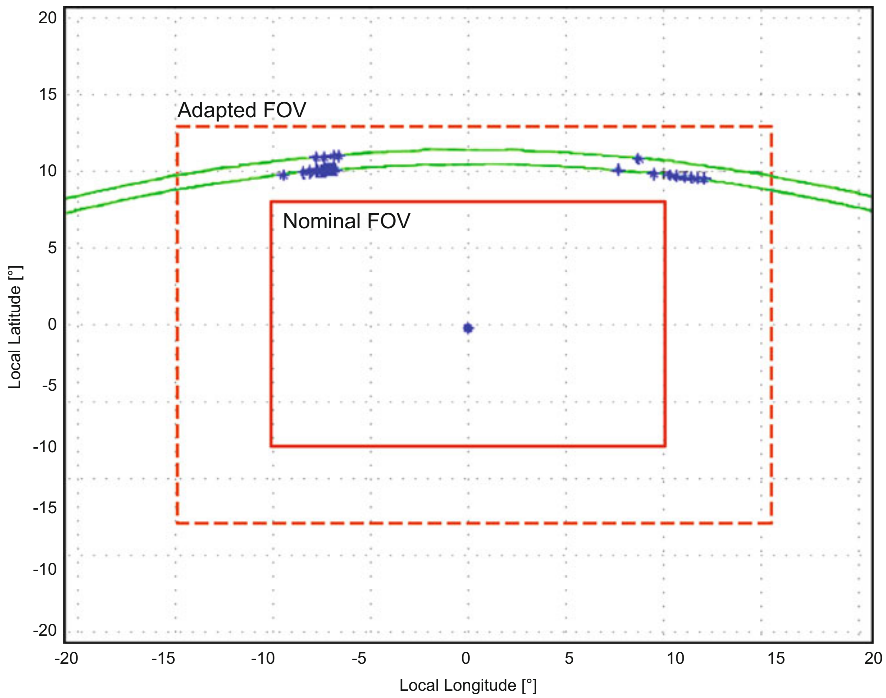

Fig. 4.34 [© (d'Amico, 2002)] Two orbits a day later than the ones shown in Fig. 4.32. *Symbols* are the same as in Fig. 4.32. A switch to the redundant star tracker is commanded from this moment on as soon as the Moon enters the enlarged FOV of 30° × 26°. Note that on the *right-hand side*, where the Moon leaves the FOV, an additional safety margin of ~1° is implemented

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Chapter 5
Mission Planning

5.1 The Planning Problem

Christoph Lenzen

5.1.1 Introduction

Mission planning plays a key role in the operations of a spacecraft, as it ensures that all resources are available and used to an optimal level and the goals of the mission are achieved. There are various ways how to define the exact meaning of mission planning, and different operations centers throughout the world use different definitions.

Within this book, we will consider mission planning as the task of preparing, organizing, and planning all relevant activities that happen during the mission, on board as well as on ground. It is therefore distinct from mission analysis or mission preparation tasks, which serve to analyze and define a mission beforehand and provide the necessary means to execute the mission.

As such, the responsibility of mission planning is to deliver the plan in form of a timeline or so-called Sequence of Events (SoE) right in time for the relevant activities (e.g., before uplink to the spacecraft). This might happen only once (e.g., the actions performed by the lander on an asteroid) or very frequently (e.g., an earth observation satellite fed with customer orders). The plan has to be conflict free, i.e., can under the given on-board and ground constraints be executed on the spacecraft and on ground without any errors. In addition, the timeline should maximize the usage of available resources and ensure that the goals of the mission are eventually met.

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The overall scope of planned activities that falls into the responsibility of mission planning varies a lot from mission to mission. It can range from a very limited responsibility, e.g., planning only simple transmitter switching for ground station contacts, to almost full responsibility for all spacecraft commands including ground station scheduling.

In this chapter, we will introduce the key concepts for designing and operating a mission planning system and present some examples of planning systems for currently active missions at the German Space Operations Center (GSOC).

5.1.2 General Overview of a Mission Planning System

Since the requirements to a Mission Planning System (MPS) and the tasks that it shall perform vary considerably between different missions, there is not one general planning system that can be used for all different needs. In practice, depending on the nature of the planning problem at hand, one will encounter a wide variety of planning systems, ranging from very large software systems to small mission planning components that might be put to work by a human mission planner.

They can therefore be categorized by their degree of automatism, ranging from fully automated planning software without any necessary user interaction to a completely manual planning process. In most cases the system is somewhere in between these two extremes: A human mission planner is supported by GUI-based software tools, which algorithmically create some aspect of the desired plan and check the overall consistency in the end. This is usually the most economical solution.

An example for a fully automated concept is the combined TerraSAR-X/TanDEM-X (Krieger et al. 2007) mission planning system (Maurer et al. 2010), which creates a timeline including the required telecommands for every payload-related activity on the two satellites and also distributes the generated information to the affected parts of the ground segment.

On the other hand the mission planning system for the GRACE mission (Braun 2002) consists of a small algorithm that is triggered by a human mission planner and creates only the sequence of commands necessary for the downlink transmitter switches. It is therefore an example of a software-assisted planning concept.

The ISS mission planning (see Sect. 5.3)—as a third example—is completely depending on manual creation of the timeline, which is only fed into software tools to display it and to allow further processing.

Another aspect by which mission planning systems can be characterized is the periodicity of the planning process. Some missions have limited duration and the timeline might be created only once before the execution of the mission, e.g., for a lander or planetary probe mission of limited length. For others, a predefined timeline might be adjusted to the current situation in regular intervals, e.g., once per week. Some earth observation satellites are an example where the amount of planning required and frequent customer requests make it necessary to limit the planning horizon and repeat the process very often, e.g., once before every commanding possibility of the spacecraft or even every time new input is available.

The periodicity of the planning process defines the way the timeline is generated. Here we distinguish between three timeline generation modes:

Fixed Plan

This type of mission is planned during the preparation phase from launch until end of mission. There is either no need or no possibility to update the plan during execution. For this approach to work, the mission objective and the approach to reach it need to be known exactly in advance. It also leads to a quite conservative estimation of the available resources.

On the other hand, the plan can be optimized by using sophisticated algorithms with high runtime durations of even days or weeks and by multiple iterations of the results with the customer (usually the scientific community).

Repeated Rescheduling

It allows rescheduling on a regular basis, e.g., each time before a LEO satellite is visible by an uplink station. It allows ingesting updated information about resources, orbit events etc. and new tasks, which need to be performed, in particular new planning requests of the customer.

Major modifications of the timeline are the baseline, especially when trying to optimize the result. Optimization, however, is limited to fast running algorithms. Another drawback is that the user has to rely on the decisions of the automated scheduler: when ingesting a new order, the user will not know about modifications in the timeline until the succeeding rescheduling has been performed. He might also not have time to overrule the decisions of the scheduler, because rescheduling usually is performed just in time.

Incremental Scheduling

The operational concept of this type of mission resembles a booking system. Starting point is an empty timeline, which is filled with each input it gets. In particular when a planning request is sent to the system, the algorithm checks what other requests would need to be (re)moved and send the information back to the user. The user then may decide whether to accept these modifications or to discard the current request.

Of course there are a lot of variations on automatic decisions; nevertheless the incremental scheduling concept is designed to keep the modifications of the existing timeline as little as possible. One will prefer this type in case the timeline shall remain stable or in case the user requires full control over the timeline. Algorithmic optimization, however, is even more restricted due to the performance requirements of the system and due to the fact that optimization usually does not imply minimal modifications on the timeline.

A commonality of most mission planning systems is that they form a central part of ground segment operations, possibly having interfaces to many other ground

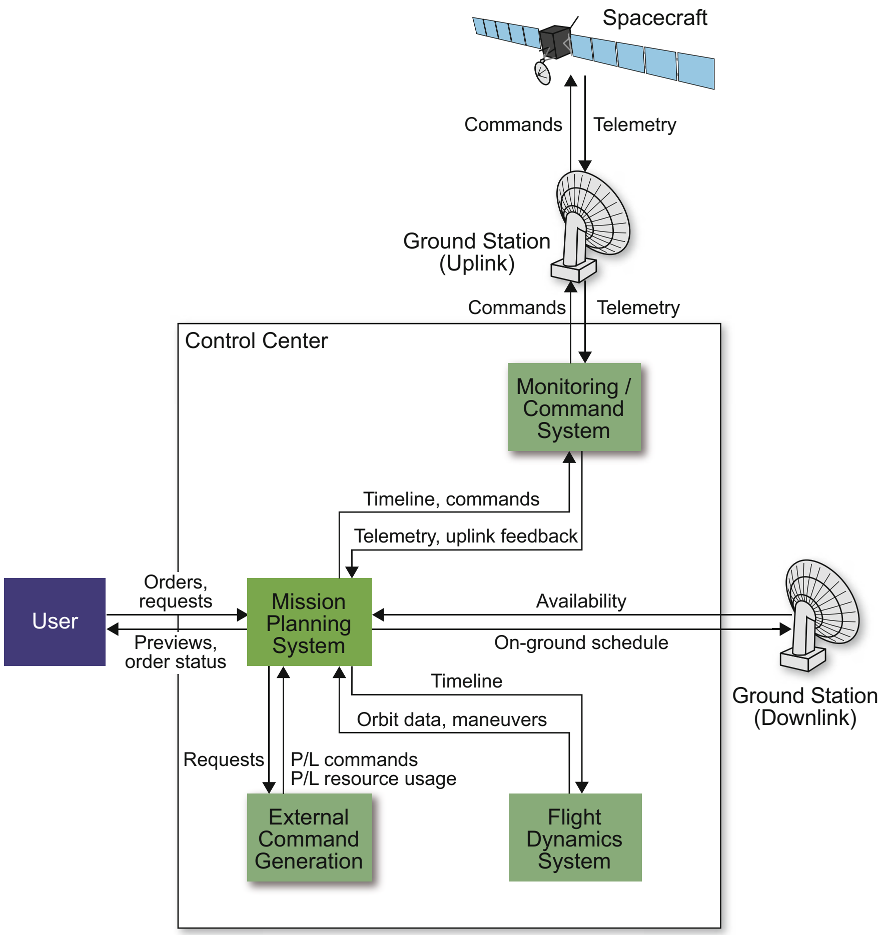

Fig. 5.1 Possible MPS interfaces

segment systems, both inside and outside mission operations. Typical interfaces of a mission planning system are depicted in Fig. 5.1. They comprise input interfaces for incoming planning requests and updated external information like orbit and maneuver information, payload configuration updates, etc. Often the availabilities of external resources have to be fed into the planning system, like ground station availabilities and booking times or astronaut availabilities. Sometimes additional external information will influence the planning result and needs to be fed into the system, for example, the cloud forecast might be used for optimized selection of optical images or evaluating optical downlink opportunities. A mission planning system might already create spacecraft commands by itself, so interfaces to the spacecraft control systems are needed to transfer the created commands and possibly receive feedback whether they were uplinked to the spacecraft and whether the telemetry indicates that they were executed on board. Finally, the output of the planning process needs to be distributed to various customers, e.g., the timeline could be published for the operations personnel, ground stations could be informed which downlinks to expect, and archive systems could be fed with the planned activities. Some mission planning systems offer immediate feedback to customers, which can control the feasibility of their requests, the planning status, and possible alternatives to choose from.

Since different missions impose different requirements, the composition and the size of a mission planning system greatly varies from mission to mission. Completely monolithic specialized mission planning applications would therefore need to be re-implemented for every new mission from scratch again. Instead, it turned out beneficial to maintain a set of loosely coupled generic tools as building blocks that can be extended and tailored to support the mission at hand. This allows for greater flexibility and reusability while allowing an easier development in a team of people. A prerequisite for this is to maintain the MPS-internal interfaces as strict and stable as possible to prevent tedious MPS-internal adaptations for different missions. A general way of expressing the planning problem at hand is needed that can be adapted to the most common planning problems. For this purpose a generic planning language is required; one example is described in more detail below.

One of the components that is required in such a modular MPS is usually a central database that holds the planning model, i.e., the objects which are to be scheduled. Also the latest versions of the timeline are stored in the database. All other components can then interact with the model and perform the necessary manipulations. Ingestion modules will import new data into the model, like planning requests, input from flight dynamics, external resource states, etc. A scheduling engine can be used to generate the plan algorithmically by fulfilling as many requests as possible while considering all constraints and resource states. Various exporting components will access the final plan and produce the desired output formats, like SoE files, command files, or downlink information files for the various ground stations. GUI components can visualize timelines and the current state of planning to the user and allow manually modifying the SoE or start automatic planning tools. Automation components might be included which trigger the planning process upon certain events. As an example, the components of the combined TanDEM-X/TerraSAR-X MPS are depicted in Fig. 5.2. Almost all of the abovementioned components are realized in this system. For the incremental planning concept however a message driven approach outperforms communication via a central database, thus a message passing component must be added.

5.1.3 Techniques for Timeline Generation

5.1.3.1 General Considerations

As mentioned above, there are several characterizations of a mission planning system, which all drive the design of the integration of the planning system into

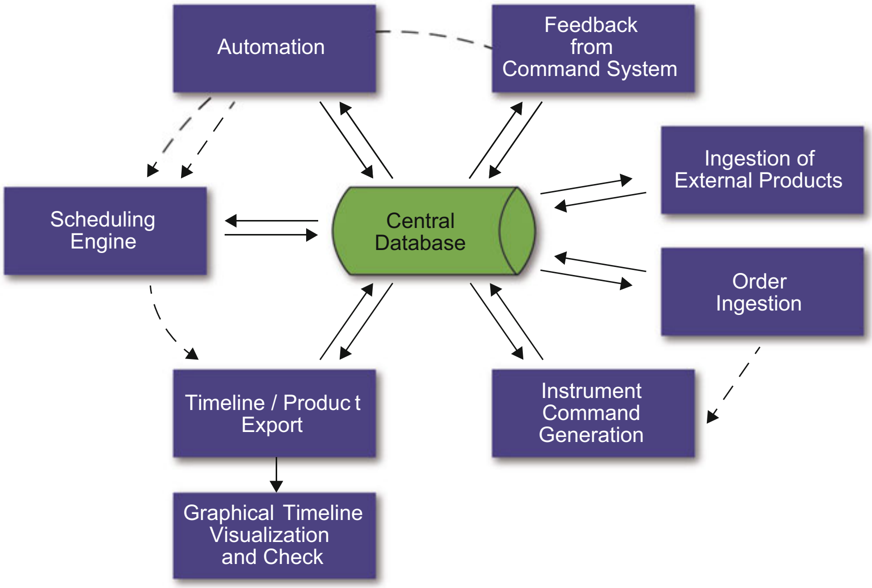

Fig. 5.2 Components of the TerraSAR-X/TanDEM-X MPS

its environment. When looking at its core, however, the objectives of the timeline generation process are the same:

- *Feasibility* and *Safety*: Each subsystem must be able to execute the timeline (feasibility) without being exposed to inacceptable high risk (safety). In particular, the timeline shall not rely on on-board safety mechanisms of a spacecraft. For example it must be avoided to command a timeline which leaves a high energy consumer switched on at the end of the commanding horizon, even if it is to be expected that a later uplink or an on-board safety feature is expected to switch it off at the proper time.
- *Benefit*: The timeline shall serve as many mission goals as possible.
- *Traceability*: In most missions, the user wants to understand why the timeline looks like it does. Since the decisions of what planning requests to include into the timeline are made during the planning run, evidence of it must be supplied by the mission planning system, too.
- *Performance*: The timeline generation process must be sufficiently fast.

Whereas feasibility and safety should be given the highest priority for all missions, the emphasis on the other three objectives may vary significantly in between different spacecrafts and missions. Even during the lifetime of one spacecraft, the goals of the mission may change, e.g., because an instrument of the spacecraft decreases or breaks down, because an additional satellite is added to the existing system (as happened with the TerraSAR-X/Tandem-X satellite twin) or because the user of the satellite changes [as happened for TET-1 (Axmann et al. 2010), which became part of the FireBird mission (Reile et al. 2013)].

In all of these cases, the timeline generation process needs to be adapted and in some cases this has to be accomplished fast. Thus another objective must be defined for the timeline generation process:

- *Flexibility*: it shall be possible to adapt the timeline generation process to meet the modified mission requirements

Whereas the first four objectives might be served best using an individually implemented system, possibly based on a small set of core components, the need of flexibility requires a full generic tool suite, which can be adapted to each mission's needs. A most welcome side effect of such a generic tool suite is that in the long term it can save a lot of manpower.

5.1.3.2 GSOC Modeling Language

The GSOC modeling language is one example of a system which allows modeling the planning problem of a typical spacecraft mission. When modeled properly, a conflict-free timeline will meet the main objective of feasibility and safety. Therefore a modeling language must on the one hand allow sufficiently accurate means to represent the real world and on the other hand it must remain descriptive and rather easy to apply (in order to avoid modeling faults) and of course it also must be manageable by a scheduling engine. It turns out that the following basic elements and features supply a good trade-off in between these three goals; see also (http://www.dlr.de/rb/Portaldata... 2010).

Timeline Entry

A timeline entry describes when to execute a certain task. Consequently the properties of a timeline entry are its start time and its duration.

Task

A task represents something that may be executed some time, e.g., one telecommand or a procedure, i.e., a set of telecommands which are executed in a fixed order. A task is considered to be *scheduled* in case there is a corresponding timeline entry for it. On task level the following properties are defined

Minimum *and* Maximum Duration *of a Task*

These values specify the allowed durations of timeline entries of this task, as illustrated in Fig. 5.3.

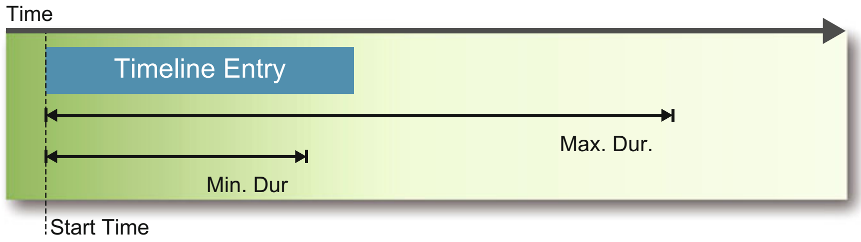

Fig. 5.3 A timeline entry consists of a start time and a duration. The duration can be restricted via a minimum and a maximum duration value

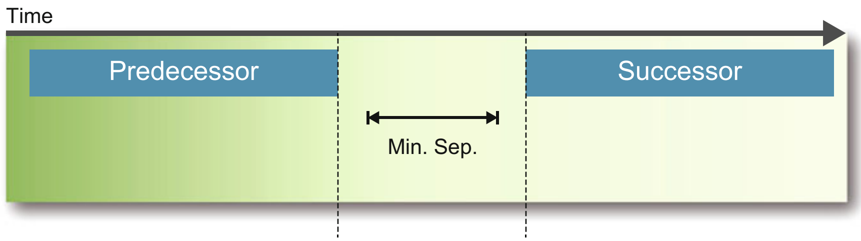

Fig. 5.4 Two timeline entries, separated by a minimum offset in between the end time of the predecessor and the start time of the successor

Time Dependency *with Other Task*

This specifies a minimum mandatory temporal separation in between the timeline entries of two tasks. It can be defined, whether the start or the end time of the timeline entries are used as reference times, as shown in Fig. 5.4.

Also a maximum allowed separation may be modeled that way by swapping the tasks and using a negative minimum separation.

Timeline

A timeline is a set of timeline entries, as shown in Fig. 5.5.

Groups

A group consists of an arbitrary number of tasks ("child tasks") and groups ("child groups"), as depicted in Fig. 5.6. In addition it is defined how many of its children must be scheduled in order to consider this group to be scheduled itself. This allows to collect tasks, which belong together, i.e., which the algorithm should treat as a unit.

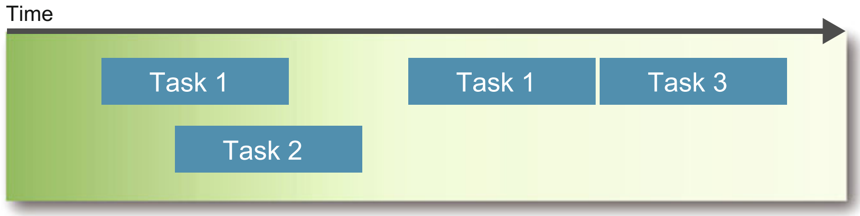

Fig. 5.5 A timeline consists of timeline entries for one or more tasks

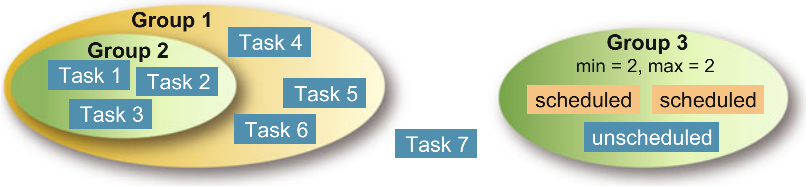

Fig. 5.6 Group 1, task 7, and group 3 are top level; group 3 requires 2 elements to be scheduled

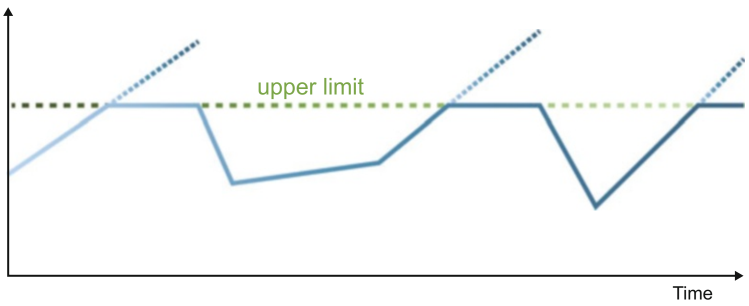

Fig. 5.7 This fill level profile might represent a simplified battery model: when the capacity is reached, the surplus supply gets lost

Constraints on group level:

- Minimum number of children which shall be scheduled
- Maximum number of children which shall be scheduled

Resources

A resource consists of a scalar time profile (*fill level*). Optionally capacity limits can be attributed to a resource: whenever these limits would be exceeded, the surplus is cut off (*lost values*), as shown in Fig. 5.7.

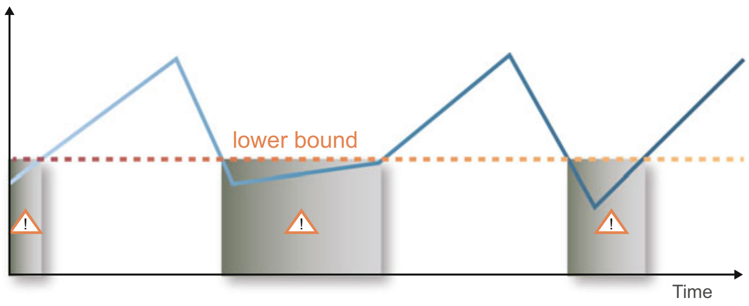

Fig. 5.8 The fill level causes a conflict, when it falls below the lower bound

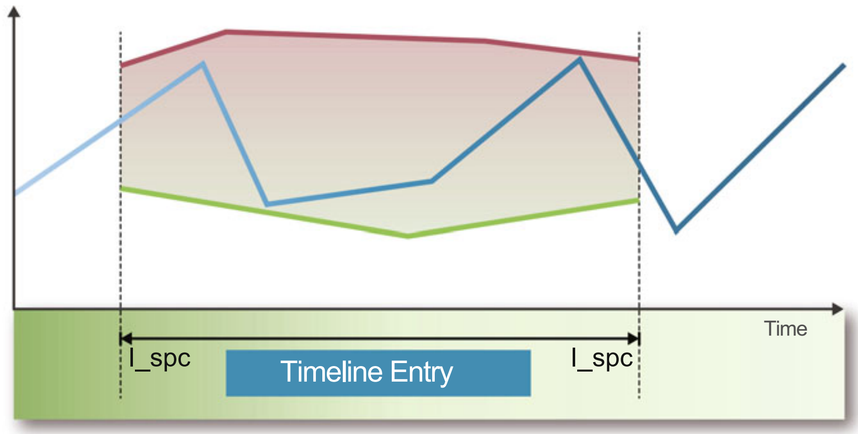

Fig. 5.9 The resource (*blue*) is constrained by upper and lower boundaries (*red*, *green*). The interval where these boundaries apply is derived from the timeline entry via the interval specification (I_spc)

In the following constraints and operations on resources and in between resources and tasks are described.

Resource Bound
A resource bound specifies a globally defined time profile, which the resource's fill level must not exceed, as shown in Fig. 5.8.

Resource Comparison
A resource comparison specifies a local resource bound linked with a task's timeline entry. The resource bound does not necessarily have to coincide with the duration of the timeline entry, but can have a start and end offset with regard to it, as depicted in Fig. 5.9.

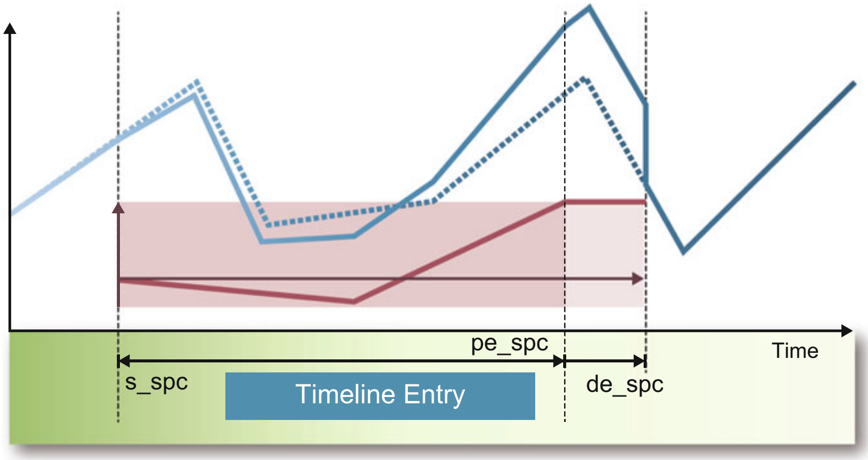

Fig. 5.10 The timeline entry has an active profile (*red*) starting at s_spc and ending at pe_spc, which influences the resource (*blue*) not only during the time frame defined by the duration of the active profile, but also during the extended validity period, which ends at de_spc

Resource Modification

A resource modification specifies how a task's timeline entry modifies the resource. The modification is defined by a change profile, which is mapped to the time axis by adding an offset start (*ActiveProfileStartOffset*, s_spc) to the start time of the timeline entry and which is cut off at the end time, similarly specified by an end offset (*ActiveProfileEndOffset*, pe_spc). However, the modification of the resource can also last longer than the end of the active change profile. In that case you need to distinguish between the end of the active change profile (*ActiveProfileEndOffset*, pe_spc) and the end of the dependency (*Dependency-ProfileEndOffset*, de_spc): In case the dependency end is greater than the active profile end, the value of the profile at active profile end is extended until dependency end, as shown in Fig. 5.10.

For the special case of accumulating resource modifications like power consumptions the dependency end needs to be set to ∞: After being consumed, the power which was e.g. extracted from a battery for the duration of the timeline entry is never available again.

Suitabilities

A suitability is a special utilization of the resource concept. It is essentially a resource which is used to model the benefit to schedule a task at a given time and thus supplies information about when to prefer the execution of a certain task. A suitability therefore cannot cause a conflict, but it quantifies at what time a task's timeline entry results in what benefit. The benefit of a timeline entry is derived from a resource's fill level around the considered timeline entry,

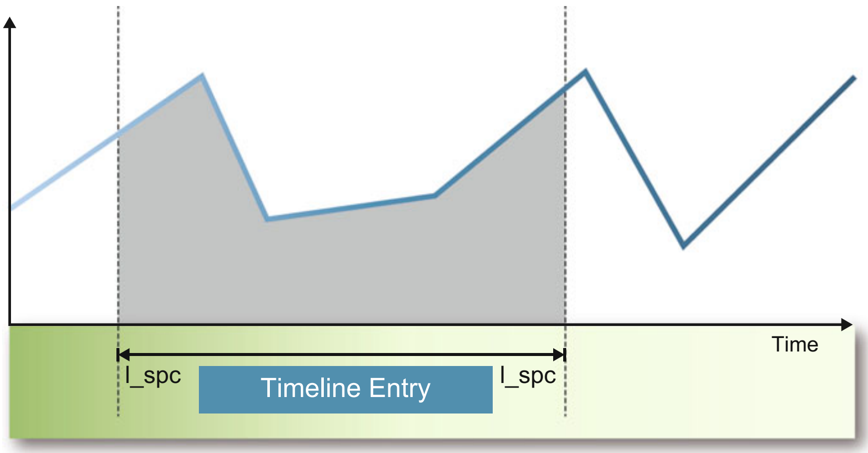

Fig. 5.11 The benefit of scheduling a given timeline entry at a given time can be quantified by a suitable mathematical operation (e.g., the maximum, the integral) applied to the corresponding suitability profile during the time interval defined via the time interval specification (I_spc)

by taking either the integral, maximum, or minimum of the fill level, as depicted in Fig. 5.11. An algorithm may prefer timeline entries with maximum benefit or it may even try to optimize the sum of all benefits of the whole timeline.

5.1.3.3 Application Examples of the Modeling Language

In this section we present a few examples how to apply the above-presented basic building blocks in order to illustrate the flexibility of this modeling language.

Opportunities, e.g., Ground Station Visibilities

In order to avoid scheduling downlinks outside ground station visibility, we define a resource with fill level = 1 wherever the ground station can receive data from the satellite, 0 otherwise, as shown in Fig. 5.12.

Each downlink task on the other hand is given a lower bound comparison, which checks that the opportunity resource has value greater or equal 1.

Equipment Resources, e.g., Downlink Antennas of a Certain Ground Station

In order to model the antenna availability of a certain ground station, we define a resource with

- Initial fill level = 0, representing the number of antennas in use

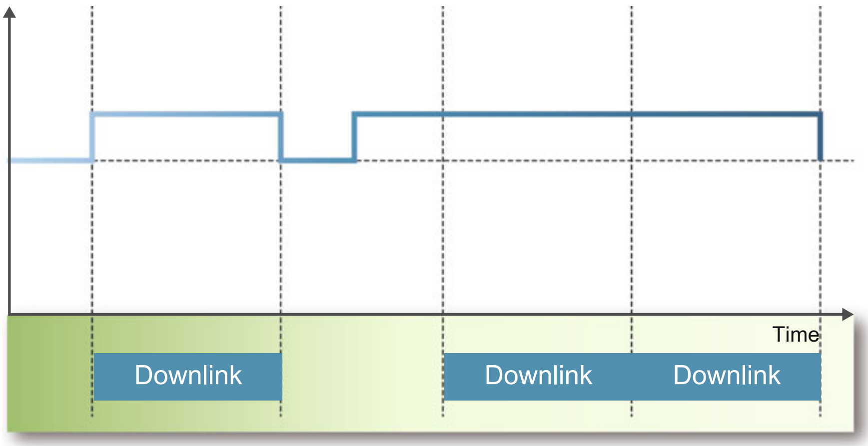

Fig. 5.12 The fill level of the opportunity resource (*blue line*) indicates where downlinks may be scheduled

- Upper bound = number of available antennas

Whenever a downlink for one satellite shall be scheduled, we also need to allocate one antenna of the respective ground station. The downlink therefore increases the fill level of the antenna availability resource. Wherever the number of available antennas is reached (= upper bound), no further downlink may be scheduled anymore, as depicted in Fig. 5.13.

Renewable Resources, e.g., Battery Discharge Level

In order to model the battery of a space craft, we define a resource with

- Initial fill level = 0
- Upper limit = battery capacity
- Lower bound = 0: you cannot take energy from an empty battery

Besides we define resource modifications:

- For consumer tasks, we define a resource modification with
  – Change profile defined in the interval [0, ∞], usually with constant negative slope
  – ActiveProfileEndOffset = 0, which means that the change profile is only considered in the time interval of the timeline entry
  – DependencyProfileEndOffset = ∞, which means that the final consumption at ActiveProfileEnd applies to the whole future (accumulating resource consumption).

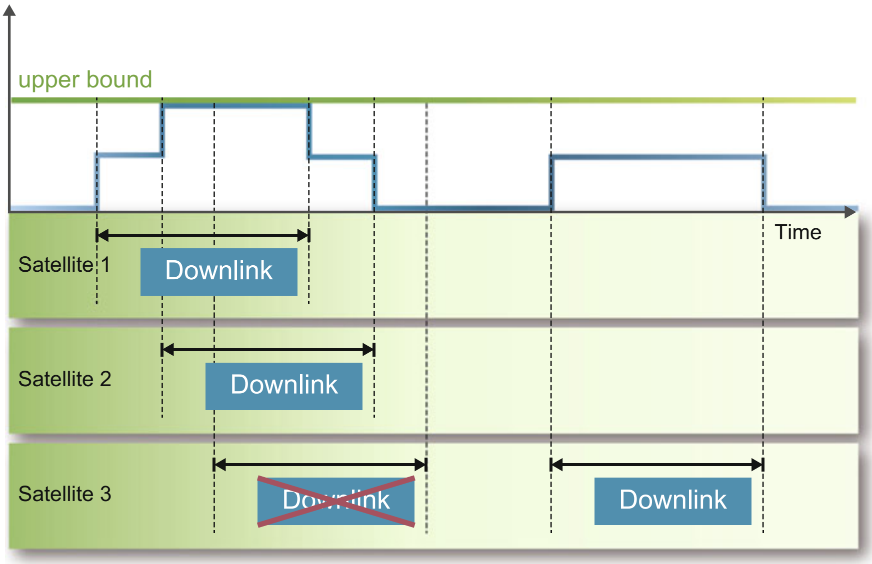

Fig. 5.13 Scheduling a downlink increases the number of antennas in use, including an offset for preparation and cleanup

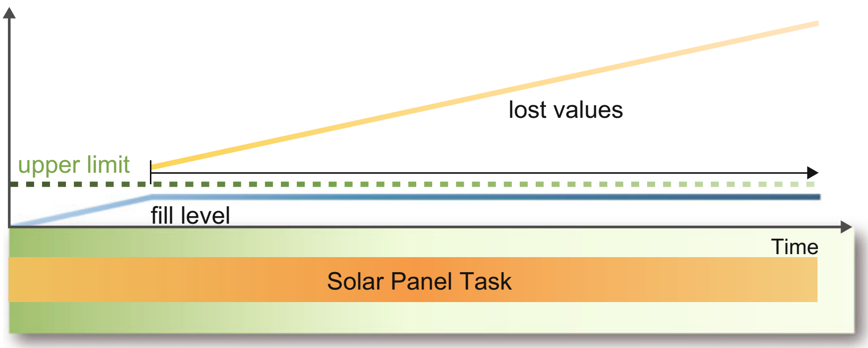

Fig. 5.14 Without satellite activities, much energy is lost, because it cannot be stored

- As supplier task, we define the solar panel task with a similar constraint, but with positive slope on the change profile. This task is scheduled all the time where the solar panels collect sunlight.

In the beginning, as soon as the battery is full, the surplus power supply is lost, as shown in Fig. 5.14.

However, the more consumer tasks are scheduled, the less power is lost, as illustrated in Fig. 5.15.

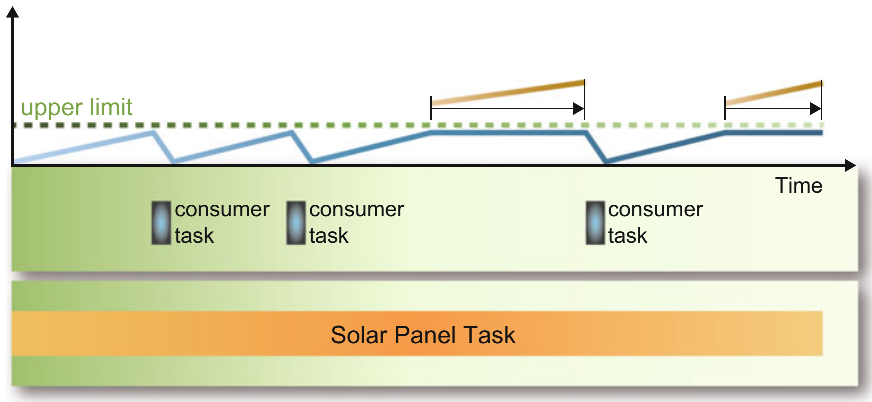

Fig. 5.15 With scheduled consumers, only little energy is lost

Gliding Windows (in Between Equipment and Renewable Resource)

Thermal constraints usually are too complex to be modeled directly. However, they may be described in constraints like "Don't turn on the instrument for more than 10 min per orbit." In order to model such a constraint, we define a resource with

- Initial fill level = 0
- Upper bound = 600 [s]

For each task, which needs the instrument to be switched on, we define a resource modification, with

- Change profile has slope = 1 [per s]
- Active profile end offset = 0
- Dependency profile end offset = 1 orbit

The effect of this modeling is visible in Fig. 5.16. Each scheduled task increases during its duration continually the "clock resource," and this modification is kept for the duration of one orbit, before it is reset to zero. The "clock resource" has an upper bound of 600 s, which corresponds to the requirement to keep the overall "clock count per orbit" below 10 min.

In order to make this model precise, a release slope may be defined which replaces the ending step of the modification profile by a ramp (not displayed in the picture).

Combining Resource Types

In the above examples, we have specified different resource types (opportunity, equipment, renewable resource). Although the resources may differ in having

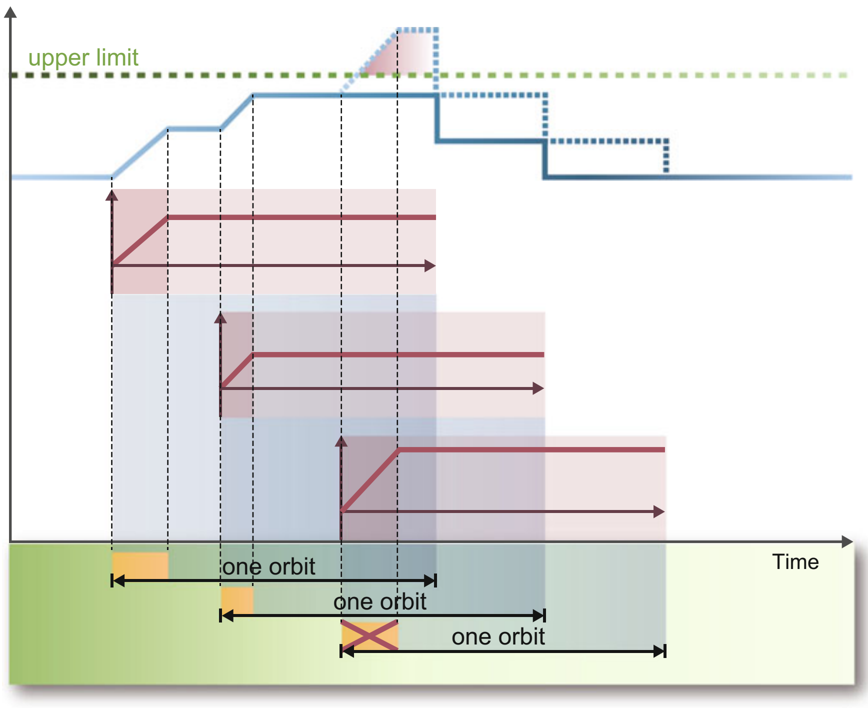

Fig. 5.16 The "one-orbit gliding window" assures that only a limited amount of on-board activities may be scheduled during any time interval of size one orbit

upper and lower limits, the distinction of the resource types depends mainly on the constraints which are defined on them. Given the fact that the GSOC modeling language does not explicitly distinguish these resource types, they can be combined arbitrarily.

For example, one may introduce a resource comparison on a resource, which models the battery's state of charge in order to assure that a certain on-board experiment is executed only if the battery's voltage exceeds a certain level.

Another use case are unavailabilities of equipment: It is possible to schedule an unavailability task, which is given a Resource Comparison with upper bound = 0. This adds a local upper bound of value 0 on the equipment resource during the time of unavailability, so no task may be scheduled there, which needs (i.e., would increase) this resource.

Besides it turns out useful to define all initial fill levels of resources to have constant value = 0 and to avoid modeling any upper or lower bounds on resources. Instead it is recommended to introduce setup tasks, which are scheduled during the whole timeline horizon. A nonzero initial fill level of a resource can be supplied by a resource modification of one setup task and a comparing resource dependency will replace the upper or lower bound of a resource. This way one may prepare different configurations by specifying different sets of setup tasks and easily switch in between these configurations just by scheduling the desired set of setup tasks.

In fact, resource bounds may be implemented using hidden tasks.

5.1.3.4 Scheduling Algorithms

Having translated the mission into the modeling language, we can now focus on generating the timeline entries, i.e., the execution times for the different activities. This can be done by a human operator using interactive software such as GSOC's planning tool Pinta. When using such software, the operator will be presented with possible timeline entries for each task he or she wants to schedule and he or she will be warned against existing conflicts in the timeline. However, this manual approach may be quite work-intensive; therefore it may be a good idea to support the process by supplying subalgorithms, which the operator may apply to certain tasks or groups. This approach can be enhanced by more and more complex algorithms, until the only remaining task of the operator is to start the algorithm and check its result. This kind of planning has been implemented within the TET-1 mission (Axmann et al. 2010).

In the end, when the same type of scheduling is required again and again or when the dependencies are too complex for a human operator to survey, the operator may be excluded completely from this process, ending up in a fully automated scheduling system. An example for such a fully automated planning system is described in Maurer et al. (2010). In this example, a priority-based Greedy algorithm (Korte and Vygen 2005) is used, i.e., the existing planning requests are considered one by one in descending order of their priorities and included into the timeline if this is possible without causing a conflict. However, there exist more sophisticated algorithms, which perform global optimization on a given quality criterion. If such an alternative algorithm was implemented and proved to be superior to the existing one, exchanging the algorithm would be quite easy as long as both algorithms use the same model.

The question of how much automation should be implemented and how much should be left to a human operator depends on the kind of mission.

Fixed Plan

For less complex instances of this type, it may be best not to write an algorithm at all but to let an expert generate the timeline manually. In case the planning problem has already been modeled, this manual timeline generation may be supported by graphical planning tools, which suggest conflict-free timeline entries for selected tasks and highlight existing conflicts and their sources in order to support fast manual conflict resolution.

For less complex but voluminous instances of this type, the manual approach may be accelerated using generic (i.e., off-the-shelf) algorithms. In this case a simple configuration will be sufficient to generate a first version of the timeline, which thereafter may be checked and improved by the operator.

On the other hand, complex missions of this type may require a highly specialized optimization algorithm. You will not find a generic algorithm for such a mission. However, modeling the problem in the standardized way still has advantages, which may be worth considering:

- Conflict check of the final timeline using generic tools
- Display of the final timeline using generic tools
- Simple integration of the specialized algorithm with existing scheduling software

Repeated Rescheduling and Incremental Scheduling

These operational concepts support ingestion of new information during the mission execution phase. This usually implies that new requirements to the planning process evolve during the execution time. Therefore using a generic algorithm not only saves implementation effort in the preparation phase but also increases the flexibility to react on such new requirements. So unless there are good reasons to implement a specialized algorithm it should be attempted to configure a generic one.

The main challenge of a generic algorithm is to design it sufficiently flexible to cover the typical demands of future problems and to keep it extensible such that the unexpected demands may be covered by generic or project-specific add-ons to the preexisting generic system.

On the other hand the algorithm and its configuration must remain simple. In case the algorithm's functionality is hard to understand or in case it has too many configuration items, it is likely not to be understood by any other than the developer himself, which makes it unlikely to be reused in the future.

The naïve approach would be to define some basic generic algorithms and to extend them whenever new features would be required. Although in the course of time these algorithms will cover most requirements, one will find them very soon hard to understand and to apply. A better approach to extend the basic algorithms is to decompose them into their very basic parts and to allow combining them, as described in Lenzen et al. (2012). This way, small and easy-to-understand "algorithm snippets" are generated, which can be combined in the desired way. Following this approach, an extension of the generic algorithm should result in adding a new algorithm snippet.

5.1.4 Summary

The current algorithms of the GSOC scheduling suite restrict to heuristics, which find good results in rather short time. Using this approach, one may freely combine the features of the modeling language. Implementing an optimizing algorithm on the other hand will require some restrictions on the planning model. For example the lost values calculation of a resource is hard to translate into the model domain of an optimizer. However, one may find approximations of the model, which may be covered by an optimizer.

5.2 Mission Planning for Unmanned Systems

Tobias Göttfert and Falk Mrowka

5.2.1 Introduction

Space missions without human beings on board are often suited for automated planning systems, since the mission objective and duration is usually known beforehand. Additionally the "human factor" is not present and the complexity of the vehicle's subsystems is lower than the one of e.g. a manned space station. There is usually a finite list of tasks that can be planned, depending on the features of the spacecraft, and a finite list of resources which have to be modeled. However, the requirements and operations scenario of non-manned spacecraft missions can vary a lot and no single Mission Planning System (MPS) fits up to now all missions that are flown in one control center. In general, the more complex and more agile the planning problem gets, the more work can and should be invested into automation and a sophisticated software system. For comparatively simpler planning tasks, which have to be performed less often, regular manual creation of the mission timeline is often the more economical solution (see also Sect. 5.3). However, in both cases, a set of common software tools for planning and scheduling is highly beneficial.

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5.2.2 Mission Planning System Example

As an example for the application of the generic scheduling language which was presented in Sect. 5.1, the combined MPS for the TerraSAR-X and TanDEM-X missions (TSTD-MPS) at the German Space Operations Center (GSOC) shall be described (Lenzen et al. 2011). This system is at the moment the most evolved and feature-rich MPS at GSOC, having grown from a single-satellite, single-mission automated planning system into a dual-satellite, dual-mission planning system that takes various correlations between the spacecrafts and the missions into account (Mrowka et al. 2011). The TerraSAR-X and TanDEM-X spacecrafts are described in more detail in Krieger et al. (2007). For here it is relevant to know that both carry a Synthetic Aperture Radar (SAR) instrument that is able to capture either two-dimensional imagery via one satellite or three-dimensional radar-based digital elevation models that make use of both spacecrafts simultaneously in an interferometric mode.

Therefore, the two satellite missions need to be considered as tightly linked in terms of planning.

Additionally, the mission is operated in a public–private partnership, where several parties can place planning requests for radar imagery, which all have to be collected at the MPS side.

In total, the TSTD-MPS has to model more than 300 resources and their constraints which are associated with the two satellites and is responsible for the scheduling of all payload-related activities, both on board and on ground. This means that not only the radar instrument activities need to be scheduled, but additionally mass memory and bus-related activities, like X-Band downlink of the radar data. In addition, an on-ground schedule is created that allows the participating ground stations to control their antennas accordingly, as well as the data processing centers to properly file the incoming payload data. The GSOC generic scheduling language (see Sect. 5.1) provides all the building blocks that are needed to describe the planning problem for this mission and the implementation of a priority-based scheduling algorithm.

For example, the simple addition of one 2D radar image is modeled via a timeline entry of a task with certain duration on the timeline. When scheduled, this timeline entry affects several resources, e.g., the power or time usage of the radar instrument, which of course is only possible if none of the resource limits is violated. In addition, scheduling the data take is only possible if there is no danger that the partner spacecraft is illuminated with the SAR instrument during that data take, modeled via another resource. The data take also necessitates file handling tasks and their timeline entries, like creation and deletion of data files for the image, which in turn affect the mass memory resource usage. On top, data downlink tasks have to be scheduled during a ground station contact, which in turns triggers the scheduling of antenna switch-on and switch-off tasks.

Fig. 5.17 Graphical view of parts of a TerraSAR-X/TanDEM-X timeline within Pinta. A short time period (approx. 45 min) and a limited set of activities are displayed on a horizontal time axis. Tasks and resources of the TerraSAR-X satellite (TSX) are shown in *green*, and for the TanDEM-X satellite (TDX) in *blue*. Note how the algorithm schedules instrument activation for the data takes and the switchover of X-Band downlinks when passing over several ground stations. 3D data takes are scheduled in parallel on both satellites

Within the scheduling process, the algorithm looks at every planning request in a defined order, based on request priority, and decides if all these conditions can be met. Only then, the data take is planned together with all additional tasks, and the resources are modified accordingly. In the end, an export tool creates telecommands for the spacecraft from the tasks on the timeline, which are then uplinked and executed. Further data files are generated for the various shareholders on ground. Figure 5.17 shows a simplified example of a TerraSAR-X/TanDEM-X timeline, taken from the GUI tool for timeline analysis, called Pinta, in which some important tasks and resources can be seen in a graphical view.

The planning process within the TSTD-MPS is performed in a regular scheme which is strictly coupled to the ground station contacts for command uplink (repeated rescheduling). On average, this results in a planning run every 12 h, where every time a new version of the timeline with a planning horizon of three days into the future is generated. However, each upload contains time-tagged spacecraft commands only for the next 24 h, still leaving 12 h of overlap from each command upload to the next one. This ensures that one failed uplink contact will not stop the mission execution.

5.2.3 Considerations on Designing a Mission Planning System

Mission planning systems for nonhuman missions are software systems, where the degrees of automation and human interaction depend on the mission requirements. Therefore creating an MPS is strongly coupled to software engineering. The design of such a system is influenced by the following key factors:

- How to map mission objectives and capabilities to the planning model and planning algorithms
- How to interact with the rest of the ground segment and serve its interfaces
- How to design the system as robust, failure tolerant, and automated as needed
- How to maximize the amount of reuse of existing components

During the mission development phases A to C (c.f. Sect. 2.1), the baseline and the complexity of the MPS are already defined and small changes on the mission design and/or requirements can influence the needed effort on mission planning side considerably.

Therefore it is advisable to analyze the mission design as early as possible also from a mission planning point of view. A number of aspects need to be taken into account, of which a few examples shall be given.

For a mission with several spacecrafts it needs to be evaluated, whether it can be dealt with by using independent planning models for the spacecrafts, or whether they are inherently coupled to each other. For example the TerraSAR-X and the TanDEM-X spacecrafts are inherently coupled in their formation flight because of synchronized data taking for 3D images, and therefore one common MPS is needed because of the common planning model for both satellites and both missions.

The level of intelligence of the on-board software needs to be taken into account, e.g., whether the telecommands available to MPS address high-level or rather detailed low-level functionalities. For TerraSAR-X spacecraft the MPS has to command also instrument activation and instrument standby in several levels, whereas other spacecrafts perform these tasks on their own if data takes are commanded.

It needs to be checked whether the command uplink scheme is very rigid and well defined or whether there is a lot of flexibility involved. In some cases even a constant uplink is possible, as it is the case for geostationary satellites. As example, the MPS for the GRACE mission (Braun 2002; Herman et al. 2012; Tapley et al. 2004) is built on a fixed scheme of weekly planning runs and incremental uplinks of the resulting commands every day, the TSTD-MPS on the basis of specific ground stations contacts.

The flexibility and optimization requirements from the user's side need to be analyzed: It makes a huge difference whether frequent replanning and thus possible changes of the plan are desired or discouraged.

It is crucial to clearly define which resources have to be modeled to what extent. For some resources a detailed model is required; for others approximations are sufficient. For example for the TerraSAR-X spacecraft, memory usage is modeled exactly; the battery level is modeled in a linear approximation; thermal resources are modeled by a simplified heuristic using a predefined time window approximation.

Also the modeling of the ground segment needs to be investigated. It can be built very complex, but often a simplified representation is good enough. For example, shall certain ground stations be preferred for data downlink because of a network connection with higher bandwidth?

Last but not least the complexity and safety of the commanding concept needs to be taken into account, e.g., for the TerraSAR-X spacecraft, for safety and autonomy reasons, every 12 h the next 24 h are commanded to the spacecraft, which needs sophisticated delta-commanding capabilities.

5.2.4 Mission Planning at Various Time Scales

Mission Planning is a task that happens on a wide variety of timescales, from years down to sub-seconds. This chapter shall give an overview about the work of the mission planners and their results on these timescales.

During all of phases of the mission, starting already in the analysis phase A, i.e., *years* before the launch, mission planning needs to be involved in the process. The mission planning team provides consulting and participates in the analysis and mission definition process. It collects mission and user requirements and ideas about the goals of the mission and prepares concepts for planning and scheduling. It defines the design of the mission planning system and the necessary components, which either can be taken from existing components or have to be newly developed. In the end, a concept for the MPS and its operation and the mission planning requirements are created and the MPS is developed and integrated, together with all necessary tests.

Activities that happen in the timescale of *months* are usually major changes in the mission, e.g., major orbit-configuration changes. Other changes might lie in the operational concept, e.g., changes in the uplink or downlink schema, different usage of the payload, and introduction of new requirements. The mission planning team gives consulting to the mission management and ensures that the feasibility of the change is given.

For planning systems that handle stable and predictable scheduling problems, a long-term preview of the timeline can also be given, which covers the timescale of months. Of course, this preview is subject to changes, depending on input of planning requests, but can serve as a valuable tool to enable the coordination between different parties that have influence on the timeline. For example, flexibility in the execution of orbit maneuvers can be used to schedule them at a time that minimizes the disturbance on nominal mission execution; different users can coordinate their requests, etc. Also a coarse preplanning of activities is sometimes performed months ahead.

Normally, the planning horizon of a nonhuman mission is in the order of *days*. Within this time frame, all relevant nominal input shall arrive at the MPS, well ahead of execution time. Those inputs are e.g. planning requests, the ground station availabilities, orbit and maneuver information, or other flight dynamics input. Also planned maintenances fall in this category.

The timeline is then usually generated some *hours* before uplink, depending on the mission planning concept, degree of automation, and requirements on the reactivity of the MPS. This short-term planning can take into account the latest status of flight dynamics data for products with a high requirement on accuracy and a limited time span of validity. The point of timeline creation also determines the order deadline, after which no late planning requests can be considered anymore.

Usually, only during the LEOP phases, activities can be planned *minutes* before their execution time, since only in this phase extended real-time operations with human personnel in the form of mission planning and flight operators are carried out.

The activities that have to be scheduled also happen on different timescales, depending on their nature. Hereby, both duration and required timing accuracy can vary a lot, from days and hours down to the *second* and *sub-second* range. A few examples are maintenance phases, which can last hours and days and usually start at manually chosen times, ground station contacts, which have durations in the range of minutes for low-earth orbiting satellites and need to be scheduled with a few seconds precision, or maneuvers, which usually last a few seconds and also need a precision in the range of seconds.

Most satellite buses provide an accuracy of execution of time-tagged commands in the range of one second, normally sufficient for most activities on board.

For applications that require a more precise control over their execution time, special provisions have to be made: First, the on-board hardware has to support this, usually via GPS-supported time correction features in the affected subcomponent of the satellite, and second, the MPS has to be able to fine-tune the start time and duration of the activity when the latest orbit information with the required accuracy is available. A prominent example is the scheduling of imaging data takes for low-earth orbiting satellites, which require a timing accuracy of less than 1 ms for a ground track accuracy in the range of meters.

Figure 5.18 shows zoom-ins into a timeline of a fictitious satellite mission to demonstrate the various timescales of activities that the MPS encounters.

5.2.5 Conclusions and Outlook

Mission planning for nonhuman missions has to cope with many different types of spacecrafts and operational concepts, leading to a variety of MPS designs. However, the well-known set of plannable activities and their potential frequent repetition usually make some sort of automatic scheduling feasible. The surrounding tools that complete the MPS are then crafted to the automation and operational

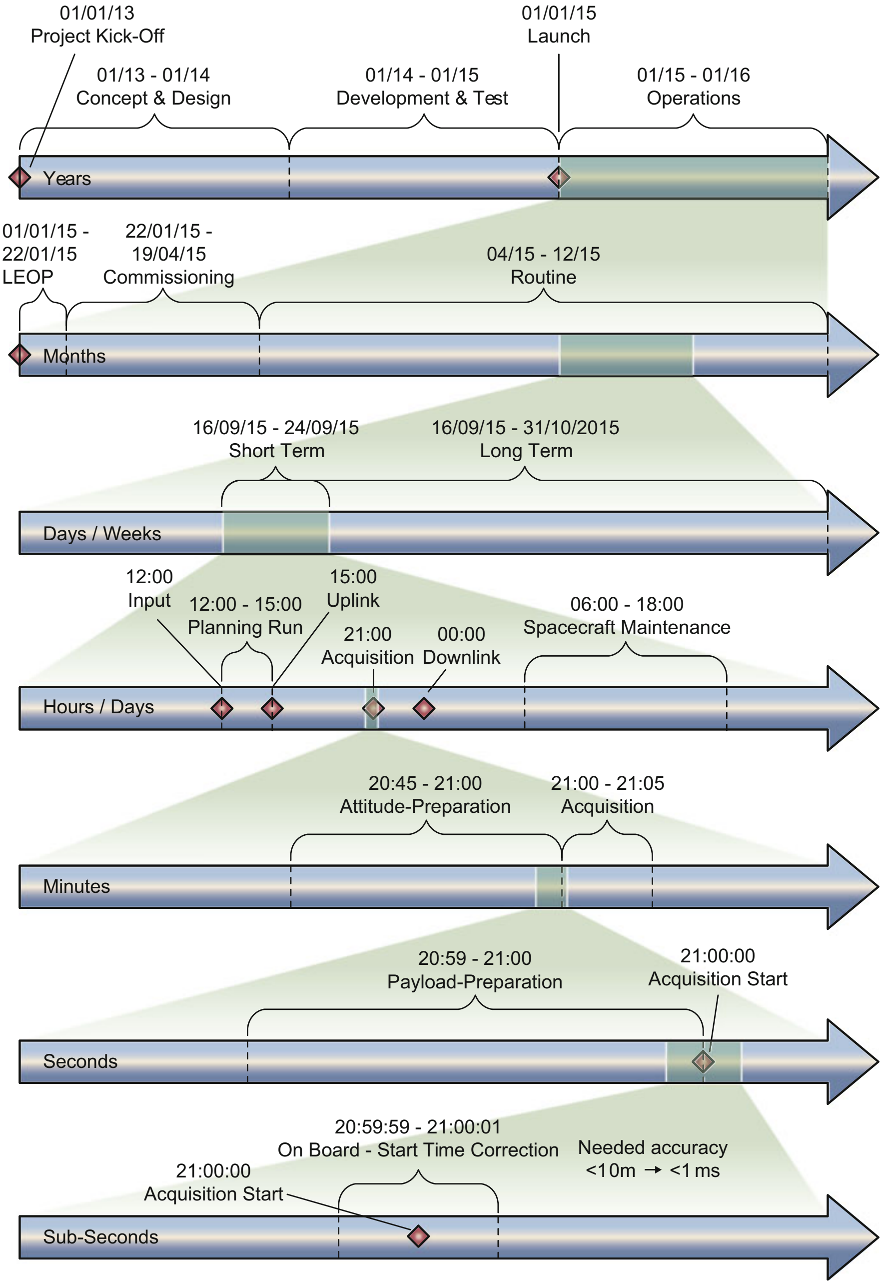

Fig. 5.18 Timeline with example activities of a fictitious satellite mission. The various levels of zoom exemplify the different timescales on which the mission planning processes and scheduled activities take place

needs of the mission. For this, a proven set of planning tools that supports all relevant scheduling functionalities and easy inspection of the resulting timeline is a crucial prerequisite.

Following the development of highly automated mission planning systems and the increasing computational capabilities of on-board hardware, new approaches increase the flexibility of mission planning systems even further to cope with the growing demands of reducing overall reaction times of the satellite system. One example are the investigations for moving parts of the planning algorithms onto the spacecraft, making it possible to invoke last-minute planning processes without contact to ground (Wille et al. 2013; Wörle et al. 2013).

5.3 Mission Planning for Human Spaceflight Missions

Thomas Uhlig, Dennis Herrmann, and Jérôme Campan

5.3.1 Introduction

Planning is a key part of human spaceflight operations, was already part of the first Mercury flights, and has significantly developed ever since. While the first flight plans were text-based tables on paper, today's plans are displayed graphically with dedicated applications, have sophisticated possibilities to include or extract information, and are available via the Internet. In the meantime, the terminology "timeline" is commonly used to describe the schedule of the astronauts and the related ground activities.

In this chapter, planning for human spaceflight operations is described using the International Space Station as a model case. This confined focus might sound like a loss of generality; however, all human spaceflight endeavors undertaken so far cumulate in the ISS project. In that sense, the ISS planning processes, phases, and tools can be considered as the heritage of the American Mercury, Gemini, Apollo, Skylab, Space Shuttle, and the Russian Восток (Vostok), Восход (Voskhod), Салют (Salyut), and Мир (Mir) planning processes. All that experience and lessons learned were used to design the mission planning for the International Space Station.

It needs to be highlighted that a number of planning and scheduling processes and products exist for the ISS in various areas: Planning is being done for example for the station attitude, for robotic tasks, for Extravehicular Activities (EVAs), for consumables, and for critical resources like water, as shown in Fig. 5.19. This article only focuses on the crew and ground activity planning—the process

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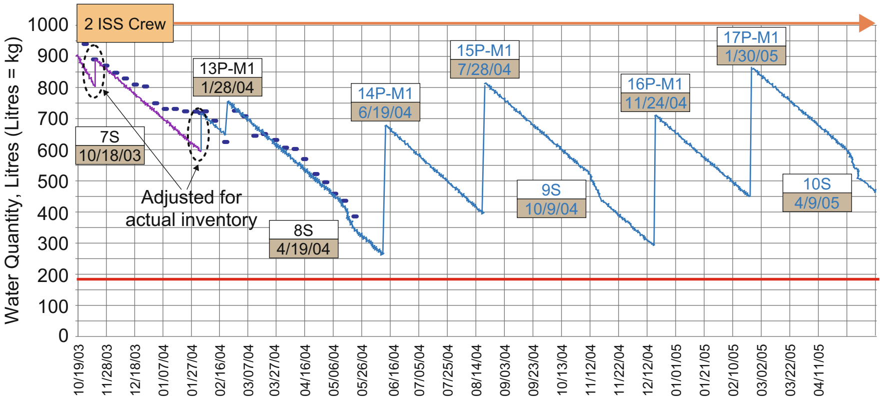

Fig. 5.19 Planning is done in various forms for human spaceflight missions. Here, an example for the on-board water availability planning is given (Reprinted from Kitmacher et al. (2005) with permission from Elsevier)

which is commonly referred to as "mission planning." All other aspects are not covered here.

Although the terminology and the planning theory are similar to mission planning for unmanned missions, the automatization of planning for the ISS is marginal. In principle, it would be possible to model activities, to define exactly the constraints, relationship to other activities, and dependencies of resources, as described in Sect. 5.1. But since the mission goals for ISS operations cannot be broken down to a few science objectives, like it is possible for a satellite (i.e., "earth observation"), the number of resources and conditions to be modeled would be immense. If the goal of a mission would be earth observation only, the conditions could be limited to some parameters concerning the position and attitude of the spacecraft, the orbital constellation with respect to the sun, if sunlight is required for optical pictures. The resources could be the available power, the data stowage situation on board, and the downlink paths.

But ISS mission goals include not only earth observation, but also experiments of almost all scientific disciplines. The experiments might have requirements for special microgravity conditions, temporal constraints on a variety of different time scales, interdependencies to "private" crew tasks (like fasting or blood sampling), priority orders, and even international or political dimensions. Figure 5.20 gives an overview about the most important resources and constraints.

Although it might be possible in theory to produce a complete list of all thinkable resources and conditions, and a state-of-the-art computer system with a corresponding algorithm could predict these parameters continually, the effort to link each activity to its related resources and conditions and thus to model it correctly would outweigh the benefit of an automated planning system. This is in particular

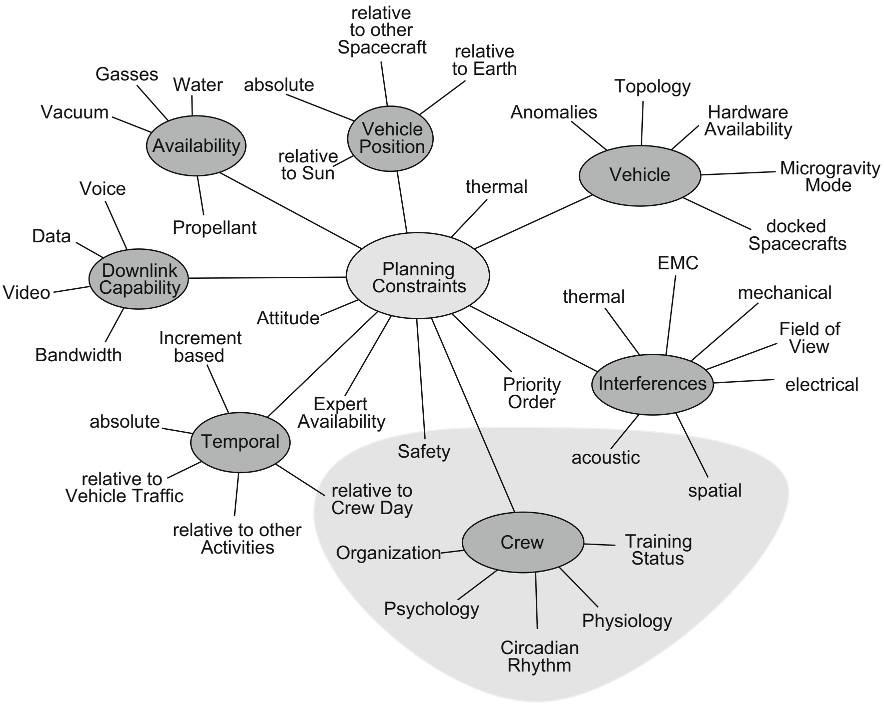

Fig. 5.20 Planning for human spaceflight can be constrained by a variety of factors. The most important ones are shown here; the ones directly linked to human presence are summarized in the *shaded area*. A mathematical modeling of them to support automatic planning algorithms is not feasible

true, since a large fraction of the activities which are on the timeline occur only once.

Also the human factor has to be taken into account, which is difficult to put into a mathematical model. Astronauts have personal preferences, which need to be respected and incorporated into the planning, if possible. There are medical and psychological aspects, which only appear if the integrated schedule is checked by an expert, e.g., there might be concerns to do more than one blood draw per day.

Due to the internationality of the ISS and multiple participants, the planning has also a political dimension, which might influence the final planning product, but definitely needs to be reflected in the corresponding processes.

For those reasons, the human spaceflight planning is still a manual process and involves different teams around the world to accomplish it. The multitude of influencing factors, which can easily change in near real time, also makes the mission planning a highly dynamic endeavor.

5.3.2 Basic Considerations

Planning for projects like Apollo or the early Space Shuttle flights is fundamentally different than planning for the Mir or the International Space Station. If the planning is done for short duration flights, the preparation and the execution phase are clearly separated. The optimization of such plans has to be very high to make best usage of the limited flight time, and due to the abovementioned separation, this goal can also be accomplished (Popov 2002).

For long-term missions like the ISS, which is continuously manned for several years, the plan preparation and the plan execution processes need to be run in parallel, which adds complexity to the planning, but also decreases the need for high optimization, since overall crew time is no longer a limiting factor (Korth and LeBlanc 2002).

However, to better be able to cope with continuous ISS operations, it was decided to break down the Space Station timescale into smaller pieces, which can then better be worked with methods comparable to project management. Those timescales are called "increments" or "expeditions", are driven by the ISS crew rotation, and have numbers assigned; currently (March 2014) the ISS is in increment 38. An ISS increment has a duration of 2–4 months; each Soyuz undocking event triggers the start of a new one. Since two Sojuz crew alternate, this translates into a crew stay on board of approximately 6 months, which is compatible with the time a Soyuz spacecraft is designed for and approved to remain in space under the very harsh environmental conditions, which are described in more detail in Sect. 1.1.

The planning processes are therefore adapted to the increment concept, as described in more detail in Sect. 5.3.5. For historical reasons two increments are always treated together in one combined planning effort—so all the considerations below are applicable for double increments, e.g., increment 37/38.

The elementary planning items in human spaceflight planning are activities (sometimes also referred to as Planning Requests, PR), which are either executed by the ground team or by the astronauts on board. The heart piece of each activity is normally a procedure (or parts of a procedure), which are called Operations Data Files (ODFs). They contain a detailed description of what has to be done using well-defined formats and standards.

These procedures also define the required resources to conduct the activities. To complete the set of planning data required for the scheduling and a proper execution of the activity, some additional info is attached to each planning request, partially detailing the data already contained in the Operational Data File (ODF), and partially adding new planning information like the resource utilization. The relationship between the ODF and the planning request is depicted also in Fig. 5.21.

The planning requests are maintained in a database at each participating control center (see Sect. 5.3.3) and are then fed into the common planning system.

During the planning process the limited ISS resources like crew time or data bandwidth need to be shared between the different international partners. It was

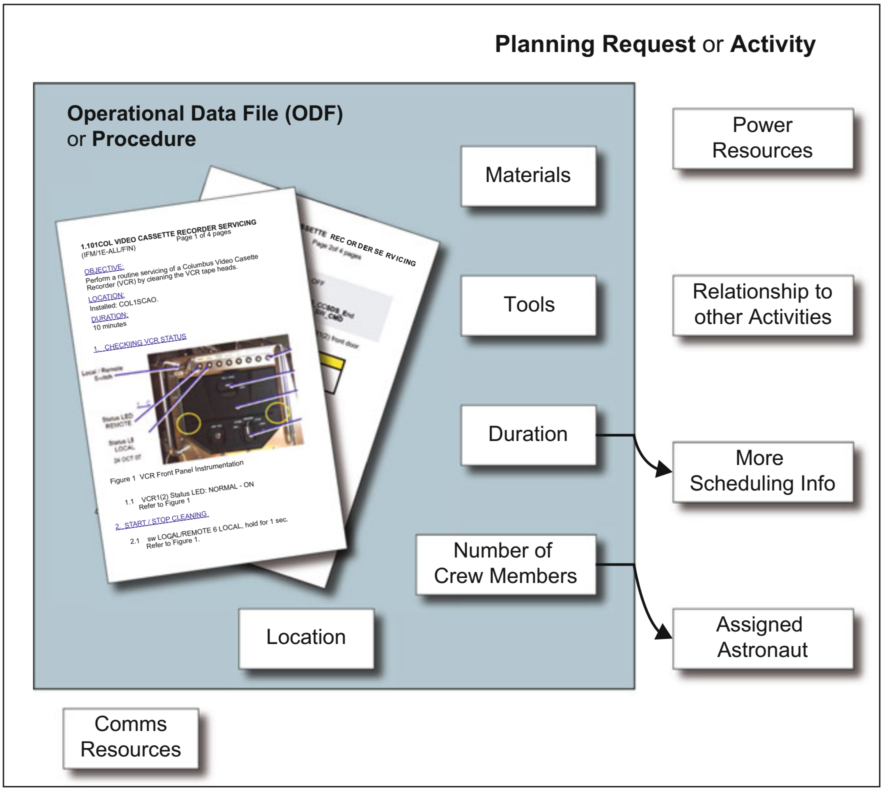

Fig. 5.21 An activity definition or planning request as elementary planning item contains as basis a procedure called Operational Data File (ODF), which defines the actual task to be performed. It contains also various information about the task like the materials or tools required for execution. The procedure is supplemented by some additional information, like the assigned astronaut—often referred to as planning attributes—which complete the set of data required for proper planning of the planning request

agreed that the ratio every partner is eligible to use is directly related to its overall contribution to the ISS program. This leads e.g. to an ESA utilization share of 8.3 % of all common ISS resources, including crew time.

5.3.3 *Planning Teams*

The increment concept allows also to better assign key personnel and to ensure a task and responsibility rotation with a frequency of approximately half a year.

The first steps of defining the content of the increment are done on a management level by every international partner during the strategic and tactical planning phases described further below.

Each ISS control center (see also Sect. 7.1) then facilitates its own planning department, whose experts are assigned for the increments. For a given increment, there are usually a number of individuals involved in the preparation phase. The terminology Long Range Planner (LRP) is used by NASA, ESA calls them Columbus Lead Increment Planner (CLIP), and the Japanese supply the 3PO (Planning, cPs, iPs Officer). We will see later that the preparatory phase does not end with the start of the increment, but the plans have to be fine-tuned again, if the actual execution day comes closer due to the highly dynamic environment of human spaceflight. To minimize the handovers and to facilitate the knowledge which was built up by the planners who worked the preparation phase, in all ISS control centers this near-real-time processing of the timeline is done by the same individuals who already prepared the first draft timelines.

One week prior to execution, the timeline for a given day is handed over from the planning personnel in the backrooms to the Flight Control Team (FCT) on console. Also within each flight control team a planning function is implemented, partially a dedicated position staffed by an expert from the planning group (like the OPSPLAN at the Johnson Space Center in Houston, partially merged with a technical position (like the COL OC at the Columbus Control Center in Oberpfaffenhofen). These real-time planners coordinate and process late formal changes to the timeline and can assist the Flight Controllers in case real-time replanning of activities "on the fly" is required.

5.3.4 Concept of Crew Flexibility

It is part of the human nature that an astronaut prefers to have certain freedom in his working day. Thus the timeline is generally considered as a proposal from the ground teams only, and the crew is generally allowed to deviate from the plan or to adapt the schedule to their individual preferences.

Some effort is currently spent within the ISS community to group activities into different categories. Activities marked as *flexible* in the timeline can be performed by the crew at their discretion—these tasks are not dependent on anything else. The daily exercise of an astronaut is a typical example.

Team Activities are tasks with constraints that are expected to be executed as scheduled. However, it may be performed differently than scheduled, but in this case a prior ground and crew coordination is required. Such activities might be dependent on special resources or conditions or are requiring support from the ground teams.

Time Critical activities need to be executed exactly at the time when they are scheduled. If this is not possible, it needs to be canceled for that day. In the timeline viewer those activities appear with a blue frame. Typical examples are live public affairs events.

In case some activities are related to each other and have to be conducted in a certain sequence, those activities are marked with the same color in the timeline viewer. Thus the astronaut still has some degree of freedom in the timeline execution, as long as he respects the given sequence of events.

Weekends and pre-agreed holidays are kept free of work if somehow possible. However, on crew request a concept of voluntary science/maintenance was established in the last years. In case the crew expresses their interest prior to their mission, the ground team put together tasks which can be autonomously executed by the crew on weekends or holidays. For each weekend, the crew then receives a pick list approximately 2 weeks in advance, from which they can select their preferred activities. These are then put into the timeline of the crew-off days; however, crew always reserves the right to resign from the tasks and protect the free days.

All activities, which appear scheduled in the timeline, are called "hard scheduled" and the time required for their execution is budgeted to the corresponding space agency. However, there is an alternate concept in use, which is called job jar or task list. The task list is populated each week with possible candidate activities which can be performed by the crew at any time. The idea behind the task list is to provide the crew a repository of activities which need to be executed sometimes, but are (at least for the moment) not anyhow critical. Typical examples are stowage audits. In case the crew has some "gray space" on a dedicated day they might consider to check the task list for further work and get ahead with the increment goals, if they wish.

5.3.5 Planning Phases Overview

Different planning phases need to be distinguished for the ISS project. The phases range from very generic planning of vehicle traffic and crew rotation far ahead to the very detailed daily planning of activities. As shown in Fig. 5.22, three phases can be distinguished. The strategic planning phase goes far into the future, and becomes tangible approximately 5 years before the execution phase. It directly transitions into the tactical planning phase, which is initiated around 1.5–2 years prior to start of the double increment.

The strategic and tactical planning phase follow an annual schedule rather than an increment-based one; therefore the temporal relationship to the start of the increment is by nature only an approximate value.

One year prior the increment the Execute planning processes are entered, which can be split into two subphases: the pre-increment and the increment execution planning. Since the planning products (see below) of the increment execution have a lead time of 17 days, this phase is started about 2 weeks prior to increment start. This phase runs throughout both increments. After the increments, corresponding reports need to be generated by the planning community, which are not further detailed here.

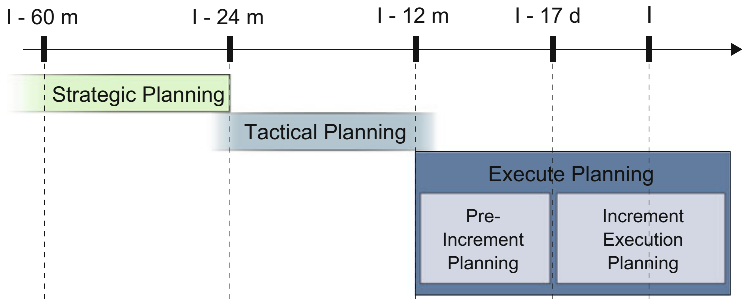

Fig. 5.22 The planning process for a dedicated increment can be split into different phases. These phases use the start of the increment (denominated as "I") as time reference. The strategic planning phase concretizes approximately 60 months prior to the increment start, the tactical planning about 24 months before. The Execute Planning phase is entered at I—12 months and can be split into the pre-increment time frame and the increment execution. The latter begins 17 days prior to the increment with the generation of the first execute planning product (see below) and lasts throughout the increment

5.3.6 *Planning Products and Processes*

5.3.6.1 Strategic Planning

The strategic planning phase is also referred to as multi-increment planning, since it covers a longer time frame; its planning horizon is 5 years in advance (Leuttgens and Volpp 1998). It is—like all planning phases—a multilateral process and defines the ISS assembly sequence, all visiting vehicle traffic, and the crew rotation on ISS and ensures that the corresponding resources and consumables are available which are required to reach the high-level operations and science objectives, which are also defined in that planning phase.

The most important documents which are generated during this planning phase are depicted in Fig. 5.23.

The Operations Summary is used for the generation of the Composite Operations Plan (COP), which contains projections of the utilization capabilities allocated to each international partner, and the development of the Composite Utilization Plan (CUP), which contains the utilization plans of all ISS partners. These two documents are combined and harmonized in the Consolidated Operations and Utilization Plan (COUP), which is also published once per year. The COUP covers five consecutive years of ISS operations, of which the first is the just completed one, for which a utilization report is provided, and the second is the one in the tactical/ execute planning phase. The subsequent years are in strategic planning: Years three and four were already contained in the last issue of the COUP, whereas year five is the newly added outlook five years ahead.

From the Consolidated Operations and Utilization Plan (COUP) and other source documents the Multi-Increment Planning Document (MIPD) and the Multi-increment Payload Resupply and Outfitting Model (MIPROM) are derived.

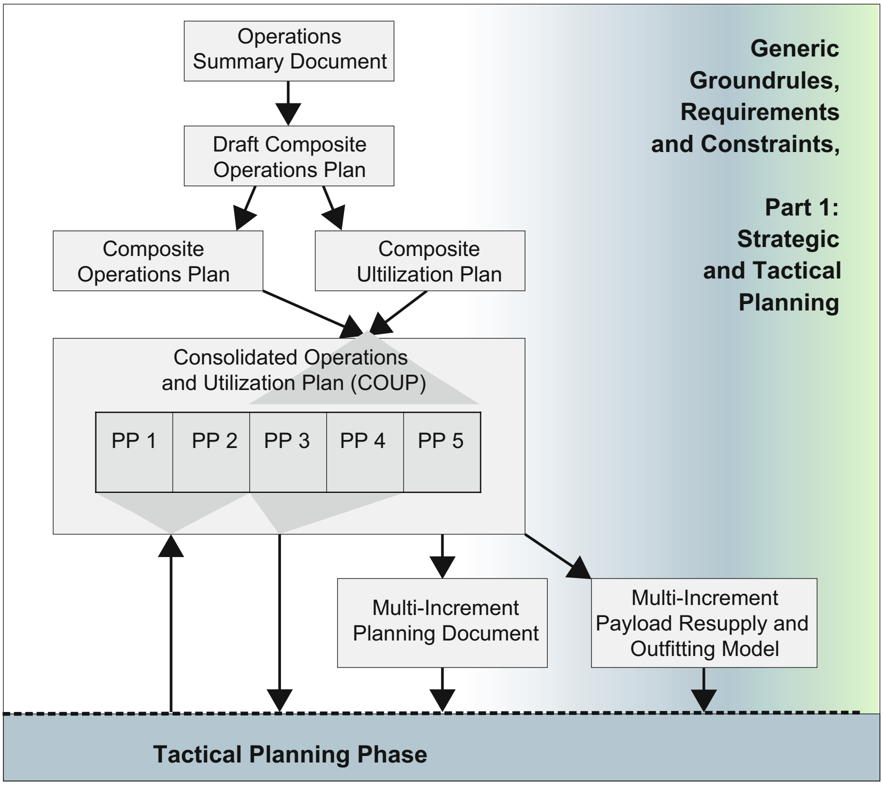

Fig. 5.23 The strategic planning process involves the generation of multiple documents which are iterated several times prior to final publication once per year. The Multi-Increment Planning Document, the Multi-increment Payload Resupply and Outfitting Model, and the COUP document feed the tactical planning phase. Results of the tactical planning are fed back into the COUP. The Generic Groundrules, Requirements, and Constraints, part 1, sets the rules and standards for the planning process (Simplified schematic)

The first one defines the ISS program tactical content and flight definition required to allow consistent planning and resource control. In addition, it identifies the projected ISS resources, accommodations, and supporting services available to the operations and utilization communities.

The latter provides long-range facility-class payload launch and disposal plans and the topology of the external payloads installed on ISS.

Both documents are also published on an annual basis and have the same outlook horizon like the corresponding COUP.

All documents described above are governed by a set of rules called the Generic Groundrules, Requirements, and Constraints (GGR&C), which accompany the entire strategic and tactical planning efforts.

Several groups are involved in the strategic planning process and the products are approved by high-level ISS program panels like the Program Integration Control Board (PICB), the Space Station Control Board (SSCB), and the Multilateral Payloads Control Board (MPCB). The strategic planning documents give directions to the subsequent planning process, the tactical planning.

5.3.6.2 Tactical Planning

Tactical planning is a multilateral ISS Program function that defines and documents the major program requirements, priorities, resource allocations, vehicle traffic, research objectives, the system-related assembly, logistics and maintenance work to be accomplished, and the payload manifest down to a sub-rack level for each increment 1.5–2 years ahead.

The tactical planning phase starts approximately 1.5–2 years before the start of the corresponding double increment. The high program level documents of the strategic phase need to be translated into requirements which can be implemented by the execute planning experts. Figure 5.24 summarizes the most important products of this planning phase.

The main document which is developed in that phase is the Increment Definition and Requirement Document (IDRD) including its annexes, which are in fact self-standing documents as well.

However, before the increment definition can start, the strategic considerations which are based on an annual timescale need to be broken down to double increments. All transportation systems capabilities and schedules, the ISS capabilities and available resources, and the operations and utilization requirements are now assigned to a dedicated increment and related flights.

The Increment Definition and Requirement Document (IDRD) Annex 5, also known as Payload Tactical Plan (PTP), marks the transition from the multi-increment-based planning to a double increment driven approach. It also details the payload planning down to a sub-rack level, and provides the corresponding up- and download manifest information including mass, volume, and additional resource requirements of each item during the transport phase. The research goals for the increment time frame are clearly defined and the required resources for each experiment are compared with the available amounts. This includes also the usage of common station items like tools, disposable gloves, wipes, and tapes. The topologies of the experiment hardware on board are laid out, which means that the configuration of the entire space station is described down to a locker or rack insert level and changes to that configuration during the increment are defined. A table lists all requirements for cold stowage of items. For each research crew activity it is already documented what time has to be allocated for it and the details of the training which has to be delivered to the astronauts on ground are documented. All special agreements between the international partners which are in place for the increment are also referenced.

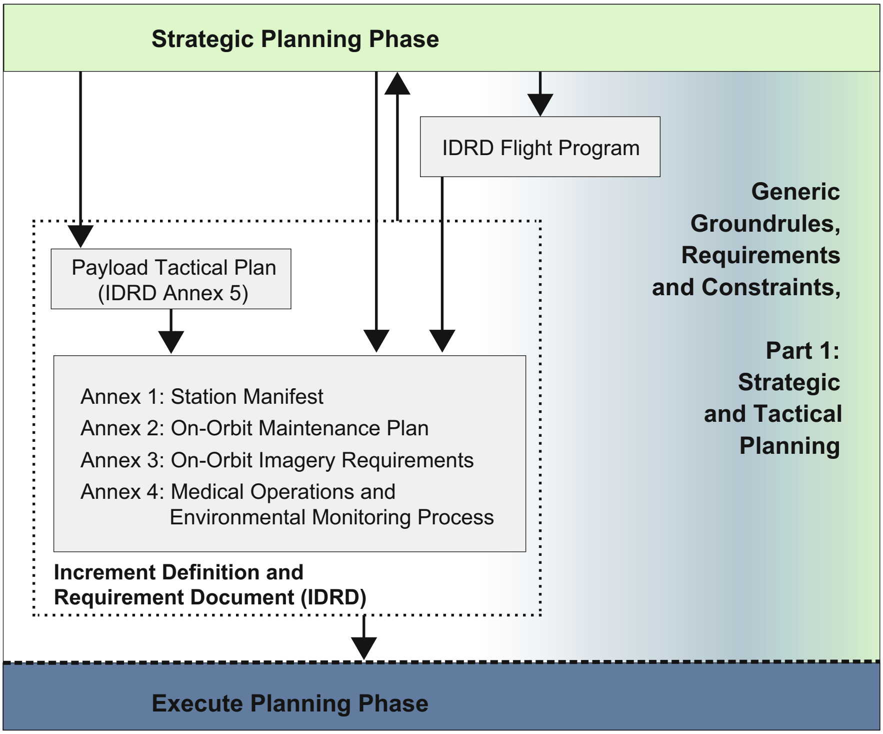

Fig. 5.24 During the tactical planning phase the outcomes of the strategic processes are translated into the Increment Definition and Requirements Document (IDRD) and its annexes, which then serves as input for the execute planning

The document generation process cumulates in its baselined version approximately one year prior to the start of the corresponding double increment.

The Increment Definition and Requirement Document (IDRD) Annex 5 is one major driver of the further planning, which is laid out in great level of detail in the other annexes of the IDRD. These annexes follow their individual development schedules and are under the control of different working groups and decision boards. Annex 1 (Station Manifest) is essentially a detailed inventory list of all planned vehicle “flights” to the space station and retour to earth. This includes not only payload items, which have been identified in Annex 5, but also system or assembly parts, resupply, propellant, cryogenics, water, or crew items. Annex 2 (On-orbit maintenance plan) of the IDRD lists all maintenance activities which are required in the double increment time frame to ensure that the station remains in a good shape. In general it is distinguished between corrective and preventive maintenance and the corresponding requirements are derived from the various design documents of all station hardware. Annex 3 (On-orbit imagery requirements) documents the requirements of on-board imagery, which might come from generic requirements, or payload or system driven requirements, or other sources. Both still and video imagery are listed. Finally, all medical requirements and the guidelines and rules for ISS environmental monitoring and control are summarized in Annex 4 (Medical operations and environmental monitoring).

The Increment Definition and Requirement Document (IDRD) serves as the main input for the next planning phases, which is commonly referred to as Execute Planning. Since the IDRD contains also a priority ranking for all activities, it also serves as important document in that regard during the increment. There might be the need for changes of the content which is driven by the latest real-time developments. Therefore the document is translated into an operational document, the so-called Current Stage Requirement Document (CSRD), which is then maintained by a common effort of the management and operations community and serves as a guideline document for task priorities.

5.3.6.3 Pre-Increment Planning

The transition into the Execute Planning phase also marks the point in time, when the planning is handed over to the actual planning teams, which then proceed with building more and more detailed executable planning products.

The two main planning products, which are developed during the Pre-Increment planning phase, are the On-Orbit Operations Summary (OOS) as a rough outline of the future Timeline (see Fig. 5.25) and the increment-specific Ground Rules and Constraints (Gr&C), which comprises a set of planning rules, which apply for the activities of the increment. Both products are generated by the Execute Planning Teams in a common attempt, which cumulates in two Technical Interface Meetings (TIM), the so-called OOS TIMs. Here, the planning for the corresponding double increments is consolidated among all international partners.

The On-Orbit Operations Summary (OOS) is the first product which has a clear planning character in its literal meaning: It assigns dedicated activities to a dedicated crew member and already puts them into a temporal context. The duration of each activity is already known and thus the planning to a granularity of one day can be accomplished.

The planning teams not only take the requirement documents of the tactical planning phase into account, but they also get the actual activity definitions from the corresponding activity owners (i.e., the payload expert centers), which already comprise of all information which is required for the planning process, as laid out in Fig. 5.21. This technical information contains the duration of the activity, the assigned and trained crew members, any requirement for resources (e.g., power) and conditions (e.g., Ku-band coverage for video or data transfer), the associated procedure, and correlations or interferences with other activities.

The international ISS partners work their contributions for the On-Orbit Operations Summary (OOS) and the Ground rules and Constraints (Gr&Cs) independently, but in close contact with each other. 19 weeks prior to the increment start (I—19w), the payload parts of OOS and Gr&Cs are collected by the Payload

Mo	GMT 98						
09-Mai-11	ISS CDR	FE-1	FE-2	FE-3	FE-4	FE-5	FE-6
EHS-ALM-START						00:10	
EHS-ALM-STOP						00:50	
EHS-TEPC-RELOCATE							00:20
PAO-HD-SETUP							00:10
PAO-PREP						00:10	00:10
PAO-HD-EVENT						00:20	00:20
HMS-AED-INSPECT						00:05	
CMS-T2-DIS-PWRDN						00:05	
WHC-UR/IF-CHGOUT						01:30	
WRS-WATER-BALNCE-PLC							00:30
WRS-WATER-BALNCE-PLC							00:30
ICV-USND2-PREP						00:25	
ICV-R/ECHO-SCAN_FE5						01:00	
ICV-ECHO-SCAN_OPR50							00:50
USND2-H/W-PWROFF						00:05	
USND2_DATA-XFER						00:10	
USND2-H/W-SU/PWRON						00:30	
JRNL-NOM-ENTRY_FE6							00:15
BASS-FAN-CALIBRATION							00:20
BASS-VIDEO TAPE-EXCH							00:10
BASS-EXP-OPS							01:00
MSG-FACILITY-ACT							00:10
MSG-FACILITY-PWR DN							00:10
VI-SCAN-OPS_FE5						00:45	
BLB-GAS SPLY-OPN						00:05	
BLB-GAS SPLY-CLS						00:05	
GHF-SCAM DOOR-OPEN							00:15
GHF-SCAM DOOR-CLOSE							00:15
GHF-HICARI-CTRG-INST							00:25
БВС-СУБА-R&R-PREP							
СОЖ-MNT							
Д33-12-EXE							
TEX-39-DNLD-D/L							
TEX-39-RSE-LCS-ON							
IMS-EDIT							
ТГК-47P-IMS-XFER							
Subtotals	06:40					05:30	05:50
Minus	00:00					00:00	00:00
TOTALS	06:40					05:30	05:50

TOTAL	TIME
MEDICAL	
OBT	
ROUTINE	
US MAINTENANCE	
RS MAINTENANCE	
ESA MAINTENANCE	
JAXA MAINTENANCE	
EVA	
US UTILIZATION	
RS UTILIZATION	
ESA UTILIZATION	
JAXA UTILIZATION	
XFERSTOW	
RS SYSTEMS	
US SYSTEMS	
ESA SYSTEMS	
JAXA SYSTEMS	
TOTALS	
WEEKLY CREW TOTAL	

Fig. 5.25 The On-Orbit Operations Summary contains already a full list of all payload and system activities, assigns them to a dedicated day of the increment and to a crew member

Operations and Integrations Center (POIC) in Huntsville/Alabama and are integrated by the NASA experts within 2 weeks. Next, the integrated payload OOS and Gr&Cs are delivered to Houston 17 weeks prior to the start (I—17w) and combined by their planners with the system inputs, which are at that point also provided by all international partners.

The integrated, preliminary On-Orbit Operations Summary (OOS) and Ground rules and Constraints (Gr&C) documents are thus ready 4 months prior to the double increment start and are now discussed in the already mentioned Technical Interface Meeting (TIM) between all planning teams to detect and resolve any conflicts.

A second iteration of the OOS and Gr&C integration then follows: All payload inputs are again collected by the Payload Operations and Integrations Center (POIC) at I - 11w and forwarded to Houston 2 months prior to increment start, where the final OOS/Gr&C document is compiled. A final planning Technical Interface Meeting (TIM) is held, which cumulates in the approval of the final planning products for the upcoming double increment.

The On-Orbit Operations Summary (OOS) plans for each day of the two increments do not yet contain the actual scheduling times of the activities nor are the standard elements of a crew day (e.g., exercises, meals) included. This level of detail is only reached in the next planning step.

The increment-specific Ground rules and Constraints (Gr&C) contain special planning rules, which have to be followed by the planning teams to ensure that all activities are planned in a proper way compliant with all scientific, technical, or medical constraints and can be executed on a non-interference base.

The Gr&Cs are used throughout the increment for the subsequent planning processes and planning product developments.

5.3.6.4 Increment Planning

During the increment planning phase, the On-Orbit Operations Summary (OOS) is successively translated in more detailed planning products, until the final On-Orbit Short-Term Plan (OSTP) is attained, which serves as the final product which is executed by the operations teams. The temporal relationship between the different products is shown in Fig. 5.26.

Each week of the increments, a seven days time period (Monday to Sunday) is extracted from the On-Orbit Operations Summary (OOS) and is transformed by the Long Range Planners (LRP) into the Weekly Look-ahead Plan (WLP). This timeline usually already contains for all activities the exact timing, and the daily crew tasks are also already included. A full working week is dedicated for the generation of each WLP.

For the Weekly Look-ahead Plan (WLP) development, which is again decentralized by all international partners and finally integrated in Houston, the Payload Operations and Integration Center (POIC) derives from the OOS a payload planning template, adds the communications resource availability, and ensures that the crew sleep cycle is adapted to the latest agreements. This template is used by all control centers to fine-tune their corresponding payload part of the week under consideration. Then the info is fed back to POIC who builds a preliminary payload WLP.

The preliminary payload Weekly Look-ahead Plan (WLP) is then further integrated with the system inputs from all partners by the planning team in Houston and finally results in an integrated preliminary Weekly Look-ahead Plan (WLP), which is available mid of the generation week. The remaining week is dedicated to discuss the preliminary product, to resolve any conflicts within the WLP, and to generate a

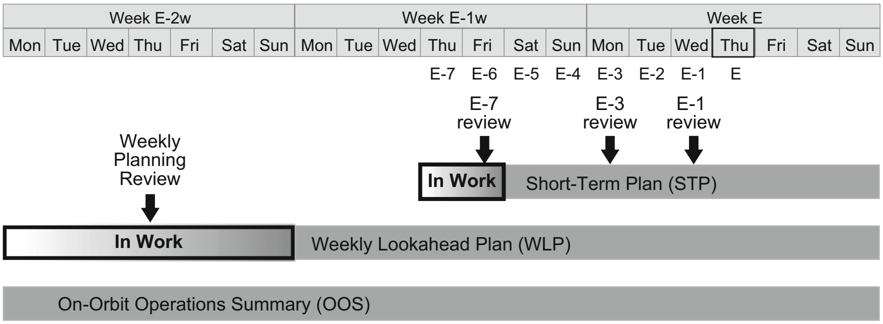

Fig. 5.26 The Execute planning phase is illustrated for a given increment day, the execution day E. A first version of the planned activities for that day is already contained in the OOS, which is prepared and approved prior to the increment start. Two weeks prior to day E the Weekly Look-ahead Plan (WLP) for the corresponding week is prepared using the OOS as baseline, which contains already a more detailed schedule for E. From the WLP the STP is derived seven days in advance (E-7) and finalized at E-6. This product is reviewed by the console teams three more times ("E-7", "E-3" and "E-1"), before it is executed finally

final WLP, which is available on Friday. The formal approval of the planning product is due on Monday of the following week, during which already the translation of the WLP into the next planning product, the Short-Term Plan (STP), starts.

The WLP development phase is each week supported by regular telecons (International Execute Planning Telecon, IEPT) between all international planning teams on Monday (approval of last week's final WLP, discussion of first issues), Wednesday (discussion of preliminary WLP), Friday (discussion of final WLP) and the Weekly Plan Review (WPR), which is a high-level meeting which involves management, various increment lead functions, and the planning community.

The approved Weekly Look-ahead Plan (WLP) is in the following used to generate the Short-Term Plan (STP). Each day of the week a new development cycle for the day one week in advance is initiated: One separate day is extracted from the WLP one week prior (E-7) to execution; each partner generates its input for the STP and sends it to Huntsville and Houston. At the Payload Operations and Integration Center (POIC) in Huntsville, the payload part of the STP is integrated and then sent to Houston, where the final integration takes place. The preliminary, integrated STP is then compiled into the On-board Short-Term Plan (OSTP) format, which can be considered the executable version of the STP, and published via dedicated software, the OSTP Viewer (OSTPV), for a review of the flight control teams on console.

This review, also called "E-7", is conducted by all three console shifts and all change requests for the STP/OSTP are collected in an electronic document, called Flight Note. The planning team on Houston side then implements all changes requested by the Flight Controllers, which leads at E-6 to a final, integrated STP for the execution day E. Also this final plan is converted into an OSTP and published in OSTPV. At that time, the STP is considered consolidated and

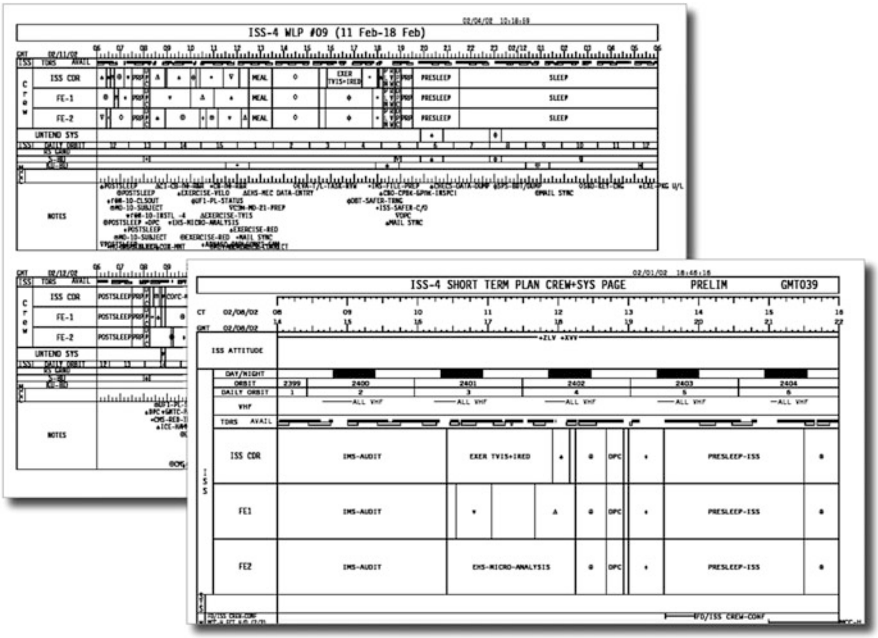

Fig. 5.27 The level of detail in the Weekly Look-ahead Plan (WLP) and the Short-Term Plan (STP) are practically almost similar

approved—the actual planning work is completed for day E and the final product, the OSTP, was handed over to the Flight Control Teams and the astronauts for execution.

The Short-Term Plan (STPs) are also subject of discussion in the above-mentioned International Execute Planning Telecons (IEPT), which are conducted every other working day (Mon, Wed, Fri). The STPs for the weekends are usually easy to produce, since Saturday and Sunday are usually "crew off days"; thus they are developed together with the STP of the corresponding Friday.

For the time of a visiting vehicle launch and subsequent days the planners may decide to develop in addition to the nominal STP a dedicated slip STP, which is then put into effect when the launch and thus the docking with the station is delayed. This is usually only done if the launch slip is likely and if a slipped launch has significant impacts on the crew timeline.

The format of the Weekly Look-ahead Plan (WLP) and the Short-Term Plan (STP) is quite similar, as Fig. 5.27 suggests.

5.3.6.5 Real-Time Planning

For the flight control teams the first time they get to see the so-called timeline on console is the publication of the preliminary Short-Term Plan (STP) seven days in advance. The publication is done via the OSTP Viewer (OSTPV) and appears as

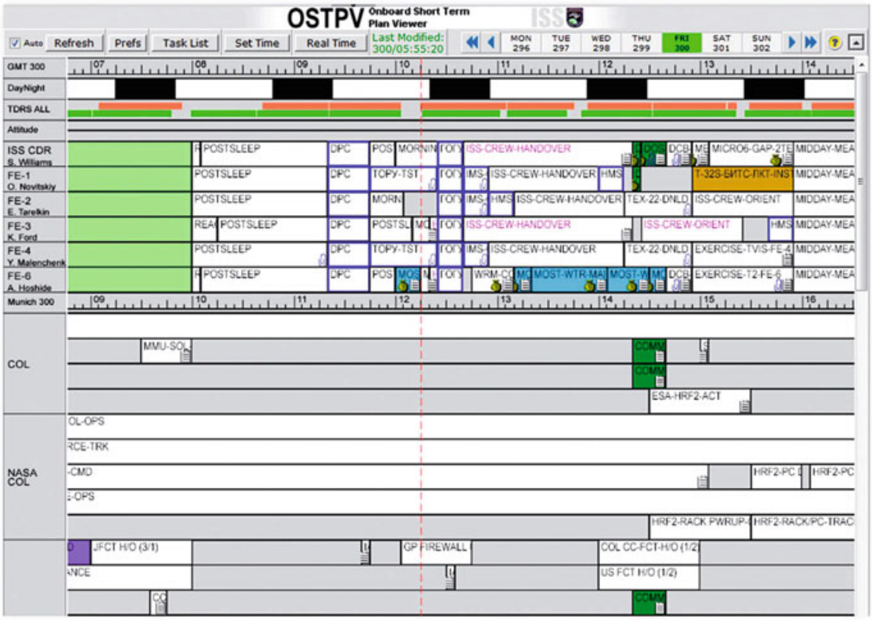

Fig. 5.28 The On-Board Short-Term Planning Viewer (OSTPV) displays the timeline on board as well as in the control rooms

shown in Fig. 5.28. As already described, the team performs a common "E-7" review of that STP and puts any change requests into a corresponding, dedicated Flight Note. The subsystem experts thus help to consolidate the final STP, which is then published in a corrected version on the subsequent day. The responsibility for the timeline is at that point also transferred from the long-range planners to the planning function within the flight control team.

As the execution of a given day comes closer, the flight control team performs two additional reviews of that timeline: The "E-3" review three days in advance and the "E-1" review the day before. The agreement for those timeline reviews foresees that only substantial errors with a potential major impact if not corrected can be subject of change. Should there be a need for a modification of the timeline a formal change request process is triggered, which requires the concurrence of all international partners.

Also outside of the formal reviews, changes to the timeline can be requested via this process, if e.g. the latest developments on board or any issues on ground would require an adaptation of the final, approved planning products.

Although the timeline is worked and optimized over weeks or even months and has passed various reviews, the astronauts still have a certain degree of freedom to deviate from the plans, as explained in Sect. 5.3.4.

The responsible control centers survey the activity execution of the crew and can, with respect to the timing, take appropriate measures in real time, in case the crew is not able to work the prepared timeline as planned. In this case the planning functions of the control centers work in the background on proposals how to reschedule the remaining crew day, to identify activities which can be postponed to another day or shifted to another crew member without impact, or—with the help of the corresponding Flight Director—to cancel planned activities with lower priorities.

5.3.7 Planning Tools

Being an international and distributed effort, the execute planning is highly dependent on computer and web-based tools which are used by the affected parties.

Some tools are used to collect the planning data, which is required to build the corresponding plans and schedules. Every ISS partner has currently its own processes and pieces of software, which reflect the requirements of their internal organization. Then the data is exported in a defined data format to a dedicated NASA tool called Consolidated Planning System (CPS), which is then used by the planners to generate the Weekly Look-ahead Plan (WLP) and the Short-Term Plan (STP).

When the On-board Short-Term Plan (OSTP) is ready to be published, the web-based OSTP Viewer software (OSTPV) is utilized which provides an easy-to-use and effective visualization of the schedule, and which is not only used by all ISS control centers, but also by the astronauts on board ISS (at least the non-Russian crew—Russian cosmonauts are required to execute the so-called Form 24, which is a tabular representation of the STP).

OSTPV allows not only a graphical representation of the schedule, but also access to the additional information, which is related with each activity, as shown in Fig. 5.20. The corresponding procedures are linked as well and can be opened by utilizing another piece of software, the International Procedure Viewer (IPV). The corresponding Stowage Notes, which list all tools and materials required for a crew procedure execution together with their stowage location, are also attached.

The process to change one of the approved planning products (WLP, STP) is also supported by a web-based tool, the Planning Product Change Request (PPCR) software. This interface grants all partners access to a “from to” form, which is used to provide all the info required to process the change request. The tool also provides a functionality which allows to seek formal concurrence of all international partners for the requested change.

The tools are currently under rework by NASA in an effort to generate a more integrated tool suite, include more automatization, and enhance the efficiency. Several years of ISS operations have resulted in many innovative ideas to improve the user interfaces, to better connect the various tools with each other, and to implement new functionalities.

5.3.8 Conclusion

Planning is a key element for mission success. Planning for human spaceflight missions is in many aspects comparable to the corresponding processes of unmanned missions, but also areas of significant differences exist. It is a manual and thus more process-oriented endeavor, whereas for unmanned missions also a high degree of automatization is possible, which requires a good mathematical modeling of the planning problem. The fact that no board computers but human beings are subject of the scheduling introduces an element, which requires also "men in the loop" for the various planning processes.

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Chapter 6
Spacecraft Subsystem Operations

6.1 Telemetry, Commanding and Ranging Subsystem

Michael Schmidhuber

6.1.1 Definition of Subsystem

This chapter deals with the operations of the spacecraft components that allow the radio frequency transmission of remote monitoring and control information of a spacecraft.

Although in many projects and system descriptions this function is handled as a subcomponent of the Data-Handling Subsystem, we decided to treat it here as a self-standing subsystem. Common names of it are TM/TC (Telemetry/Telecommand) or TCR (Telemetry, Commanding, and Ranging).

The subsystem as it is described below serves only the role of radio link and not of any processing of or insight into the transferred data which is done by the Data-Handling Subsystem (see Sect. 6.2).

This subsystem can also be used for orbit determination, a function that is commonly referred to as "ranging" or "tracking."

The future prospect of optical communication is not covered here.

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6.1.2 Signal Characteristics

A good and fundamental description is contained in Sect. 1.3; this chapter highlights only special aspects that are of interest for the TCR operations task.

6.1.2.1 Frequencies

Electromagnetic signals are typically divided by their frequency into ranges called "bands." Refer to Table 1.3 for an overview of ranges and frequencies.

The frequencies used for a given spacecraft are selected by its corresponding communications requirements. S-Band (~2 GHz) is currently used mostly for low-earth orbiting spacecraft that transmit actively only over assigned ground stations, or for all space missions during their launch and early orbit phase. This comparably low-frequency band is suited for a relatively easy design of round-transmission signal characteristics. Obviously this is a useful feature in case the angle between the spacecraft antenna pointing direction and the direction towards the corresponding ground station (antenna aspect angle, see Sect. 6.1.3.2) is continuously changing.

It is also beneficial for contingency cases with uncertain spacecraft attitude, which does prevent an accurate antenna pointing. S-Band antennas are currently widely available around the earth at many ground stations which makes cooperations during LEOPs and emergencies easier.

X-Band (~8 GHz), Ku-Band (~12 GHz), and Ka-Band (~20–30 Hz) are using higher frequencies which result in higher possible data rates. These bands are mainly used during the routine phase of the spacecraft lifetime. Their bundled signal characteristic is used to avoid interferences and to save energy, but require exact pointing.

There is currently a tendency to shift operations to higher frequencies. This is partly due to pressure from other spectrum users (mobile ground communications) who want to use these bands for other applications and the fear of spacecraft operators of signal interference with increasingly crowded frequency regions. But mainly it is caused by the increased demand in telemetry downlink bandwidth.

The transmission of signals in high-frequency ranges like Ka-Band is highly dependent to atmospheric conditions and is thus affected by moisture and rain.

All frequency uses have to be coordinated with and approved by the ITU (International Telecommunication Union).

6.1.2.2 Polarization

Like visible light and any other type of electromagnetic radiation, radio signals can be polarized, which means that the direction of the oscillation is not randomly distributed, but follows a defined behavior. Two main types of polarization are

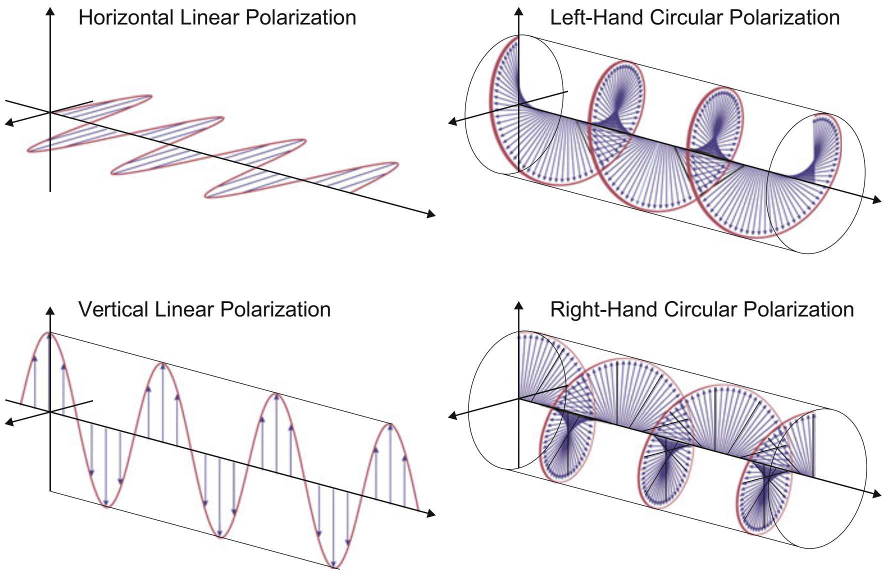

Fig. 6.1 The different types of polarization of electromagnetic waves. The oscillations of the electric field are used to describe the polarization type

distinguished: linear polarization and circular/elliptical polarization, as depicted in Fig. 6.1.

Both types of the phenomenon allow two independent ways of transmission on the same frequency: Two signal waves with perpendicular linear polarization can be considered as independent channels as well as circular/elliptical polarized waves with different senses of rotation. This can be used intentionally in communications, to make double use of a frequency. The receiving antenna has to be designed to filter out a selected polarization.

For stationary links the linear type is easiest to realize. For example, the polarization patterns of satellite television signals are either X- or Y- polarized. If receiver and transmitter antennas are likely to be rotated (or even rotating) around the signal direction, the circular polarization is used. Left-hand circular polarized (LHCP) and right-hand circular polarized (RHCP) can then be used to distinguish the two independent channels.

6.1.2.3 Side Bands and Side Lobes

Every antenna produces a signal not only in its main direction, but also generates side lobes, as depicted in Fig. 6.2.

Depending on the antenna design, these side lobes are weaker in strength and can normally not be used for signal transmissions. However, they may result in false

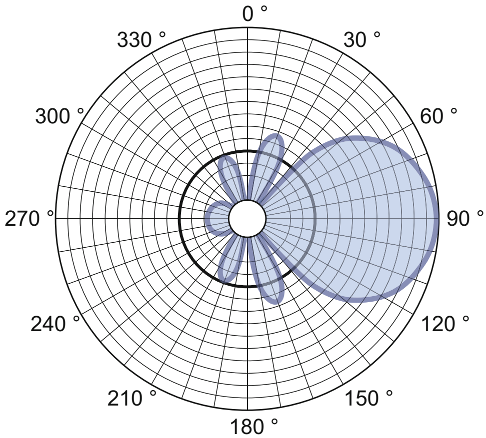

Fig. 6.2 The antenna pattern including the main and the side lobes

receiver lock conditions during the phase of antenna alignment, as discussed below. See Sect. 1.3.3.1 for further details.

Not only in the spatial domain also in the frequency domain inadvertent side effects can occur.

If a carrier frequency is modulated by a signal, side frequencies are generated. They contain the signal information. The characteristic of this frequency pattern is dependent on the transmitter design (filtering). In the initial phase of frequency alignment strong side frequencies can lead to an incorrect receiver lock (misusing the side band as carrier frequency). The resulting demodulated signal is normally not usable as it is considerably weaker in strength.

6.1.3 Design

6.1.3.1 Subsystem Elements

The design of this subsystem is largely driven by the mission requirements and system details will rarely be identical on two spacecraft. But the main building blocks are mostly recurring. They are:

- Antenna
- Receiver
- Transmitter
- Routing and Switching unit

In the block diagram in Fig. 6.3, an uplink signal from the ground station is picked up by an antenna and guided to the receiver. There it is demodulated from the carrier signal. The resulting output is still an analog waveform. It is routed to the TM/TC Board of the on-board data-handling subsystem. This board is considered

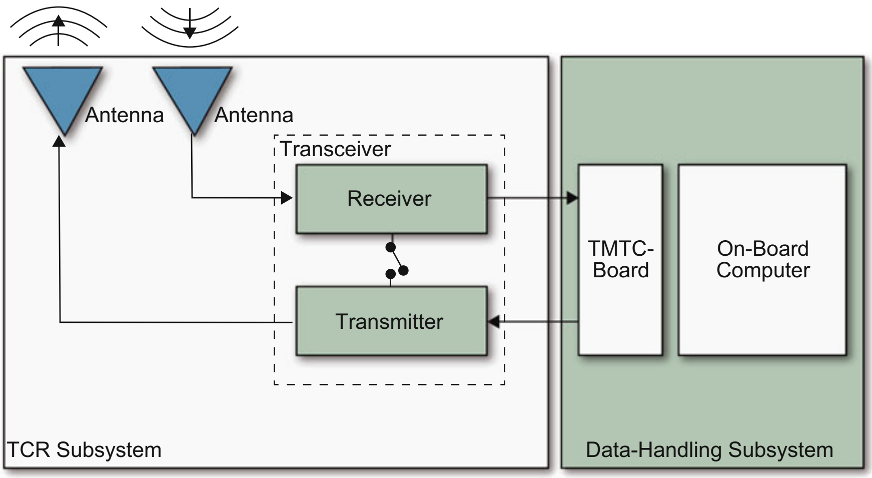

Fig. 6.3 Block layout of the TCR subsystem. It shows the fundamental components and the separation against the Data-Handling Subsystem

part of the data management system and is described in Sect. 6.2.2.1. Reversely the transmitter gets its input for the downlink from the on-board computer (OBC). It modulates it onto a carrier frequency and routes it to the transmitting antenna.

Some systems are designed to allow a direct signal connection between receiver and transmitter for the ranging function. This technique is rarely used in low-earth orbit (LEO) where angle tracking and GPS tracking are dominant, but is standard in orbit altitudes where GPS cannot be used (GEO/interplanetary). Ranging is explained in more detail in Sect. 6.1.5.2.

Antennas can be used for reception and transmission at the same time if interference of signals is avoided. This is done by using an uplink frequency that is different from the downlink frequency and by using filters in the signal paths.

The functional unit of receiver/transmitter is called a transceiver (transmitter–receiver).

For Payload data often a dedicated communications system, operating in a different frequency band is used (see Table 6.1 and Fig. 6.5). In case of scientific satellites this follows the same principles as described here. For communications satellites refer to Sect. 6.6.

Figure 6.4 shows an example of a geostationary communications satellite TCR subsystem. The same data can be received and transmitted through any of the four shown transceivers. The S-Band systems are used as a redundant pair during LEOP and during emergencies. The omnidirectional antenna patterns make transmission independent of the spacecraft orientation. The S-Band receivers cannot be switched off to ensure they are available during emergency situations. The Ku-Band systems are used during the routine phase when the spacecraft is fixedly oriented to the earth. The antenna has a bundled characteristic that needs less energy to operate and

Table 6.1 Examples of frequency usages for different types of satellite missions

	Mission type	Orbit	S-Band	X-Band	Ku-Band
GRACE	Scientific earth observation	LEO	TM/TC and Payload	–	–
Terra-SAR-X	Scientific/commercial earth-observation	LEO sun-synchronous	TM/TC	Payload	–
Eutelsat W24	Geostationary communications	GEO	TM/TC during launch and early orbit phase and emergencies	–	TM/TC during routine phase Payload Channels

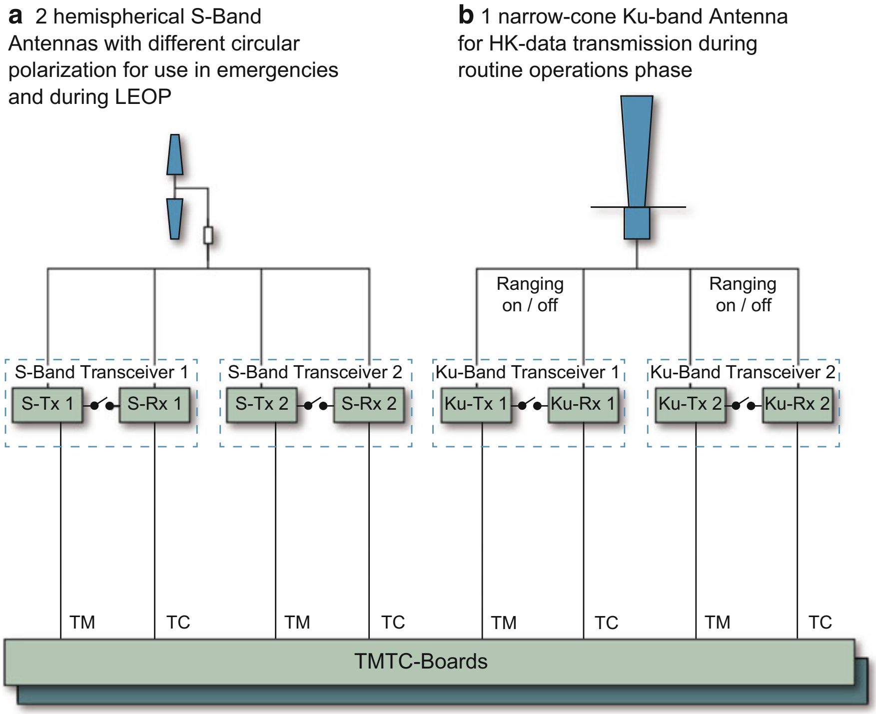

Fig. 6.4 Example TM/TC System layout of a geostationary communications satellite

does not as easily interfere with other RF signals. The Ku-band transceivers offer more functions like selection of different frequencies and some cross-coupling that is not shown in the figure. All four transceivers allow ranging, as this is the most precise orbit determination method available for geostationary satellites.

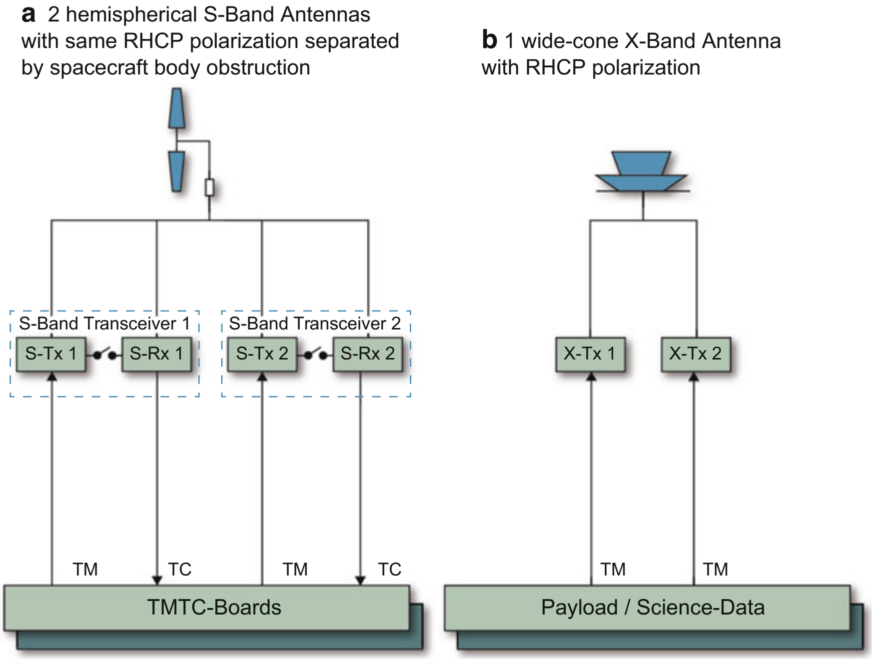

Fig. 6.5 Example layout of a scientific satellite in low-earth orbit

Communication satellites usually offer a beacon functionality to assist ground receivers in locating the satellite. This may be either a separated transmitter or one of the telemetry transmitters is used. This signal should to be permanently available.

In comparison Fig. 6.5 shows the TCR system of a scientific LEO satellite, based on the TerraSAR-X design. A ranging function is not used as the spacecraft can use GPS data for orbit determination. Only an S-Band system is used for real-time off-line data. The transmitter output power can be adapted to allow transmission with high data rates requiring more power and more exact pointing of the antenna to the ground station. The payload data system is separated and uses X-Band as an even higher data rate is needed for the payload data. A wide-angle pattern is needed on the payload antenna because the spacecraft is oriented along its orbital track and its attitude relative to the ground stations is therefore changing during passes. The payload antenna is nadir pointing.

6.1.3.2 Spacecraft Antenna Layout

Antennas can be distinguished by their transmission and reception characteristics, the so-called antenna pattern. It describes the sensitivity of the antenna dependent on the direction the radiation is coming from. It can either be directional or

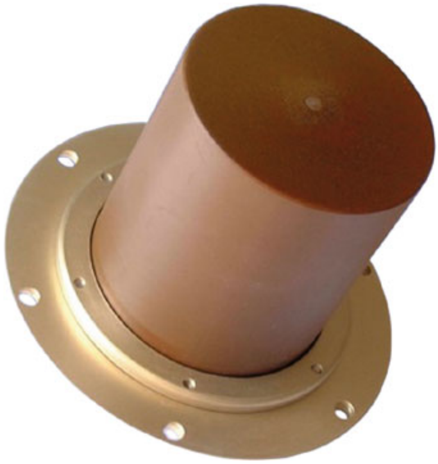

Fig. 6.6 A typical S-Band antenna for Low-Earth Orbit application with hemispherical antenna pattern and either right-hand or left-hand circular polarization. The shown model has a height of 100 mm. Image used with friendly permission of STT-SystemTechnik GmbH

nondirectional (omnidirectional). The higher the frequency the more directional is the characteristic. Normally the pattern is the same for reception and transmission. Typically directional antennas provide a higher antenna gain and are used for routine tasks with high data volume. Nondirectional antennas are used for tasks that need to be robust against spacecraft attitude changes like in emergency situations or when no high rate data transmission is needed.

A real spherical, omnidirectional characteristic cannot be achieved with one antenna as the spacecraft body and the antenna structure will always block a significant part of the wave propagation. Therefore two antennas on opposite sides of the spacecraft body are used to form a spherical coverage. This is in particular the case on low-earth orbiting missions. Here, two S-Band antennas with hemispherical antenna patterns are used (Fig. 6.6).

When two antennas radiate the same signal on the same frequency and polarization in the same direction the signal quality will be significantly degraded due to interference effects. One way to avoid this is to design the patterns of the two antennas with a gap between them, causing a belt with no transmission as shown in Fig. 6.7 (left). This case is easy to implement and does not put additional requirements on the ground stations. However, if the satellite orientation causes the belt pointing in the direction of the ground station, the contact to the spacecraft will likely be lost or severely limited. Attitude and antenna aspect angle have then to be carefully monitored and considered for operational impact.

An alternative design is shown in Fig. 6.7 (right). The two antennas have wider angle of reception causing a significant overlap and thus transmission in all directions, but to avoid interference they emit and receive radiation with different polarization. This design produces no gaps, but needs a more sophisticated ground segment, being able to receive and transmit with two polarizations.

The two polarizations chosen for this design are typically LHCP and RHCP, as those are not dependent on the axial orientation of the antennas with respect to each other.

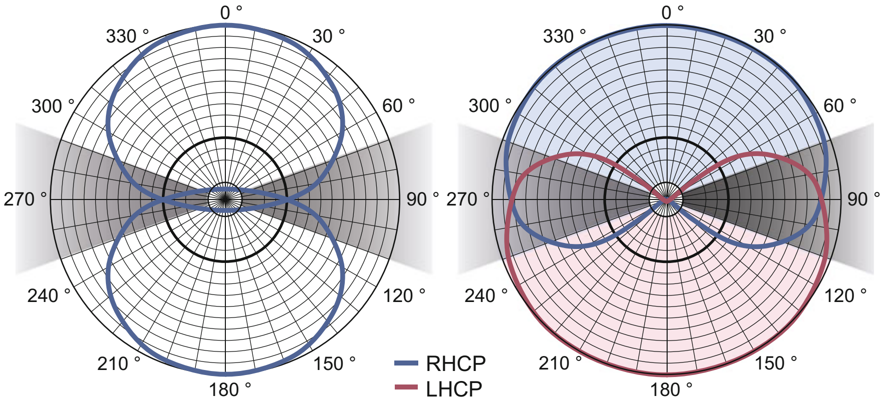

Fig. 6.7 (**a**) The two signals coming from the two antennas have the same polarization. In the overlap region (*darkly shaded belt*) the signals are too weak to result in a receiver lock. (**b**) The two signals coming from the two antennas have different polarizations. In the overlap region either signal can be used

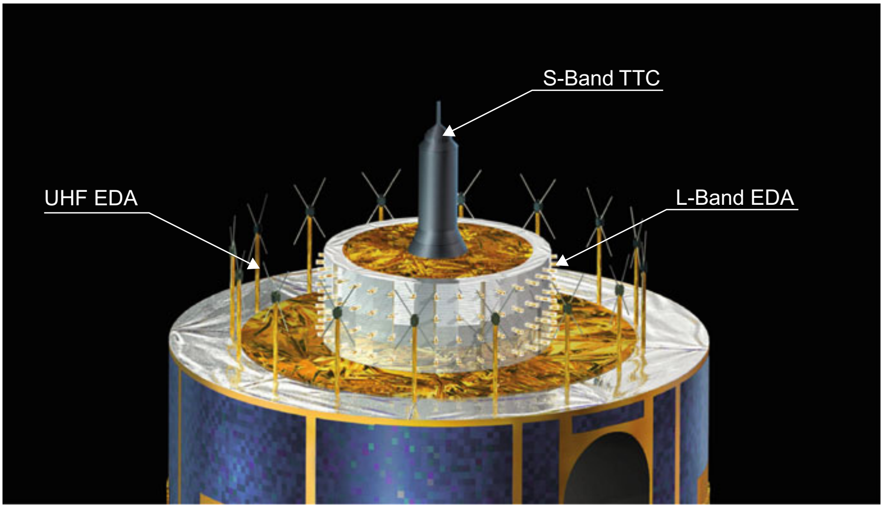

Fig. 6.8 The antenna section of a geostationary meteorological satellite. The spacecraft is spin stabilized and rotates around its symmetry axis (vertical direction in image). Photo: ESA

In some cases a complex overall antenna pattern is required. This can lead to designs with antenna arrays of more than 20 elements. Figure 6.8 shows the example of the Meteosat Second Generation spacecraft. The S-Band antenna is located on top at the spin axis and is of the type shown in Fig. 6.6. The L-Band antenna (~1.5 GHz) and the UHF (~400 MHz) antenna are composed of multiple elements pointing in radial direction. To avoid interference and unnecessary power consumption only the element pointing towards earth is activated [electronically despun antenna (EDA)].

6.1.3.3 Redundancies

To guarantee the function of the subsystem even after failures of components, more than one unit is supplied. Real redundancy is only reached when the backup component can take over the complete functionality. Complex electrical components like receivers and transmitters are therefore provided redundantly and independently, typically by providing a second system in parallel. Mechanical components like the antennas are very robust and are in many cases only singly available. A real redundancy is therefore not given. In these cases a limited redundancy can be reached by using other antenna systems like in the example of a GEO mission where a cabling problem during LEOP caused the complete loss of the Ku-Band system. The satellite was then operated in S-Band for the complete mission duration, without any loss of functionality, but with the negative effect of permanent omnidirectional S-Band radiation that caused interferences with other spacecraft.

Another example was the Galileo to Jupiter mission. Interplanetary missions typically have an omnidirectional low-gain antenna in addition to a high-gain main antenna that is folded for launch. In the case of Galileo the main antenna reflector failed to deploy. The complete mission was then handled with the low-gain antenna, causing higher efforts in on-board preselection and preprocessing and accepting a reduced data return.

6.1.4 Monitoring and Commanding

6.1.4.1 Automatic Gain Control

A central role in operations is the monitoring of the uplink Automatic Gain Control (AGC) level. The AGC is a circuit in the receiver that controls the signal amplification (or attenuation) to keep the signal strength in a defined range for subsequent components (Bullock 1995). The characteristic AGC parameter is the ratio of the signal level to an internal reference level. The term is used synonymously to the received uplink signal strength. The units are decibel; scale is logarithmic and negative. Typical values are between -50 dB (strong signal–low amplification) and -110 dB (low signal–high amplification) depending very much on the receiver type and the antenna characteristics. Values that are either far too high or too low may result in difficulties to demodulate the signal. If no signal (i.e., only noise) is received, very low values, e.g., -150 dB, will be indicated. Any remaining indicated signal level is caused by internal and external radio noise.

Operations shall be done only when the signal is in the range given by the manufacturer. It shall be planned ahead to either pause operations or switch over to another ground station with a better transmission situation.

The monitoring of the AGC level is important for successful operations. However the received signal level cannot easily be influenced. Therefore it is important to monitor the AGC evolution and make predictions about imminent development. For example, it is not advisable to perform commanding for an orbit maneuver over a station whose signal will become weaker or will be out of sight. An early switch to a different ground station should be considered. In some cases like an unusual spacecraft attitude, a boost of uplink power can be advisable.

6.1.4.2 Loop Stress

The signal is demodulated by a Phase Lock Loop (PLL) circuit in the receiver (Bullock 1995). This circuit generates a reference oscillation at the nominal frequency and compares it to the actually received signal. If the received signal is of exactly the same frequency, the loop is "in sync" or "locked." If a difference is detected, then the loop circuit is able to adapt the reference frequency automatically within a certain range to eliminate the difference and establish the lock status. This offset to the nominal frequency is called loop stress. It is provided by the PLL circuit as a voltage and can be converted into kHz. It has limits beyond which the PLL cannot compensate the difference. In this case the lock status is lost, and the signal cannot be decoded. Note that the PLL circuit cannot follow very fast changes in frequency as it has a certain inertness to adapt the frequency. Nor can it detect and adapt to frequencies too far away from the reference frequency even if the signal frequency is within the allowed loop stress range. To still achieve a lock the ground station has then to adapt the frequencies of its transmitter.

The difference in frequency causing the loop stress can come from imperfect adjustment in manufacturing, shifts in oscillator properties both on the sender and the receiver side (temperature variations, aging). The largest part, however, is caused by the Doppler Effect due to radial velocities between sender and receiver. This part has a typical values between ± 50 kHz for a low-earth orbiting spacecraft transmitting in S-Band when it appears at the ground station horizon. The variation of loop stress during a ground station pass is depicted in Fig. 6.9. A possible compensation for this effect is task of the ground station operations. Either a constant offset or a variable adaptation of frequencies can be applied to keep the loop stress within the limits and enable a successful receiver lock.

For interplanetary missions this effect can be much higher and the receivers have to be designed for this task. The interplanetary probe Voyager 1 is heading away from the sun at currently around 17 km per second. This causes an S-Band Doppler shift of ca -100 kHz. Finally a ± 185 kHz due to the earth orbital velocity over the course of a year has to be considered. However, this Doppler shift remains largely constant over short periods. Only a small shift of ± 3 kHz shift variation caused by the earth rotation is observed during ground station passes of a few hours.

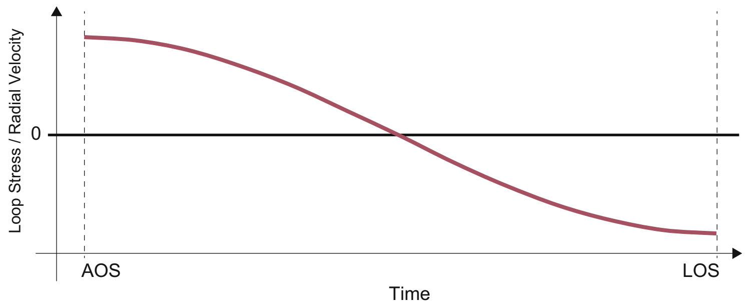

Fig. 6.9 The expected frequency shift caused by the Doppler effect during the ground station pass of a low-earth orbiting satellite. Acquisition of signal (AOS) and loss of signal (LOS) are the times when the satellite appears and disappears at the ground station horizon. The frequency shift is proportional to the radial velocity between satellite and ground station

6.1.4.3 Lock Status

As described above a lock status is necessary for the correct processing of the signal. Most receivers will indicate the lock status in telemetry. In other cases the status can be deduced from the AGC level and loop stress readings. It is worth noting that according to the ECSS-104 standard (ECSS 50-04C) a summary indication of the reception and lock status of all receivers is shown in the so-called Command Link Control Word (CLCW) telemetry. This flag indicates if at least one receiver is in lock. Since the CLCW has a fixed position in the telemetry transfer frames, it can be extracted from the telemetry stream by ground station equipment without complicated telemetry processing. Station personnel can use this to check successful uplink acquisition after an uplink sweep.

6.1.4.4 Polarization Prediction

Spacecraft that are not oriented to the earth (but e.g. to the sun) will have large changes in the antenna aspect angle (see Figs. 6.10 and 6.11) during a ground station pass. The antenna aspect angle is the angle between the ground station and the antenna boresight direction as seen from the spacecraft. This can result in bad reception conditions for the antenna/signal/polarization in use and a different polarization may be better suited for reception. However, although most ground stations can receive the downlink in more than one polarization (diversity), all stations uplink only in one polarization. A change of this polarization can take several minutes during which the contact to the spacecraft is lost and a new uplink sweep has to be performed. This situation can be anticipated if the TCR engineer makes a prediction about the expected polarization during an uplink period. The necessary calculation can be made easier if the antenna aspect angle is routinely predicted by the flight dynamics system for each ground station (see Sect. 6.1.5.1).

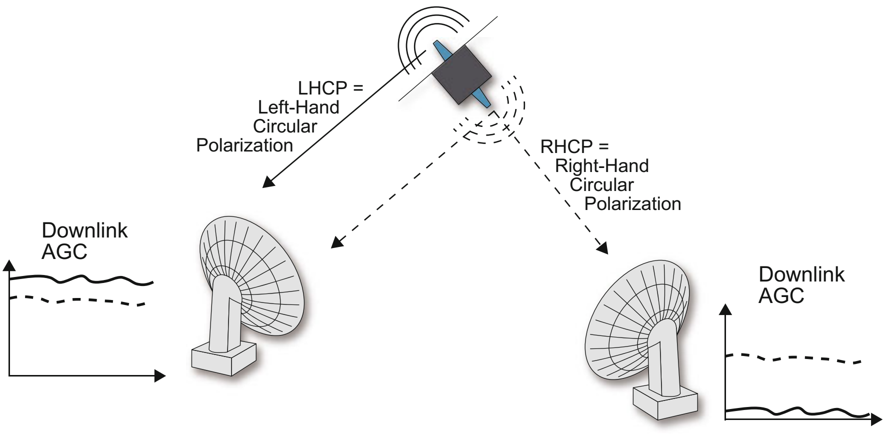

Fig. 6.10 The two ground stations can determine the reception condition at the satellite by looking at both received signals and use the stronger signal's polarization also for uplink

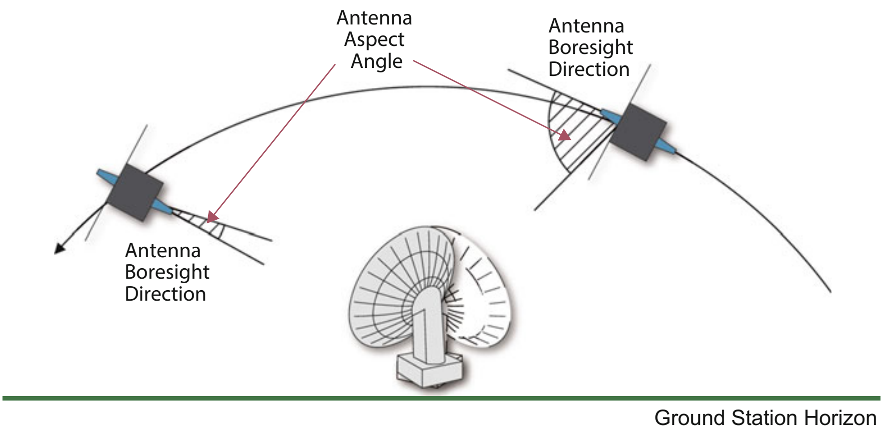

Fig. 6.11 The antenna aspect angle is the angle between the main receiving axis of the spacecraft antenna (boresight direction) and the direction to the ground station

6.1.5 Operational Situations

6.1.5.1 Acquisition and Loss of Signal

An acquisition of signal (AOS) has to be performed each time a change in uplink is done, e.g., caused by a station handover. It is a task done by ground station personnel and under normal conditions no action is required from the TCR engineer. To ensure a lock of the spacecraft receiver, the uplink signal frequency is shifted up and down in small steps in the vicinity of the expected frequency. At one of these steps the receiver will be able to lock onto the signal and can follow further variations. This is called a "sweep." However, as described above the station personnel check only if any receiver goes into lock and do not analyze the signal quality. It may happen that the satellite receiver goes into lock on a side band either in direction (side lobe) or in frequency (side band). This may produce a lock status, although the signal is not usable. It will result in a low AGC level (low signal–high amplification) and in case of a frequency side band a high loop stress. Uplinked telecommands can fail. A low AGC will also be seen in the ground received AGC and can therefore be detected by observant station personnel. If a situation like this goes undetected by the ground station, it is the task of the TCR engineer to inform the ground station and suggest a re-sweep.

The signal reception quality and strength are also affected by the operational situation. A major factor is the antenna aspect angle (Fig. 6.11). This angle changes for all satellites that are not earth-oriented and do not have steerable antennas. Depending on the antenna pattern the received and transmitted signal strengths are weakening when the angle becomes larger.

All transmissions are affected by atmospheric attenuation and refraction. This effect becomes very large and unpredictable when the signal has to travel large distances through the atmosphere near the ground station horizon. This is the reason why the time of loss of signal (LOS) cannot be known exactly. Operations planning shall take this into account and place critical operations into periods of stable reception.

6.1.5.2 Ranging

Ranging is a function that allows the determination of the spacecraft orbit. Unlike vehicles in lower earth orbits which nowadays simply use GPS to track their positions, satellites in higher earth orbits or on interplanetary cruise trajectories cannot locate their own position. Usually this is done using the ranging technique. Here the uplink signal is routed from the receiver directly to the transmitter and is radiated back to the ground stations. The on-board delay between reception and transmission time is exactly known and was measured on ground before launch. The ground station measures the signal round-trip time from which the distance between spacecraft and station can be calculated. Taking and processing the measurements are tasks of the ground station and the flight dynamics team. The impact on the flight operations team is that during the ranging period, roughly 5 min every 30 min for a GEO mission, no commanding shall take place in order to not distort the ranging signal. And also, depending on the transceiver model, the ranging function has to be switched on and off. Some receivers require after each interruption of the uplink to reset the uplink coherency function.

For interplanetary missions the impact is larger in that the ranging tones take away power from the telemetry signal which might be critical due to the weak signals coming from large distances. Also the operations pauses needed for ranging are considerably longer than for earth orbits (Bryant and Berner 2002).

6.1.5.3 Doppler and Coherency

Along with ranging, usually Doppler measurements are taken to improve the accuracy. The radial velocity component between satellite and ground station can be measured by determination of the frequency shift due to the Doppler Effect. For this method the downlink signal frequency has to be known precisely. As spacecraft usually have no highly stable oscillator on board, the self-generated downlink signal cannot directly be used for this purpose. In that case the uplink signal is used as frequency reference. Within the receiver, the downlink frequency is generated in a defined ratio (221/240 for S-Band) from the received uplink signal and thus becomes as stable as the ground station signal. This is called coherency.

Depending on the transponder model the coherency status may have to be commanded manually each time the uplink signal starts (or each time after the uplink was lost). In times without uplink the transmitter generates the frequency based on its own (imprecise) oscillator. A frequency jump will likely happen when coherency is started or stopped. This jump can be too fast or too far for the PLL circuit in the ground station receiver and can result in a loss of lock at the ground station. In that case telemetry will be lost and has to be reacquired.

Doppler shift, Coherency, and Ranging are normally used in parallel and the measurements influence each other. Ranging without coherency is possible and has been used in interplanetary missions (Reynolds et al. 2002). Also some geostationary satellites use this.

Interplanetary missions may actually use a highly stable oscillator at least during some mission phases like asteroid flybys in order to use Doppler measurements without the need for coherency.

6.1.5.4 Antenna/Transponder Selection

It depends very much on the design of the spacecraft if it is necessary to select the antennas used for transmission and reception. The less switching is involved the more robust the design and the simpler the operations. Mechanical devices like waveguide switches should be switched only if necessary and only without radiation load.

Typically receivers that are also used for emergencies are never switched off and are always able to receive over some antenna. Transmitters are usually only switched on if they are in use, to save energy and to avoid unnecessary frequency blocking. Only in contingencies these settings are commanded manually. Typically the commands are generated as part of the operational timeline and are automatically included.

6.1.6 Outlook to Future Developments

The future of the TCR subsystem and operations is predicted to lie in higher frequency domains and optical transmissions.

The proposed use of relay satellites may result in different antenna designs with steerable antennas that allow tracking of the relay spacecraft. The TCR subsystem may be supplemented by an optical communications system that also allows receiving telecommands and transmitting telemetry.

6.2 On-Board Data-Handling Subsystem Operations

Michael Schmidhuber

6.2.1 Definition of Subsystem

This chapter describes the operations of the spacecraft components that handle the on-board distribution and processing of data.

This subsystem is here called on-board data handling (OBDH), although also other abbreviations are used. The terms OBC or satellite board computer (SBC) are sometimes used synonymously to OBDH, but strictly spoken they describe only the core computing component.

Often the TCR signal transfer components are encompassed in the definition of the OBDH subsystem; however, in this book they are described separately in Sect. 6.1.

In our definition the subsystem includes the TMTC (Telemetry and Telecommand) Board, the OBC units, and the on-board data distribution components. Although the design of the subsystem is extremely diverse between different spacecraft, it is also a showcase on how the use of international standards can commonalize the utilization of spacecraft.

6.2.2 Fundamentals

In the following we introduce the basic components of the OBDH in more details.

The examples given here are taken from a classical communication satellite series (Spacebus 3000 by Thales) and modern LEO scientific spacecraft

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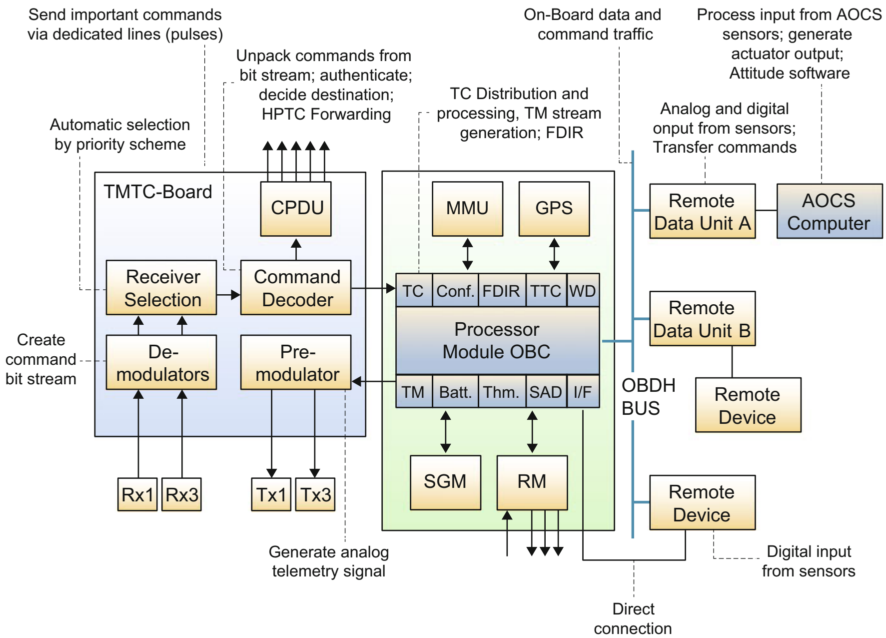

Fig. 6.12 Fictitious block layout of the components found on typical satellites. They are shown without redundancies

(TerraSAR-X by Astrium and TET by Kayser-Threde, AFW, and DLR). They allow demonstrating the basic principles that can be found in some way on all unmanned spacecraft.

6.2.2.1 Subsystem Elements

Low-earth orbiting spacecraft nowadays add a GPS receiver for time synchronization and orbit measurements and a mass memory module to store large amounts of payload data. Also in many cases the AOCS (Attitude and Orbit Control System) software can be included in the main processor.

Figure 6.12 shows an example of an OBDH system. It contains a fictitious collection of components taken from a classical communication satellite series (Spacebus 3000 by Thales) and a modern LEO scientific spacecraft (TerraSAR-X by Astrium and TET by Kayser-Threde, AFW, and DLR). They allow demonstrating the basic principles that can be found in some way on many unmanned spacecraft. Actual implementations will use subsets and variations according to the mission requirements and dependent on the manufacturer's design decisions. The components are explained in the following.

The TMTC board has the task to decode telecommand signals into logical structures, to transport them to their destinations and in the opposite direction to generate the telemetry signal out of the bits and bytes. It allows the isolation of dedicated command streams like high-priority commands (see next paragraph) and it contains authentication functions (see Sect. 6.2.3.4). Information about the result of the telecommand execution is merged into the telemetry stream along with the standard telemetry coming from the OBCs. The board is usually implemented as firmware running on robust and fast hardware. The generic design is standardized and largely independent of the spacecraft. The main standards for telecommand and telemetry are coming from organizations like CCSDS (Consultative Committee for Space Data Systems), ISO (International Organization for Standardization), and ECSS (European Cooperation for Space Standardization) formats.

Not all telecommands are processed by the OBC. Low-level commands are already handled by the TMTC Board. They serve very basic functionalities like switching of OBC to backup that have to work even in emergency situations. The TMTC Board includes a logic that allows a limited number of these commands. This is called Command Pulse Distribution Unit (CPDU). The pulses are forwarded to their destinations via dedicated cables. Also devices that need a higher electrical current than can be provided by normal data lines use this mechanism. Therefore they are also called “high-power telecommands.” A common example is the ignition of pyrotechnical devices that may release a folded antenna structure. The according command pulse needs a specified duration. This can is usually a settable command parameter. In case an HPTC command is not successful it may be helpful to increase the pulse duration. It is not possible to time-tag HPTCs.

On-Board Computer

Sometimes also called the Processor Module (PM), the OBC is a programmable computer that contains the satellite-specific software to manage complex spacecraft functions. Some functions may be located in dedicated, separated units: In Fig. 6.12 the attitude control functions are implemented in a separate computer module (AOCS Computer).

All software can potentially contain programming errors or may require updates, e.g., to adapt to deteriorating hardware. Therefore mechanisms are implemented that monitor the activity of the OBCs (see Sect. 6.2.4.2) and that allow uploads of new software (see Sect. 6.2.4.3).

Additional modules like a safeguard memory (SGM), mass memory units, and Reconfiguration Modules (RM) are located in close vicinity around the processor modules. They are described in Sect. 6.2.4.2.

Mass Memory Unit

Only spacecraft with permanent, reliable ground contact can be designed without extensive data storage capability. Interplanetary probes and low-earth satellites need to store data until it can be transmitted to ground. Nowadays solid-state memory units have replaced devices like tape recorders that have been used in earlier times. These memory units can be attached to or integrated in the OBC or they can be located next to payload instruments. Figure 6.12 does not show a mass memory device as a geostationary satellite is displayed. More detail can be found in Sect. 6.2.4.5.

Data Bus and Data Cable Harness

Most spacecraft use a data bus system for communication between the OBC and remote devices. Standards like MIL-1553 are used as protocol. Selected data may be transported over dedicated cabling to the OBC if it is of special importance with regard to safety or robustness or if the remote unit does not support the bus protocol. Remote Data Units (RDU) are used to connect sensors and actuators to the bus.

For telecommands it shall be noted that some commands are converted into electrical pulses that are transferred via dedicated cabling. These are the high-priority commands that are generated by the TMTC board as described above.

Additional Components

Low-earth orbiting spacecraft include a GPS receiver for time synchronization and orbit measurements. Note that GPS reception is conventionally only possible in orbit altitudes of up to 7,000–8,000 km above the earth. Nevertheless modern geostationary satellites also include GPS receivers for use in the transfer orbit, and with limitations also in the geostationary orbit.

In classical geostationary communication satellites usually the AOCS software has its own processor located in a remote device.

Other remote devices can be used for functions like power distribution and data bus interfacing.

6.2.2.2 Redundancy

Due to its key function, usually the OBDH system has a redundant layout. This means that all components are doubly provided to protect against failures. Depending on the design of the spacecraft, redundant devices may be switched off or may be set to a standby mode. In any case at least the redundant TMTC Board is powered all the time as it must be immediately able to take over the telecommand reception from the active board.

The redundancy concept requires careful balancing of costs and risks. It is quite common that not each and every function is fully and independently redundant, the latter one meaning that e.g. a given redundant fuel valve can only be controlled (including telemetry) from the redundant OBC, but not from the prime one. This reduces significantly the complexity of the system and thus the costs, but is also reducing the redundancy level: If the redundant fuel valve needs to be used, also the OBC has to be switched over and consequently loses its redundancy as well. Those limitations have then to be considered for the in-orbit test (IOT) campaign and also for the contingency procedures.

In technology demonstration projects more complex and advanced layouts have been tested. For example the satellite TET has a layout with two pairs of computer units that monitor each other and can swap the roles of "worker" and "monitor."

Usually the redundancy is intended for safety purposes. However, in some cases the redundant units can be used for additional activities like generating a second, independent telemetry stream.

6.2.2.3 Telemetry Parameters

The elements that contain information about the spacecraft and that shall be transmitted to ground are called the telemetry parameters (sometimes also called telemetry points). They can contain status information (like ON/OFF flags), numerical data (like temperatures or counters), or binary data (unstructured). Their value or meaning has to be coded into a binary format. In most cases it is important to save bandwidth and therefore the smallest possible coding is used: Flags can be 1-bit values; the length of the bit pattern of integer numbers depends on the value range of the corresponding parameter. Measurement values either use an interpolation table or use a standard real number format like IEEE 754. The coding used is described in the spacecraft database.

6.2.2.4 Telecommands

Telecommands are the instructions that allow controlling the spacecraft from ground. They are also defined in the spacecraft database. A telecommand has an identifier and it may have a number of command parameters that modify or specify their behavior. Commands may switch devices, set values in registers, or transport a binary data segments.

An important part of the commands is the "address" part that describes which part of the OBDH shall receive the command. This is described in Sect. 6.2.3.5. The remainder of the command is the command data. It is composed of the beforementioned command parameters and the command identifier.

6.2.2.5 Spacecraft Database

All the information on how the commands are packed into the uplink stream and how they are decoded from the downlink stream are defined in the spacecraft database. Among its elements is the complete set of available telecommands, the complete telemetry item set, the definition of packets or frames, respectively, or the scripts running on board. It contains in most cases also so-called derived parameters. These items are treated like telemetry, but their values are basically result of ground system calculations which use telemetry or ground processing information as input. The formulas behind the calculations are usually developed by the flight team according to information by the manufacturer and added to the database.

For telemetry parameter limit values can be defined in the database. Violations of these limits can be indicated in the telemetry display system with a visual or audible alarm notification, but it can also be checked later on in an off-line system [e.g., for dumped telemetry that was not processed by the real-time Monitoring and Control Software (MCS)]. An alert staging is possible when additional limits are provided that raise only prewarnings. Multiple limit sets can be provided that allow adaptation of limit values to different mission phases. “Summary alarms” can be defined that indicate a violation of any of its assigned parameters. This is needed to monitor modern spacecraft that typically have myriads of telemetry parameters.

The database can also contain information like flight procedures and telemetry display pages. Due to its central function for the manufacturer during the preparation and for the operations team during the mission the database is also called Mission Information (Data-)Base (MIB) or Mission Data Base (MDB).

The database is created by the manufacturer and is on the one hand embedded into the spacecraft computer software and on the other hand provided to the control center in an agreed exchange format.

The operations team needs to validate its ground system and operations using the database. The operations team should ideally give feedback to the spacecraft manufacturer about the usability of the database. A frequent issue is the naming convention of parameters and commands. Each entry, telecommand, or telemetry point is identified by a unique database identifier.

For operational safety it is important that the naming of parameters is done systematically and ergonomically and from an operator point of view. This can differ considerably from the perception of the manufacturer.

For example, sometimes mnemonics are assigned to the telemetry parameters that are very long and of different length. This makes flight procedures and telemetry display systems cluttered and confusing. A good operational mnemonic has between 5 and 10 characters and follows a stringent systematic. An early dialogue between control center and manufacturer is advisable to reach a common approach.

During a mission the database is normally updated to include new functions, correct errors, or react to unplanned events. However, it is of highest importance that the versions of the database on ground, on board, and in the documentation are identical. Therefore strict and formal change processes and configuration control are required.

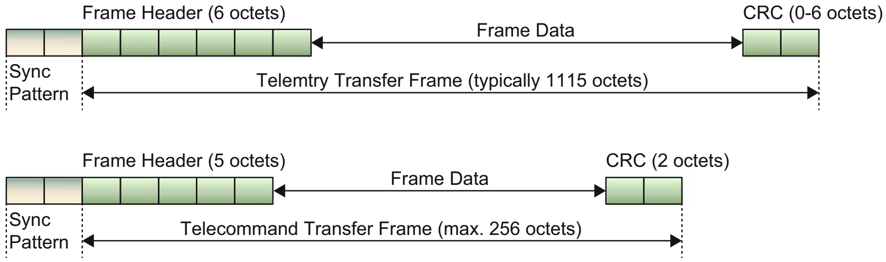

Fig. 6.13 Transfer frames for telemetry (*upper part*) and telecommand (*lower part*)

6.2.3 Space to Ground Data Streams

6.2.3.1 Data Transport

For both uplink and downlink data is transported in transfer frames. We will explain the principle mainly at the example of telemetry. The telecommand transmission works similar.

All data streams consist of very long series of bit values. This stream is structured into successive pieces of equal length called transfer frames. All transfer frames start with a non-changing “sync pattern” that allows the receiving station to recognize the beginning of a transfer frame even after interruptions. The transfer frames also contain header information that indicates the transfer frame size, a frame counter and a virtual channel identifier (Sect. 6.2.3.5). The frame trailer contains a checksum [cyclic redundancy check (CRC)] that allows detecting transmission failures. Typical examples are provided in Fig. 6.13. Standard sizes are 1,119 bytes for telemetry and 256 bytes for telecommand streams.

6.2.3.2 Frame-Based Telemetry

This type of telemetry is sometimes also called PCM (Pulse Code Modulation) telemetry format.

The telemetry parameters are assigned and distributed to a set of several different “minor frames.” These are of a fixed length and fill a transfer frame completely. The different minor frames are transmitted one after the other and repeated cyclically. Each minor frame has a header that contains the frame ID to allow identification. The complete set of minor frames is called “major frame” or “format.”

Figure 6.14 shows an example for a minor frame. Telemetry parameters are assigned to certain positions within the available data space as defined in the spacecraft database. The coding of the parameter values depends on its use and can occupy any number of bits. Mostly the parameters are grouped into and aligned to data chunks of 16 or 32 bits called words or double words.

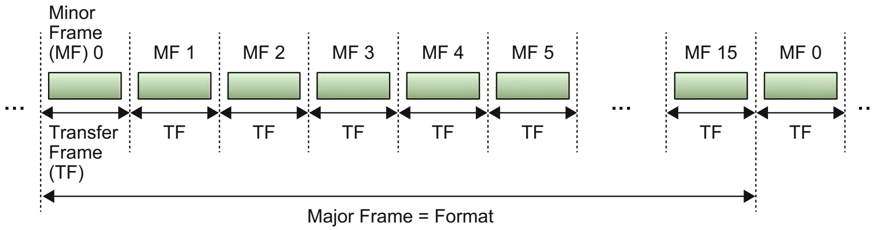

Fig. 6.14 The set of minor frames defines a major frame and is repeated cyclically

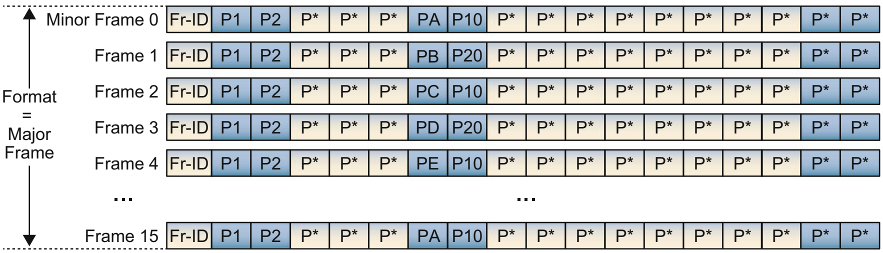

Fig. 6.15 An example for frame telemetry. P1 and P2 designate parameters that are transmitted in all 15 minor frames. Parameters PA-PP are defined only once per format (or major frame). And parameters P10 and P20 are put into every second minor frame

If each telemetry parameter is assigned to only one minor frame then all values are transmitteed only once per major format. However, it is possible to assign important and especially dynamic parameters to multiple or even all minor frames as shown in Fig. 6.15. In that way it is tried to balance the importance of the parameters with the available bandwidth. This rigid scheme can, however, not be sufficient for all operational situations. To overcome this, it is usually possible to dynamically redefine some areas during flight and thus change the selection and the sampling rate of telemetered values, which leads to various improved methods and concepts like dwell, dump, pages, oversampling, and subsampling.

Frame telemetry is a basic method that allows transferring data in a simple way. It was employed since the early days of spaceflight, but is being superseded by packet telemetry (ECSS 50-04C 2008) since the late 1990s and beginning twenty-first century by packet services like described in the ECSS PUS (ECSS 70-41A 2003), which are discussed in the following chapters.

6.2.3.3 Packet Data Structures

The increasing demand for higher data rates and more telemetry parameters as well as the general tendency to include more software functions in spacecraft design led to the need for more flexible and efficient data transport methods. A prominent

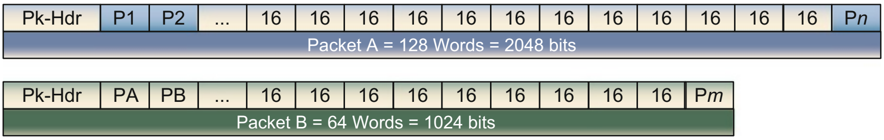

Fig. 6.16 Two example packet definitions. Typical chunks of data are 16 bit and are also called words. They contain telemetry parameters at defined positions

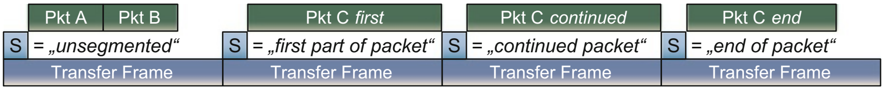

Fig. 6.17 The segmentation layer allows filling the transfer frames with parts of large packets or several smaller packets

example is the CCSDS packet telemetry and command standard (CCSDS 133.0-B-1). This standard implements many modern mechanisms of data transfer.

In this concept the parameters and commands are grouped into logical packets. Any number of packet definitions can be stored in memory. Their size can vary up to 216 = 65,536 bytes (octets) (Fig. 6.16).

The packet header contains information that allows identifying the packet and its length.

The data stream itself is still organized in frames of fixed length, which act basically as containers for the packets. As shown in Fig. 6.17 each frame contains a small element called segment, which can be considered as a management layer for the packets. It allows to multiplex several packets (or packet parts) of any length into the frame structure and thereby to distribute the bandwidth capacity among several destination devices. The telecommand segment layer can contain a sublayer for authentication. In that case an authentication trailer is added after the segment and thus the packet size is reduced. The authentication is handled by the TMTC board of the spacecraft. It protects against illegal commanding of the spacecraft.

Telemetry packets can be organized such that they are generated at a fixed rate or on request or on event. For example, command execution confirmation (or rejection) messages will be generated on event. The processing of packets is performed by the OBC and puts a relatively high load on it. It has to provide the mechanisms for buffering and organizing the telemetry stream. The frame and segmentation layers are handled on hardware level inside the TMTC board.

Telemetry packets can be switched off or on, they can be sent with different data rates—depending on the current operational situation, which makes this telemetry concept very flexible and efficient, but also complex and not very transparent.

Again, the definition of the various packets is contained in the MDB.

6.2.3.4 Packet Utilization Standard

So far we have described how information is transported, but not how it is handled on application level. To achieve a uniform approach to spacecraft control, the concept of service types was defined by ECSS in the Packet Utilization Standard (PUS) (ECSS 70-41A 2003). The idea is to not randomly write data patterns to (TC) and read others from (TM) on-board registers and interpret the result in a mission-specific manner, but to rely on standardized services for this.

These defined services are much more far-reaching than the classic TMTC tasks, which may (a little impudently) be summarized as

- Send a telecommand to a destination
- Load telecommands to the time-tag buffer
- Send telemetry to ground
- Configure telemetry

The services defined in the PUS now cover a vast field of data management functions, reaching from memory management to time distribution. However, not only standard services are defined in the PUS, but there is also room for mission-specific definitions.

This approach ensures that the manufacturer/the mission can tailor the standard for their implementation. A possible negative result is that it may happen that manufacturer-specific solutions of basic tasks may be implemented in private services and are effectively undermining the standardization.

The list of defined services in Table 6.2 shows that on the one hand the tasks are now grouped into various aspects but on the other hand it makes clear that also very advanced concepts are included that previously were only present fragmentarily or not at all. An example would be service 4 that allows statistical information about on-board data to be requested in a formal way. The service defines the necessary data structures and functions. In that way manufacturers and operators are “coaxed” towards thinking about advanced concepts. The hermetic art of spacecraft control now has an open language. Like before, however, implementing advanced services may produce a large development effort on the manufacturer and may therefore be avoided, or tailored. Standardization will make the reuse of ground systems easier, in the long run also for space systems.

6.2.3.5 TMTC and Security Management

Flow Control Mechanisms

This function is used to assure safe operations by protecting against faulty transmission of data. The first measure is to include error control and forward correction information. This is done on low level (coding and transport level) and cannot be modified during operations. The keywords are CRCs (cf. Sect. 6.2.3.1) and

Table 6.2 The currently defined standard PUS services

Service type	Service name
1	Telecommand verification service
2	Device command distribution service
3	Housekeeping and diagnostic data reporting service
4	Parameter statistics reporting service
5	Event reporting service
6	Memory management service
7	Not used
8	Function management service
9	Time management service
10	Not used
11	On-board operations scheduling service
12	On-board monitoring service
13	Large data transfer service
14	Packet forwarding control service
15	On-board storage and retrieval service
16	Not used
17	Test service
18	On-board operations procedure service
19	Event-action service

randomization (a deliberate coding of data to assure bit synchronization). The operational tasks are limited to configure the MCS on ground to accept or reject faulty telemetry frames for diagnostic purposes and to monitor the uplink frame quality that is reported in low-level telemetry generated by the TMTC Board.

The second measure is to introduce a quality of service (QOS). This is used to recognize and recover data lost in transmission. The basic means for this are counters. Frames and packets carry counters that are tracked and checked for gaps.

In the uplink channel this can be a closed-loop process by using the COP-1 protocol as described in ECSS 50-04C (2008). This mechanism is implemented on transfer frame level. The control center can choose to transmit either in AD (“acceptance”) or in BD (“bypass”) mode. Simplifying a little bit, in AD mode the commanding process waits until it receives the uplink confirmation of the previous command before it transmits the next one. Some MCSs allow an automatic retransmission of lost commands. Apart from synchronization tasks the command link is not necessarily slowed down compared to the BD mode if the “sliding window” mechanism (ECSS 50-04C 2008) is applied where a settable number of frames is uplinked before the confirmation is received. Obviously the AD mode can only work if telemetry is available. For blind transmissions or unstable connections the BD mode has to be selected.

This mechanism is called FARM (Frame Acceptance and Reporting Mechanism) (ECSS 50-04C 2008) and is done on low level in the TMTC Board. For telemetry no such mechanism is standardized. A possible retransmission has to be done on MCS application level or manually initiated by control center staff.

Routing Mechanisms

The established standards allow to precisely address source, destination, and the route of telecommand and telemetry data units. ECSS (ECSS 50-04C 2008) defines the following qualifiers:

- The Spacecraft ID (SCID) is a worldwide unique number that protects against uplink signals intended for other spacecraft (sana 2014). It is important to note that simulation systems, engineering models, etc. usually have separate IDs. The correct selection may be a source of problems during the switch from simulation to the mission.
- The virtual channel (VC) ID in the uplink path distinguishes if a signal is intended for the prime or the backup decoder (TMTC Board). Only the addressed decoder will forward the command. Note that this flag is set in the ground command system. Normally it is independent from the spacecraft database and should be easily changeable. In the downlink the VC allows to interleave different data channels that may be processed by separated ground systems. The low-level implementation allows separating the channels without knowledge about the spacecraft database. This is commonly used for data dumps (Sect. 6.2.4.5) that are processed on ground in a different way or by different control centers.
- The multiplexing access point (MAP) ID is analyzed within the TMTC Board and directs the command to different devices. Usual destinations are the CPDU (high-priority telecommands; see paragraph below), the Authentication Unit, and the prime and backup OBC. Most telecommands will be accepted only at a specific destination, but in the case of OBC MAP IDs this can be used for addressing the backup OBC over cross-strapped connections (Fig. 6.13). Like for the virtual channels, this parameter can usually be set dynamically in the MCS. Alternatively backup commands can be defined in the spacecraft database for this purpose.
- The Application ID (APID) is used to specify the destination on packet level. It is evaluated by the OBC.

Authentication

Another mechanism that is located inside the TMTC Board is the authentication. This is used to ensure that only the legitimate control center can control the spacecraft. The uplink stream is usually not encrypted, but encrypted signatures are attached to the segmentation layer. The signature includes an encrypted counter to protect against replays. A set of secret emergency keys is implemented into the authentication device on board and it is possible to upload more keys for daily usage. There are special telecommands available to control the mechanism.

Flow control, routings, high-priority commands, and authentication mechanisms are nicely described in MA28140 Packet Telecommand Decoder (2000).

Encryption

For military spacecraft, but also increasingly for civil applications, it has become usual to encrypt either the complete or parts of the data transmission. There needs to be an encryption device and a set of keys on board. To enhance the protection the keys have to be changed in regular intervals. It depends on the implementation how this is exactly handled and even where the encryption device is located inside the OBDH. Two main usage scenarios are common:

- Encrypt the complete data stream. The en-/decryption on ground side takes place in the control center.
- Encrypt only payload data. The en-/decryption on ground side may take place in the user center. This may even be done with multiple users where each user has his or her own set of keys and can only extract his own data.

6.2.4 OBDH Management

6.2.4.1 General

Operating the data-handling subsystem is under normal conditions mostly embedded into the routine mission tasks. During mission preparation and in case of anomalies or special campaigns, however, the complexity of modern spacecraft makes this a broad and demanding field. The work related to the spacecraft database tasks mentioned in Sect. 6.2.4.3 may occupy several persons including the data-handling expert. During mission preparation the MCS of the control center needs to be adapted; a task that is heavily connected to knowledge about the OBDH system. During mission execution the functions of the MCS needs to be understood to fully make use of the spacecraft abilities, especially during non-nominal situations. In the following chapters the basic operational tasks related to the OBDH are explained.

6.2.4.2 Safeguard Mechanisms

Fault Detection, Isolation, and Recovery

To protect a spacecraft from damage or loss of mission, usually fault detection, isolation and recovery (FDIR) mechanisms are provided where a robust supervising unit monitors a functional unit. It is vital for safe operations to understand this mechanism and to be able to interpret the telemetry correctly and quickly as the knowledge is mostly needed in critical situations.

The FDIR functionality may be spread over several components like the reconfiguration module (cf. Sect. 6.2.2.1) and the OBC. The layout will be different for each satellite platform using different approaches and levels of severity.

Functionalities can be installed using hardware or software or both. Usually failures that are not threatening to the spacecraft health are not handled autonomously and are left for ground (control center) interaction. Errors that can be corrected by the unit itself are usually reported but otherwise ignored by the higher level FDIR, e.g., the EDAC (error detection and correction) correction of memory bit flips using available redundant information. This philosophy is, however, currently changing with the availability of event action services like described in the PUS (ECSS 70-41A 2003).

Basically FDIRs are monitoring system properties (telemetry values, bus voltages, hard-wired signal inputs, keep-alive signal, etc.) for deviations beyond allowed limits, report about the incident and execute a predefined action: The surveillance defines a list of one or more health check parameters (HCP) taken either from normal on-board telemetry buffers or from dedicated signal lines. Their values are monitored with a dedicated frequency (e.g., with 1 Hz) and compared with defined and possibly configurable thresholds.

Depending on the device and its criticality, a single or repeated (after n consecutive) parameter limit violation of one or more parameters (e.g., majority voting) then triggers the FDIR situation and the corresponding action, e.g., to switch to a redundant device.

This occurrence is usually registered in some kind of error log, but in some cases it has to be deduced by the ground team from the telemetry: In emergency cases, the memory including the log could be erased by OBC reboots, telemetry was not set to be recorded or there was no ground visibility. For this reason e.g. geostationary satellites configure their telemetry to send out the error log file continuously. In the worst situation, the error log file might be the last signal the control center sees from the spacecraft.

Usually it is possible to configure the monitorings and actions of the safeguard mechanisms. In some cases notifications shall be issued, but no action be triggered. If redundant equipment is activated by an FDIR, usually there is a different and limited FDIR on the redundant device. This shall usually stop the system from switching back to a presumed faulty prime equipment in case of a second or persisting external trigger. For certain operational tasks is necessary to disable some FDIRs (e.g., during an orbit maneuver). This will typically be covered by the according flight procedures.

The recovery actions can consist of one or more discrete steps. It is common to use some “macro” functionality, releasing a sequence of individual commands on board. This can again be implemented as software (potentially as On-board Control Procedure wrapped into a PUS service cf. Sect. 6.2.2.4) or hardware. The device for the latter case is called a reconfiguration module (RM). (Electro-)mechanical registers send out command signals over dedicated cables. Software macro command mechanisms usually use standard command packets (like the ones sent from ground).

Table 6.3 shows an example for an emergency macro command sequence of a geostationary satellite. In many emergency situations the spacecraft is put into a stable safe mode by the AOCS computer. This mode is called sun acquisition mode

Table 6.3 The main steps of a SAM macro command sequence

Step	Title	Description
1	Disable all ground-triggered or higher level macro commands	Avoid contradicting actions
2	Switch off payload	Service is interrupted, avoid useless power consumption, avoid uncontrolled RF spilling
2	Adapt battery charge control	Adapt to new electrical situation
3	Start thermal regulation	Adapt to the new thermal situation
4	Activate substitution heaters	Because payload is switched off, use heaters to avoid thermal imbalance
5	Switch off payload-related equipment	Reduce power consumption
6	Reconfigure the BAPTA to reset point	Rotate solar arrays to $-Z$ axis (=sun pointing) and stop rotation
7	Reconfigure telemetry	Reset to emergency configuration (display the error log file, show basic telemetry)

(SAM) because it reorients the spacecraft to the sun and starts a rotation around this direction. A FDIR mechanism inside the OBC detects this mode and adapts the satellite to the new situation. The shown steps are executed. The real sequence contains more than 50 discrete commands and delay statements. If redundant commands are available both are used to make the sequence more robust.

The recovery sequences are reprogrammable. But this option is used very rarely; usually this is the domain of the manufacturer. Changes should be done only cautiously and after consultancy of the spacecraft manufacturer. The correct content of the sequences should be checked regularly, however.

The operational tasks are to monitor for occurrence of an FDIR. This day-to-day supervision can largely be handled by automated telemetry alerts and configuration checks (concheck). Note that for LEO satellites the telemetry checks have to be done off-line, as large parts of the telemetry are stored on board and are dumped in compressed format. They have to be unpacked and processed on ground. This is mostly done in separated off-line data processing systems.

Safeguard Memory

The components and the software used for spaceflight applications are very durable and well tested. The FDIR mechanisms and the redundant layout add to this. Nevertheless it can and will probably happen during the life time of a spacecraft that the OBC has to be restarted either by ground command or after an anomaly that triggered an according FDIR.

Like for normal PCs the RAM memory is cleared during a restart. The OBC software will be loaded from a PROM or EEPROM and is configured for a default standard spacecraft configuration (typically the launch configuration). The configuration set contains items like momentum wheel selection, redundant device selection, etc.

If during the mission a redundant device was activated due to failure of the prime equipment, then the default software configuration does not take this into account, which may lead to another triggering of the reboot or to a dangerous situation for the spacecraft. Therefore a device is included in the design of the OBDH called SGM. This is a protected register. It will not be erased during power failures and is especially guarded against radiation of other environmental influences. From this memory the OBC can take current settings that may be different than the default settings. Depending on the complexity of the spacecraft this can be a few fundamental setting or elaborate data areas. Configurations that are relevant on hardware level may be stored in electromechanical configuration relays. This may be the on/off status of the authentication mode of the TMTC Board.

The content of the SGM has to be updated after each relevant reconfiguration of the spacecraft. The according activity is normally part of the flight procedures. Depending on the abilities of the OBC it may be a tedious, time-consuming task. Nevertheless the update shall not be delayed too long. After the update it is good practice to dump the SGM content to ground in order to check the correct writing. Also the operations team should have enough information to interpret the content of the memory dump. The possibility to change the content of the SGM is protected and special commands are needed to allow and afterwards disable write access.

6.2.4.3 On-Board Software Maintenance

Modifying the Mission Database

Two different aspects of the Modifying the Mission Database (MIB) have to be considered: the spacecraft implementation and the ground implementation. Of course both have to be coherent at all times. Nevertheless it is not necessary to update the on-board version each time the ground version is changed, since the on-board part can be considered a subset of the ground MIB, as explained below.

The on-board side has the task to define how to unpack telecommands from the uplink stream and to which destination device to forward them to. A possible change might be to reduce the list of accepted telecommands. This can be a measure to protect against unwanted commanding from ground. Also a change of the interpretation of telecommands is possible, however, only in conjunction with an OBC software update. For example when the on-board software for time-tag register management is updated, potentially the new software is able to accept new telecommands; the on-board database needs to be updated to accept the new command pattern.

The most common change, however, is the change of the definition of telemetry packets. This is expected to happen mostly for new spacecraft without sufficient operational experience where the initial definition may need to be adapted after a while or after anomalies or failures when the existing observability may not be sufficient. Since telemetry packets can be very long and may consume a large portion of the bandwidth, it may under certain circumstances be better to define a small diagnostic telemetry packet for the particular situation that can be then sent down at high rates.

In all cases the ground database has to be updated accordingly.

Mostly the on-board database is implemented as integral part of the on-board software and is not modifiable separately. Some aspects like new TM packet definition can be modified using PUS service 3 (Reporting Service, cf. Sect. 6.2.3.4). The persistency of updates and service requests after OBC reboots has to be checked.

The ground system database implementation includes many aspects of the ground system that are adaptable and do not affect the space side:

- Including commands, which are already available on board, but were originally not allowed for the control center
- Introducing new commands that are just redefinition of complex, already existing commands with many specific parameters into a simple short command or that allow only a reduced parameter range
- Modifying the calibration of parameters (e.g., status texts) for telemetry items or telecommands
- Adapting the telemetry ground limit entries
- Introducing or modifying derived parameters or ground parameters (e.g., ground station telemetry)
- Introducing telemetry parameters for improved and safer operability (e.g., define the bits of "status words" as separate parameters giving them text calibrations)
- Modifications of display page definitions or command sequences that are embedded in the MIB

Mostly the changes have to be coordinated with and agreed by the spacecraft manufacturer. Strict versioning and configuration control should be in place. Changes have implications on the flight procedures and the documentation. The effort can be considerable. Database tools can help in keeping flight procedures and the database consistent.

Whether the modification of limit entries (alerting thresholds as described in Sect. 6.2.2.5) in the active MCS during run-time is allowed under configuration control is debatable. Depending on the work practice established at the control center and especially for geostationary satellites a relaxed approach is possible. Remember that the alert mechanism mainly serves the purpose of situational awareness for the first-line operator. It may make sense to adapt the limits temporarily to avoid flooding the operator's attention. A compromise is to use temporary database overwrite mechanisms during run-time of the MCS system. This preserves the original database in the configured installation. This situation should be cleared during the next regular database update. Database modifications are considered critical and should be tried on a test or simulation system beforehand.

OBC Software Maintenance

When the manufacturer provides a new software version for the OBC, the necessary operations tasks are the uplink of the software data, the management of the software image on board, and the boot process.

Typically the software is split into smaller parts and embedded into data transfer commands. All uploaded data has to be checked for completeness and correctness when it is reassembled on board using checksums or by dumping all data back to ground with subsequent comparison with the original one. The software then needs to be transferred to the memory area of the OBC that will be used during the boot process. It is important to make sure that the original software can still be accessed in case there is a problem with the new software during the boot process that makes it impossible to communicate with the spacecraft. There are several possible methods to prevent this and the spacecraft manufacturer has to provide a corresponding procedure. When the first boot and the IOT of the software were successful, the new software needs to be configured as permanent boot software.

All these aspects can be handled by advanced memory management services like provided by the PUS or by using conventional telecommands.

6.2.4.4 Execution Management

Time-Tagging

Telecommand time-tagging is used to issue pre-loaded commands at specific times when either the spacecraft is out of sight of a ground station or there is the possibility to lose the command link to the spacecraft, or when exact execution timing is required.

Nearly all telecommands can be time-tagged and the ECSS Standard (ECSS 50-04C 2008) provides a method for it. Excluded from this are the hardware decoded high-priority telecommands that are handled by the TMTC Board. If a Time-Tagged Telecommand (TTTC) is received it is captured by the time-tag software (or schedule handler for the PUS spacecraft) and put into a memory table. This table contains the defined execution time and the actual telecommand. When the on-board time equals the execution time the command is released to its destination. The time-tag table (or schedule) can have restrictions and peculiarities. In some spacecraft it is possible to have more than one TTTC for the same execution time. Some spacecraft allow only adding time-tags to the end of the list and in chronological order, thus forcing complete deletion and resending of all TTTCs when a new one shall be inserted in the middle. Some designs use register numbers to store and address memory locations.

The extent and scope of usage depends on the mission needs. A geostationary satellite has nearly permanent ground contact and needs time-tagging only rarely:

- To put potential rescue commands a little into the future to protect against possible lockouts in the uplink path (including ground failures). These TTTCs will usually be erased or updated when no contingency occurred
- To ensure nominal commanding to protect against loss of uplink signal
- For precise timing of commands (a rather rare thing on classical communication satellites)

Consequently geostationary satellites need to store only a small number of time-tagged telecommands (less than 100).

Low-earth orbiting spacecraft with only few ground contacts or interplanetary missions with long signal propagation times depend heavily on this capability. Therefore they have hundreds or thousands of slots for TTTCs. Two aspects are of special importance here:

- To switch on and off the ground link (telemetry) during the predicted contact times to optimize the available contact time. An example case is described in Sect. 6.1.5.4.
- To control the payload. This can be the timing of payload operations and payload data collection.

The mission timelines of LEO satellites and interplanetary probes are rather complex. The rudimentary operations provided by earlier spacecraft platforms were making operations very demanding. For this reason the PUS service 11 (Scheduling, ECSS 70-41A 2003) introduced advanced methods for this. This standard foresees to define subschedules than can be activated, edited, time-shifted, and deleted separately. It is even possible to interlock schedules providing a way to make progression to the next TTTC dependent on the success of the previous step and so on.

To keep track of the schedule activities it is recommended to use a ground tool that models the schedule and can compare this model with a schedule dump during the next visibility period. Also advanced ground systems allow matching stored and dumped command execution confirmation (or rejection) events with the telecommand history of the command system and displaying any mismatches.

The time-tagged commands and the schedule are of course dependent on an available time-source on board. If the time becomes unreliable for any reason (e.g., an OBC reboot), then the time-tag registers will be erased.

On-Board Control Procedures

Advanced spacecraft allow storing complete sequences of commands on board. This is similar to the schedule mechanism without a predefined execution time. The procedures remain in memory and can be reused and adapted. This can reduce the bandwidth demand in the uplink channel drastically, when repeating procedures are very long and only a few key parameters have to be changed. Again the PUS standard has a service for this (PUS 18). Together with event monitoring and action services (PUS 5, 12 and 19) it gives the spacecraft a high degree of autonomy.

6.2.4.5 Mass Memory Management

The management of mass memory storage devices is of concern mainly to missions in LEO or on extraterrestrial missions that have to store payload data.

In earlier times data was typically recorded on tape recorders e.g. for the Voyager mission to the outer solar system. Only in rare cases mechanical hard disk memory drives were used (Bussinger et al. 1993). In recent times the predominant design is to use solid-state memory.

Different storage concepts have been devised.

Ring Buffer

A ring buffer is a robust and simple mechanism. That is ideal for continuous data streams that can be expected to be cleared in a predictable regular interval and also when their handling character is FIFO (first in–first out). Here the physically available (linear) storage is modeled in ring shape and thus creates an apparently infinite memory range (Fig. 6.18).

There are typically two or more pointers:

- Write pointer. This is the address of the physical storage cell that will be used for the next write action. After writing one record the pointer is incremented (and of course at the end of the physical memory range turned around to the beginning of the range). It is a matter of mission design if the write process is stopped when it reaches or overpasses the delete or read pointer. A monitoring of the distance between the pointers may be necessary for the ground control team. Possible measures against this situation would be to throttle down the write data rate or to arrange for additional downlink time.
- Read Pointer. This is the address of the physical storage cell the next data will be read from. After the reading action the pointer will be incremented (or turned around as above). A potential overpassing of the write pointer is not a problem, but leads to a duplication of already downloaded data which has to be handled in the ground system. The read pointer may be pushed by the write pointer. The reading process may be triggered from ground or it can be an automatic continuous process. There may be several read pointers that can be used by different users.
- Delete Pointer (optional). This can be a pointer that lags behind the read pointer and indicates ranges that the ground control center has successfully retrieved and approved to be deleted. The delete pointer may be a barrier for the write process. In other implementations this function can be taken by the read pointer.

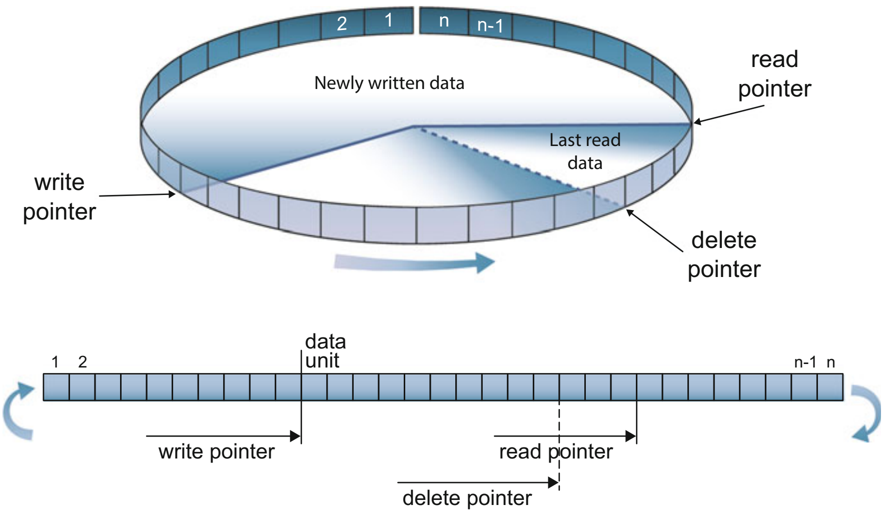

Fig. 6.18 Ring buffer model. The linear memory cells are accessed as a circular range. Each time a pointer reaches the end of physical memory it is reset to the beginning

However, if a mission cannot risk to lose data a ring buffer concept may not be the ideal solution.

The application process and the operators on ground (or the mission planning process) need only very little knowledge about the storage. The difference between read, delete, and write pointer define filling grades. Also it will usually be possible to modify the read pointer e.g. to repeat an unsuccessful dump action.

File System

A file storage system replicates data structures that are similar to the known data files systems in PC computers. These files are linear data storage areas that are of a certain common nature. Files can be created, named, filled, closed, read, copied, modified, concatenated, and deleted on board. They can be uploaded from ground and downloaded (dumped) to ground. They have a unique handler or name and they may have common attributes like file create date, last modified date, and access rights or other meta data attached to it. A very appropriate use for this is the handling of on-board software patches or instrument measurement items (images, data takes) that are of a piecewise nature. It is also possible to fill files with continuous datastreams like housekeeping telemetry, but this is currently quite uncommon and has no special advantage over a linear buffer or a ring buffer. The necessary operational tasks can be all of the abovementioned file methods. Partially they may be taken over by automatisms.

Linear Storage

Linear storage is the third method for large storage devices, especially if they are used for housekeeping telemetry or continuous measurements. It is preferred over the ring buffer if the stored data must not accidentally be overwritten. Usually also here pointers for reading and writing are used but also random access methods can be employed. If the reading, writing, and deletion management is fully automated on board, then no difference to a ring buffer exists. Therefore the strength of the linear buffer lies in the random access and the protection of written data which implies a manual data management. Also per definition a linear store will be full at a certain point in time and a new store has to be used until the old store has been completely read and can be emptied. The operational tasks will be to direct the data stream to a selected store, to monitor the filling level, to control the reading process (possibly repeating unsuccessful dumps), and to reset the pointers to zero when a store is full and has been read completely.

Advanced mass memory concepts can define several instances or combinations of these storage methods. These storage areas (sometimes called “stores”) may be filled with data from different sources or it can be sampled at different intervals or be targeted at different ground destinations.

The management of the data that is written to the storage usually involves the mission planning and the data-handling expert. They have to define what to store and at what rate. This has to be checked against the available memory and the time to the next downlink occasion. The process to download the stored data (payload or housekeeping) is called dumping. Higher data rates may be used. In order to save downlink bandwidth and mass memory it is possible to use data compression techniques (Evans and Moschini 2013).

Memory maintenance is usually not necessary for solid-state memory. Typically it will be equipped with EDAC mechanisms. Tape recorders’ reels had to be rewound regularly to avoid tape-sticking and data sections had to be regularly reformatted and rewritten (in different arrangement) to avoid magnetic fading and copying effects.

6.2.5 Summary and Outlook

The OBDH and its operation have changed fundamentally in the last decades as the on-board storage and processing capabilities have increased dramatically. This chapter presented the fundamental activities that are connected with the operations of the Data-Handling Subsystem.

A few trends are obvious to increase in the near future. From the end users there will be an even higher demand for data storage and for data protection. The mission complexity and the shorter development cycles will demand more on-board flexibility and autonomy. This will result in more powerful computer hardware and advanced software on board and therefore, like on ground, a higher demand for software maintenance and adaptation during flight.

Standardization of data access and maintenance methods as described in the Packet Utilization Standard (ECSS 70-41A 2003) is used to streamline the systems and methods. This process is still ongoing, but already it has shown positive harmonization effects that in the end all parties involved will take profit from.

6.3 Power and Thermal Operations

Sebastian Löw, Kay Müller, and Sina Scholz

6.3.1 PTS Design Aspects

6.3.1.1 Power System

All electrical systems require electrical power in order to function. For most stationary electrical applications on Earth this power is generated by an electric generator which basically consists of a conductor rotating in a magnetic field, using the physical effect of electromagnetic induction. The means to move the conductor are to a high percentage obtained by converting heat into mechanical energy with the heat originating i.e. from chemical energy contained in combustible substances such as coal or oil or gas. Another possibility is to use mass defects of atomic nuclei. For convenience, many appliances are spatially separated from their power supply. Energy is provided by a power plant located kilometers away using one of the abovementioned means to generate energy which is then transported to its various consumers via the electrical grid.

Whenever mobility is involved, the means to provide power need to become smaller and lighter in order to be able to be taken along. Also the separation of power source and consumer is no longer present. Like many power plants, cars or aircraft convert chemical energy contained in fuel into thermal and then mechanical energy via combustion. Their engines are significantly scaled down versions of the large power plants. For even smaller devices such as cell phones and laptops, batteries become the power supply of choice since they can store a limited amount of energy.

For space missions the concept of mobility becomes especially severe. Spacecraft cannot be refueled like airplanes or cars or recharged by plugging them in like cell phones. Therefore, all the elements necessary to provide electrical power have

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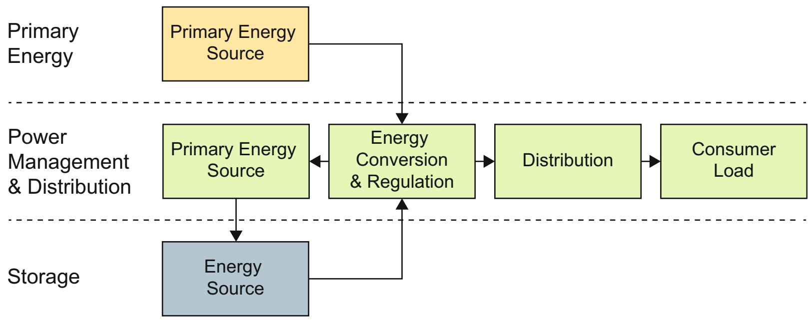

Fig. 6.19 Main functional blocks of a power supply system

to be put on board and designed to be sufficient throughout the desired mission time. All approaches to spacecraft power systems are variations of this theme with the following commonalities:

- Electrical power usually needs to be generated on board (Primary Energy).
- It needs to be conditioned, which means to be converted and regulated, to suit the needs of the various units, specifically operating voltages and currents. Some units might require a constant voltage for example. This usually leads to separate power buses,[1] regulated and unregulated, which connect their respective units (Power Management and Distribution).
- The spacecraft might need more power than the primary power source can provide and must also be able to function when no primary power is available at all. Therefore, the possibility to store energy must be implemented. This is normally done by using batteries which makes some sort of charge control necessary. Stored energy can then be fed back into the system at a later point (Storage).

A schematic drawing of a spacecraft's power system can be seen in Fig. 6.19.

Energy Sources and Storage

The basis of each energy supply is a physical effect which is exploited technologically. Examples are numerous; see Table 6.4.

[1] In general, a spacecraft bus consists of a Command and Data Handling System, a Communications system and the appropriate antennas, an Electrical Power System, a Propulsion System, Thermal Control, an Attitude and Orbit Control System, a Guidance, Navigation and Control System, as well as a structure and trusses that holds everything together. A subset is the power bus mentioned here which connects the solar panels and battery with a unit called the Power Control and Distribution Unit (PCDU).

Table 6.4 Physical phenomena used for the generation of electrical power in spacecraft

Physical effect	Application
Photons increase the energy level of electrons in semiconductors	Solar panel
Chemical reactions between electrodes and electrolyte move electrons from one electrode to the other	Battery, fuel cell
Temperature gradients between different electrically conductive materials create an electrical potential (Seebeck Effect)	Radioisotope thermoelectric generator (radioactive decay as heat source)
Nuclear fission of heavy elements produces less heavy elements with lower mass per nucleon. The difference in mass is released as kinetic energy of the fission products	Nuclear fission reactor

Solar Panels

One widely utilized option for power generation is the photovoltaic effect. The principle of photovoltaics, literally "voltage through light," makes use of the fact that electrons in semiconductors can be excited by photons—such as solar irradiation—to a higher energy level but cannot "go back" due to a built-in junction in the semiconducting material. As such a solar cell is a charge separator. The electrons can then move through an external circuit driven by the electrical potential created by the separation of the charges. The potential and thus the voltage are dictated by the semiconductor materials that are used.

For initial estimates for satellites orbiting the Earth, a solar irradiation of $S_{\mathrm{Earth}} = 1{,}367\ \mathrm{W/m^2}$ can be assumed. Given a desired electrical power P_{Earth} to be generated and an efficiency μ one can approximate the required solar array area to be

$$A = \frac{P_{\mathrm{Earth}}}{\mu S_{\mathrm{Earth}}}. \tag{6.1}$$

The solar power per square meter at other distances r from the sun can be computed by including the square of the distance ratio:

$$A = \frac{P}{\mu S_{\mathrm{Earth}} \left(\frac{r_{\mathrm{Earth}}}{r}\right)^2}. \tag{6.2}$$

Thus, at twice the distance from the sun only a quarter of the power is available per square meter and the area of the solar panel has to be increased accordingly to reach the same electrical power.

In order to generate a desired voltage a solar panel consists of multiple solar cells which are electrically connected in series—whereas the voltages of the individual cells add up. One such series of solar cells is referred to as a "string." To reach a certain desired current output multiple strings are connected in parallel. In order to avoid damage of the solar cells by a reversed current (which occurs when a battery is discharged), each string is usually decoupled from the bus using a diode. Modern solar cells have efficiencies of more than 30 %. Since there is no filtering of sunlight in space they can generate more power than similar cells on the ground. The panel temperature has a rather large influence on efficiency. Low temperatures yield more power output. This is the case because at lower temperatures the band gap is larger which means electrons that have been excited by incoming photons have received relatively more energy which in turn results in a higher voltage. The loss of power at the maximum power point is about 0.5 %/K. High temperatures of up to about 130 °C cannot be avoided though, as the solar panel is naturally exposed to solar irradiation. With increasing age, degradation will also become an issue.[2] A loss of efficiency of 0.5 % per year can be assumed for crystalline cells even though the real value will usually be below that. For thin film solar cells the degradation is much more rapid in the first 1,000 h, up to 25 %. Almost no degradation occurs after that, though (Deng and Schiff 2002).

Electrochemical Cells: Batteries

Similarly to a solar cell, an electrochemical cell generates a voltage by separating charges. The process by which that separation occurs is in this case a chemical reaction.[3] The energy needed for that charge separation is contained in chemical energy. There are electrochemical cells in which the chemical reaction can only work in one direction. The original state can thus not be restored and the cells can only be used once. These are called primary batteries. A type of battery whose chemical reaction can work in both ways is called a secondary battery. In these, the reversal of the current restores the chemical substances to their previous state (charging) so that the charge separation can start anew (discharging). The familiar term battery refers to the combination of several electrochemical cells.[4] Like solar cells, electrochemical cells are connected in series (a string) to generate a higher voltage than each cell individually could provide. These strings are then connected in parallel in order to reach a desired current. A main figure of merit for batteries is the amount of ampere hours they can hold.[5] Almost always, electrical energy generated by solar cells is being used to charge the batteries so that the energy that is stored in the battery can be used at a later point when no sun is available or more power is needed. A surplus of solar-electrical energy is usually designed into

[2] Destruction of the semiconducting material but also degradation of the covering glass.

[3] Oxidation (loss of electrons) and reduction (gaining of electrons) in each half cell (one electrode and its surrounding electrolyte).

[4] Although in colloquial terms a battery can also refer to a single cell like the ones you can buy in the supermarket. Even those often consist of more than one cell.

[5] For example from a 100 Ah (ampere hours) battery a current of 1 A could be drawn for 100 h, or a current of 100 A for 1 h (or combinations thereof).

the power system so that the solar arrays can both supply power for the spacecraft loads as well as power for charging the battery at the same time. The battery then also needs to be designed such that it alone can supply the spacecraft loads when no solar irradiation is available. Peak loads that can be much higher than the usual ones need to be taken into account as well.

In principle, however, the charging/discharging process cannot go on indefinitely. Since the battery hardware itself, specifically the electrodes, take part in the chemical reaction the electrode material degrades with time. Degradation can occur both chemically as well mechanically. For instance, the electrode material dissolves in the electrolyte and the electrode material becomes damaged due to the constant in- and out flux of ions. Degradation occurs in all types of batteries. To some extent, measures can be taken to limit these effects. Accurate temperature control for instance can help limit side reactions that produce no electricity and only degrade the electrodes. Performance can also be improved by making sure all the cells perform similarly (cell balancing). If some cells perform worse than others their charge/discharge cycle is shifted to lower charge levels which means these cells are already fully charged during the charge process when the others are not. Since the rest of the cells are not full yet the charging process continues while the "bad" cells are being overcharged which in turn become warm and affect the battery as a whole.

An older type of battery that is still in use is the NiH_2 battery.[6] It is well known for being long lived as well as rather robust over a relatively large temperature range. However, it suffers from a lower energy density (J/kg) and self-discharging while the energy efficiency is 85 %.[7] A more modern battery that does not have those downsides is the Lithium-Ion battery. Such a battery is comparably lightweight, has a high energy density, and experiences virtually no self-discharge. However, it is sensitive to high temperatures and high depths of discharge. Its efficiency is between 80 and 90 %.

Electrochemical Cells: Fuel Cells

In a conventional battery the chemical reaction takes place between the electrolyte and the electrode material. Another type of electrochemical cell is the so-called fuel cell. It can be distinguished from batteries in that the chemically reacting substances are not part of the battery structure such as the electrodes but are added from the outside like fuel, hence the name. Fuel cells therefore do not suffer from degradation but they have the disadvantage that the reacting substances have to be stored in

[6] First used in 1977 on the NTS-2 satellite.

[7] Energy efficiency defines how well the stored chemical energy is converted into electrical energy.

additional tanks. Almost exclusively, hydrogen and oxygen are used with hydrogen as the fuel[8] and oxygen as the oxidizer. In this case the efficiency is up 60 %.

Radioisotope Thermoelectric Generators

While solar arrays in combination with batteries make sense when a spacecraft is relatively close to the sun, the required area for generating the same amount of power increases with the square of the distance to the sun, as indicated by Eq. (6.2). Therefore, most spacecraft for explorations beyond the orbit of Mars[9] use a different power supply called a radioisotope thermoelectric generator (RTG). The device contains radioisotopes, usually Pu-238 due to its long half-life of 87.7 years. The decay of the plutonium heats up the surrounding casing and thus causes a temperature gradient which according to the Seebeck effect generates an electric voltage. This system has several advantages as it can provide power for decades, has no moving parts, and is therefore autonomous and maintenance free. Its greatest disadvantage is the low availability of Plutonium and the associated high prices. Also, the thermal efficiency of an RTG is at best 8 % rather low.[10] Due to large area contamination threats in case of a launch failure RTGs are often criticized by the public. These are designed so that they can withstand such an event, though. Famous examples of spacecraft to use RTGs are Cassini, Voyager, as well as the Mars Science Laboratory Curiosity rover.

Nuclear Fission (and Fusion)

The most powerful energy source in practical use on Earth today is the process of nuclear fission. Only a few tests in the direction of nuclear powered spacecraft were made, although a nuclear reactor as a primary power source would represent an enormous step forward for many types of space missions: The amount of power generated on board determines among other things what kinds of and how many payloads can be taken along. A reactor in combination with an electric propulsion system could potentially also provide an advantage when it comes to improving travel times within the solar system.

Using nuclear fission or fusion, energy is gained by making use of a physical effect that directly converts mass into energy. The mass of an atomic nucleus is slightly lower than that of the sum of its nucleons (protons and neutrons). Therefore,

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[8] Hydrocarbons are also used as fuel, but the main differentiator is the type of electrolyte used. These range from solutions to polymers and ceramic oxides.

[9] The Juno mission to Jupiter launched on August 5, 2011, is a notable exception. The spacecraft is powered by large solar arrays.

[10] NASA is working on a combination of radioisotopes as heat source and a Stirling engine for the energy conversion called a Stirling Radioisotope Generator (SRG). Such a system promises a fourfold increase in efficiency to about 30 %.

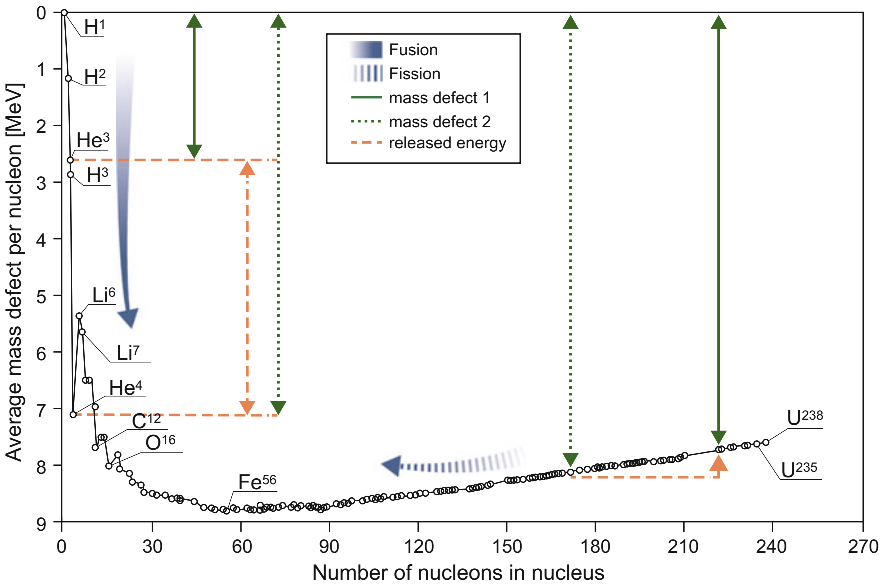

Fig. 6.20 The principle of the mass defect and the possibility to use it for energy generation. The average mass defect depending on the number of the nucleons is shown. Higher mass defect is equivalent to lower average mass per nucleon. Moving from nuclei with low mass defects to nuclei with high mass defects leads to the release of energy. Conversely, moving from nuclei with high mass defects to nuclei with low mass defects requires added energy

average of the mass per nucleon is also lower. This is called the "mass defect." It depends on the number of nucleons, i.e., the element and isotope. As can be seen in Fig. 6.20 the mass per nucleon first decreases with the number of nucleons in the atom and then increases again.

It is possible to use these differences in average nucleon mass of different elements and isotopes technologically for the generation of energy. If the nuclear reaction happens in the direction from heavier to lighter nucleons the difference in mass is released as the equivalent energy $E = \Delta mc^2$. Splitting heavier elements into lighter ones is called fission and combining lighter elements to form heavier ones fusion. Fusion can theoretically provide about a factor of ten more energy than fission. This is due to the fact that the mass defect increases much more rapidly for lighter elements, as seen at the very left of Fig. 6.20.

"Mass Defect" and "binding energy" are to be understood as synonymous since the binding energy is the energy equivalent of the mass defect. The binding energy is not part of the "energy content" of nucleus. It is the energy that would have to be added in order to separate the constituent particles of that nucleus. The added energy is then used to make up for the difference between the lower mass of the nucleons in the nucleus and the higher actual mass of the protons and neutrons in their separate state. The same holds true when different elements are compared. If Helium-3 were to be generated out of Helium-4 (fission) the difference in mass (or mass defect) would have to be added as equivalent energy. Similarly, as stated above, that same amount of energy is released in the opposite fusion reaction.

Ultimately responsible for the shape of the curve seen above is a combination of the strong interaction and the electromagnetic interaction. The strong interaction is what holds the constituent elementary particles of protons and neutrons together due to the different "color." A residual strong force is what holds together the nucleons in an atomic nucleus. Due to this attracting force there is a potential trough—energy would have to be added to separate the particles or, the other way around, energy is released as the particles enter this bound state. This released energy is the "binding energy" and the cause of the mass defect as described above.

The residual force is strong enough to overcome the electromagnetic repulsion of protons but has a relatively short range. That short range is the reason why atomic nuclei are not of arbitrary size. The electromagnetic force becomes relatively stronger with increasing atom size, preventing the nuclei from entering a more bound state. Therefore, the mass defect also decreases, starting at Fe-56.

Power Regulation, Conversion, and Distribution

Batteries can be charged if the power generated by the solar panels exceeds the needs of the spacecraft's systems. On the other hand, that power is not needed all the time and can in fact be harmful. Once the battery is fully charged additional power from the solar panel would actually damage it. Thus, a method to limit the charging of the battery and thus control the power generated by the solar panels must be employed. Often, a maximum power point tracker is used. This is essentially a controller with the dual purpose of firstly keeping the operational point at the maximum power point, as explained in Fig. 6.21, when the battery needs to be charged and secondly terminating the charge process of the battery by shifting the operational point towards the open circuit voltage when the battery is fully charged. This is done by increasing the resistance of the charge regulator and thus the combined regulator and bus load.

The solar array always operates where the generated power is sufficient to cover the then present bus load. When the bus load increases to values the solar array alone cannot provide the battery is used as well and its voltage decreases. This causes the maximum power point tracker to shift the voltage of the solar array towards that of the maximum power point in order to provide as much power as possible. This goes hand in hand with the usage of the battery. As soon as the maximum power point is reached the battery provides the rest of the needed power while the solar panel remains at its maximum power point. The battery is discharged. When the bus load decreases again power generated by the solar array is available and is used for charging the battery. Once the battery is fully charged the charge current is decreased by gradually shifting the operational point of the solar array closer to the open circuit voltage, thus generating less power and preventing overcharging the battery.

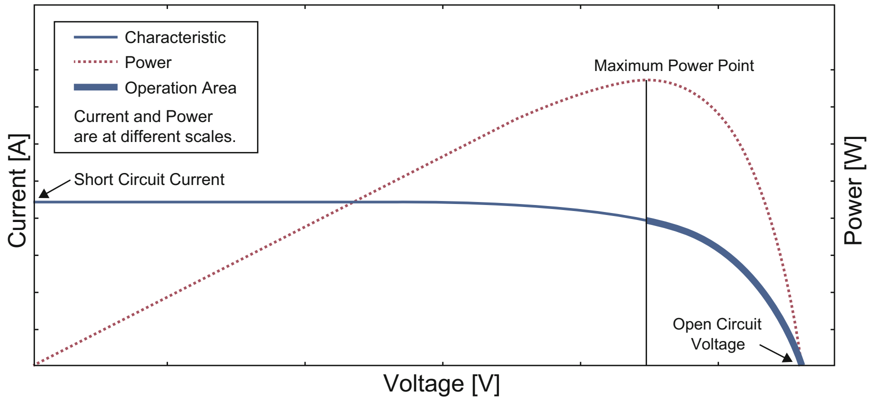

Fig. 6.21 Qualitative characteristic and power output of a solar cell. The *blue graph* shows how the voltage of the solar cell depends on an external load, i.e., a resistance. The open circuit voltage on the lower right is given by the type of semiconducting material that is used and is the highest voltage that can be generated by the cell. It would appear if the resistance is infinitely high, when a circuit is open. The other extreme is a situation where the current can flow without resistance. This current is limited and called the short circuit current. When an external load is increased the voltage drops more and more quickly. From the voltage and the corresponding current, the generated power can be derived via multiplication, which results in the *purple curve*. The generated power reaches a maximum before dropping again and is zero at the open circuit voltage and the short circuit current. The operation area is chosen to the right of the maximum power point

Another cheaper but less sophisticated method is to use shunts for every string of the solar array which can then be decoupled from the main bus individually. The unused power will then be turned into heat at shunt resistors which are usually located in the Power Control and Distribution Unit (PCDU).

Power needs not only be provided per se but provided at specific places in specific forms and at specific times. Spacecraft loads are usually the various on-board units such as electronics boxes, instruments, or reaction wheels. Many of these units might require a specific constant voltage to operate, therefore making a voltage conversion necessary. Consequently, most satellites have both an unregulated and a regulated voltage bus. The unregulated voltage bus usually consists of the connections between the battery and solar panels as well as the charge controller and other associated hardware. The unregulated voltage will therefore be whatever voltage is present at the battery and can vary significantly over time. For units that require a constant voltage, converters are necessary that convert the voltage to a specific and constant one. A voltage that is often used is 28 V, but lower ones are also in use such as 5 V for on-board computers for instance. Usually, the functionality of controlling and distributing power described above is combined in a box mostly called PCDU. The PCDU also provides means to protect against overcurrents. The input to that box is the unregulated power from the battery and solar panels. The output is both unregulated and regulated/converted power. The power is then routed to the units that require it. The PCDU is usually

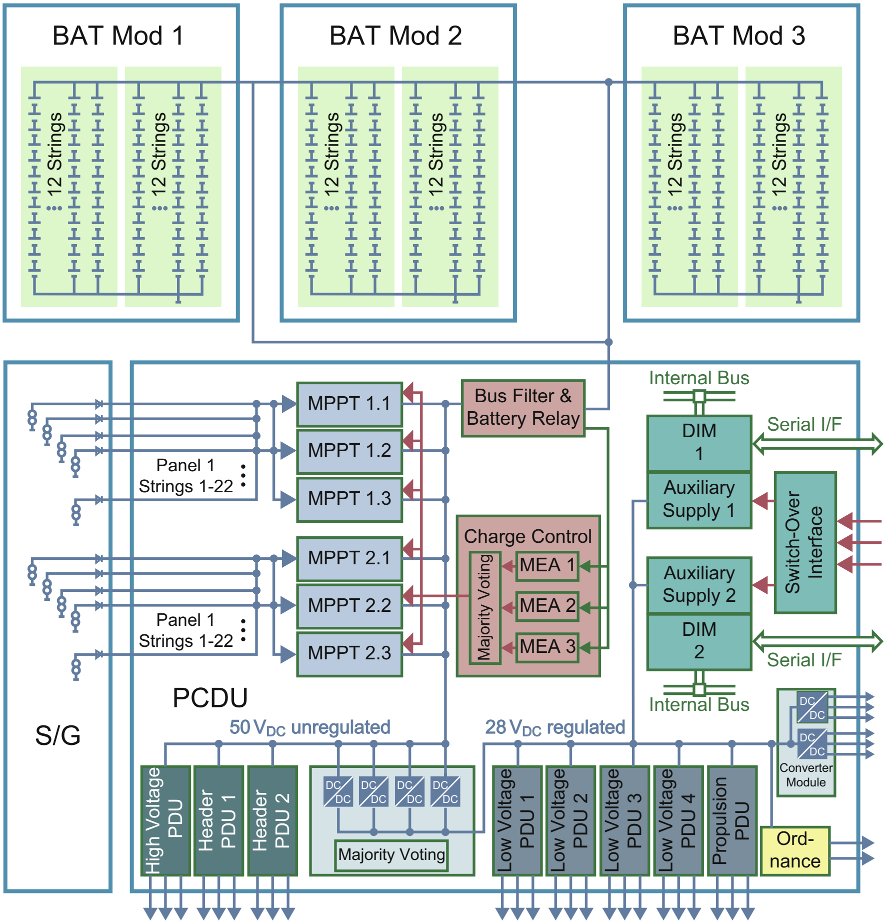

Fig. 6.22 Power Control and Distribution Unit (PCDU) schematics of the TerraSAR-X and TanDEM-X spacecraft. It is connected to the battery modules (BAT Mod) and the solar panel (here, S/G for solar generator). Inside the PCDU the connections of these two energy sources to the maximum power point tracker (MPPT) can be seen. From there the power is routed towards the unregulated 50 V bus and the regulated 28 V bus. These connections are themselves divided into various Power Distribution Units (PDUs). The connection to the on-board computer in the form of the Digital Interface Module (DIM) can also be seen

controlled by the OBC via an interface. In Fig. 6.22 schematics of the PCDU of the TerraSAR-X and TanDEM-X spacecraft can be seen. The spacecraft loads are comprised of the bus components such as star cameras, reaction wheels, computers, and payloads.

All the various spacecraft functions are physically located in individually placed boxes. This allows for easier design as each box can be created on its own and only

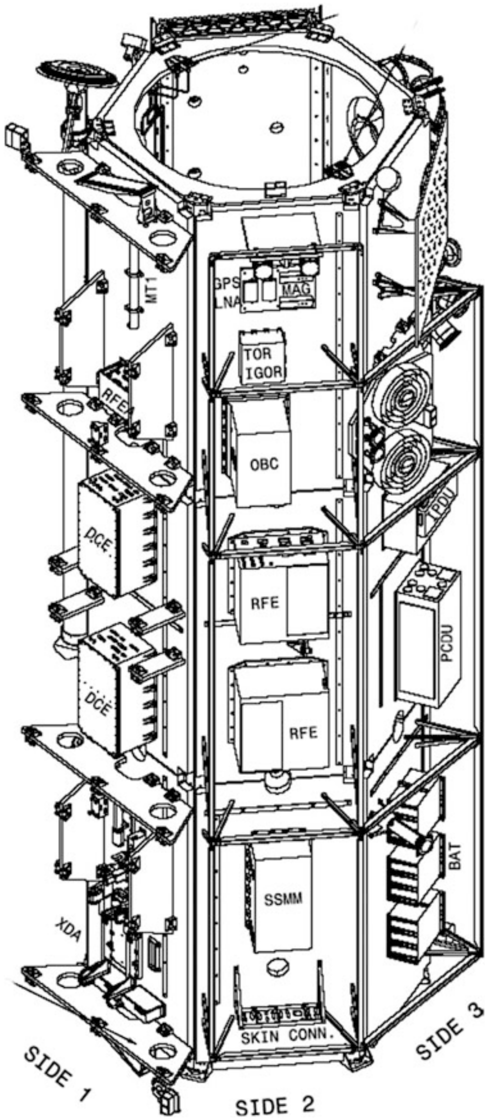

Fig. 6.23 Different functions are split into different units. In this example for TanDEM-X one can see for instance the three battery modules as well as the PCDU closely together on the lower right-hand side. Other units are the on-board computer (OBC), Radio Frequency Electronics (RFE), and the Solid-State Mass Memory (SSMM) in the middle. The solar panels are not visible. They are mounted at Side 1 across the whole length of the spacecraft

the interfaces have to be determined beforehand. What such design philosophy can look like can be seen in Fig. 6.23.

6.3.1.2 Thermal System

Another major topic in spacecraft design are the temperatures the spacecraft and its components have to function in. For many internal units these temperatures are

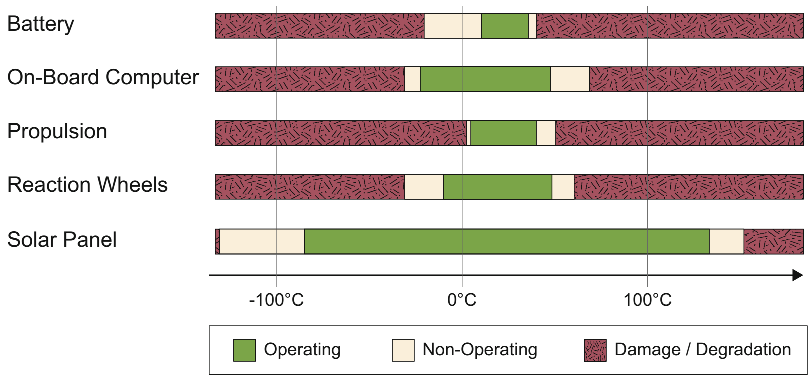

Fig. 6.24 Common operating and nonoperating temperature ranges. Nonoperating means that a unit is not expected to function at the low or high end of the range but. However, it is expected to function when the temperature is again within the operating range

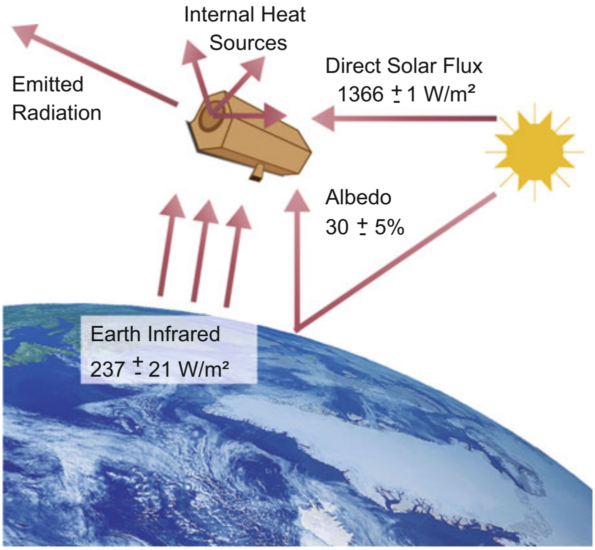

Fig. 6.25 Heat sources and radiation flux

desired to be at about "room temperature." Other units that are exposed directly to the space environment such as solar panels have to be able to withstand a wider temperature range, e.g., from −90 °C to +130 °C. Most attention needs to be drawn to the units with a very narrow operating temperature range. As can be seen in Fig. 6.24 the battery and the propulsion system are often drivers for the thermal design.

The overall temperature balance of a body is determined by the heat that it produces itself and the heat it receives from the outside versus the heat that is transported away from it (see Fig. 6.25). Since internal heat sources have to be

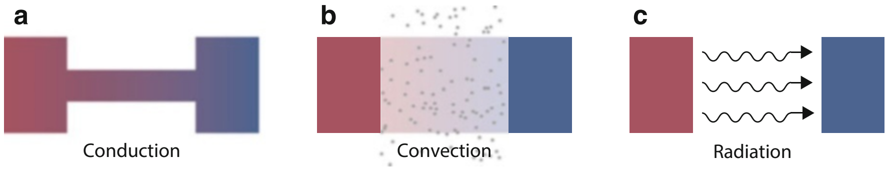

Fig. 6.26 Heat transfer mechanisms

taken into account the thermal system is closely intertwined with the electrical power system. The operation of units changes the temperature conditions and the temperature conditions determine the operability of units.

The additional heat generated on board must also be carried away. It becomes therefore evident that the transfer of heat plays a fundamental role in spacecraft design.

Very generally, there are three mechanisms of heat transfer: conduction, convection, and radiation. A temperature difference is always the driver with other parameters determining the effectiveness.

Conduction is the process of heat being transferred from one atom to the next within solid bodies but also within fluids and gases. This principle can be seen in Fig. 6.26a. In physical terms, the material-dependent property of how well heat is conducted is called *thermal conductivity*. It is evident that the thermal conductivity is much higher in solid bodies since there the average distance between atoms is much smaller than it is in gases, for instance. But of course, this property also varies widely between different solids.

Convection (see Fig. 6.26b) is another effective heat transfer mechanism that specifically makes use of the non-solidness of gases and fluids: they can move and thus carry away heat. There is free convection—where density differences caused by temperature differences result in movement in the presence of gravity and thus the transport of heat—and forced convection where the fluid moves due to an external cause, e.g., a pump or a fan.

The third mechanism is heat radiation (see Fig. 6.26c) where the heat transfer occurs by electromagnetic waves in the infrared range of the spectrum. Every physical body radiates heat. The radiated power of a body that exchanges heat with its environment exclusively via radiation is determined by the fourth power of its temperature T, its emissivity ε, its radiating area A, as well as the Boltzmann Constant σ.

$$P = \varepsilon \sigma A T^4. \tag{6.3}$$

When these three principles of heat transfer are applied to spacecraft design it becomes clear that convection cannot be used the same way it is on Earth where it is a prominent natural and technical mechanism for temperature regulation. A technical example is a car which partially uses air from the environment to cool the engine. In free space, there is no external medium present that could allow for convection to happen. Free convection cannot occur at all due to the "lack of gravity" which could induce air movement. Within a spacecraft only forced convection is possible. In the pressurized modules of the International Space Station (and even some satellites), for instance, the air has to be constantly kept in motion artificially by a ventilation system.

The absence of convection and the absence of matter around the spacecraft leave radiation as the only means to transport heat *away* from the spacecraft. With only one effective cooling mechanism it is clear that for most missions—despite the common knowledge of space being "cold"—the concern is not so much that a spacecraft becomes too cold but rather that is becomes too hot due to the lack of effective heat rejection capability.

Of course, the prevention of heat loss can be a concern as well. But it is obviously easier to heat up a unit by switching on an electrical heater than it is to cool something by "switching on heat transfer."

Therefore, for first estimations, the assumption is made that a spacecraft needs to be able to emit all the power it generates.[11] That means cooling needs to be "built in" at the very beginning of the design phase while heating could in theory be treated as an afterthought—provided enough electrical power is available.

Also, variations in the power consumption of a unit and the resulting changes in the heat balance are usually handled by adjusting the heating rather than the cooling function of the spacecraft. It is, however, possible to build spacecraft where the thermal regulation works completely passively and autonomously. This can be done for spacecraft which do not experience large variations in their power generation and consumption. Then the radiated power can be set accurately enough for all the units to remain within their operational temperature range without the need for additional heating.

Regulating the temperature is at first dealt with by insulating the spacecraft using multilayer insulation (MLI) foil. This serves two purposes: It prevents radiative heat loss into space and due to its reflective golden layer also excessive heat-up from the sun or the Earth's infrared radiation. In order to allow for heat to escape special radiative surfaces are employed. These should be placed strategically so that they face deep space and thus minimize heat absorption from external sources which would decrease the radiator's effectiveness. As can be seen in the equation for the radiated power [see Eq. (6.3)], the ability of a surface to emit heat is directly proportional to its emissivity. Consequently, radiating surfaces are usually treated with paint or coatings which have a high emissivity, for instance with white paint.

The emitted energy also is directly proportional to the fourth power of the temperature. Therefore, a higher radiator temperature would significantly increase its thermal energy rejection capability and thus allow a smaller area and

[11] Since the radiated heat needs to be equal to the generated heat large radiators can be seen where a lot of power is generated, i.e., on space stations or designs for nuclear spacecraft such as the canceled JIMO mission.

consequently a lower spacecraft mass. However, in a passively cooled system the radiator needs to be cooler than the unit it is supposed to cool in order to maintain the heat flow from warm to cold. For passive heat transfer the radiator temperature is thus rather low, often below 0 °C. A high radiator temperature is in principle possible, but this comes at the cost of having to employ active heat transfer which can transport heat from a warm unit to an even warmer radiator. Such a system functions similarly to a common refrigerator where heat is transported from the inside of the refrigerator to the warmer outside. Energy needs to be added to allow for the "heat pump" to work. Due to the complexity of such a system and the added mass of the active components it is usually not employed on satellites unless absolutely needed. Examples are often scientific spacecraft that have infrared telescopes as payloads which need to be cooled to very low temperatures.

Heat transport (and prevention thereof) within the spacecraft itself is primarily realized via the usage of materials with either low or high thermal conductivity. High conductivity is used wherever heat transport is explicitly desired. This is often the case when heat shall be transported from a unit to a radiator. Low conductivity is used for parts of the spacecraft that shall be thermally insulated. Solar panels are an example since they are by definition oriented towards the sun and thus rather warm. Thermally insulating them shall prevent too much heat from entering the spacecraft itself via the panel structure or attachment joints.

Another common heat transfer component is the heat pipe. A heat pipe is in principle a closed pipe that contains a working fluid which evaporates at the high temperature end; the gas fills the pipe and then condenses back into liquid form at the cool end. A wicking material causes the fluid to move back to the hot end where it can evaporate again. Heat pipes are common thermal connectors to radiators.

Thermal Control predominantly makes use of electrical foil heaters. These are in principle wires encapsulated in Kapton foil that are being heated up by resistance heating. The other part of thermal control is the need to measure temperature. Such a temperature measurement is mostly done by thermistors which are electrical resistors where the dependence of the resistance R on the temperature is well known, e.g.,

$$\frac{R}{1\,\Omega} = 1{,}000\left(1 + 3.9083 \times 10^{-3}\left(\frac{T}{1°\mathrm{C}}\right) - 5.775 \times 10^{-7}\left(\frac{T}{1°\mathrm{C}}\right)^2\right) \tag{6.4}$$

for a common type of thermistor called PT 1,000[12] (T is here the temperature in °C without unit).

Whether or not a heater needs to be switched is determined by the OBC which checks the temperature readings of the thermistors at certain intervals. These readings are then compared to on and off thresholds which can be changed from the ground. When the temperature is below the on-temperature the heater is being

[12] "1000" since the resistance at 0 °C is 1,000 Ω.

switched on and when it is above the off-temperature it is switched off again. Any temperature measurement (or any measurement in general) is originally analog and must be converted into a digital signal in order for the computer to be able to use it. This conversion usually leads to an uncertainty in the measurement. The bit range to represent a certain temperature range then determines the accuracy of the measurement. For instance, a 12-bit string can describe 4,095 values, which allows for a temperature range from -160 °C to +160 °C to be accurately measured to within 0.078 °C which is sufficient for temperatures. Thus, in reverse logic, the desired accuracy of the measurement determines the bit range in the design phase.

Operational Needs

The above is one example of a general principle: The design of a spacecraft must meet the operational needs. Therefore, thorough design and testing need to be employed to have accurate measurements and avoid interference among units and sensors. For instance, the magnetic field sensors on board a spacecraft whose purpose is to measure the Earth's magnetic field for attitude determination must not be perturbed by other magnetic fields generated on board. For this reason magnetic fields like those generated by high currents need to be avoided by using twisted wires with opposing currents which cancel each other's magnetic fields. Unavoidable disturbances must be known beforehand and included in the on-board software, preferably in the form of commandable settings so that they can be adjusted later in the mission. Another example are batteries that might need to be operated within tight limits—especially towards the end of missions. Here, accurate measurements of state of charge, voltage, currents, and temperature are absolutely critical to ensure proper handling from the ground. It must therefore be known by the spacecraft designer what to include in the telemetry and how and where that telemetry is measured in order to allow for safe and reliable operations.

6.3.2 Operations

6.3.2.1 Preparation Phase

Satellite power and thermal system operations start several years before a spacecraft is launched. As soon as the satellite design is determined one has to consider the consequences and necessary actions for operations. Depending on the satellite design (see previous chapter) and the mission requirements on the power and thermal system, an operations concept has to be developed.

A so-called power thermal model is generated for most satellite missions. This model e.g. allows estimations how long operations, maneuvers, or data takes can last without discharging (and/or heating up) the battery too much. The thermal model part contains on the one hand all critical temperatures (e.g., instrument

Fig. 6.27 Deployable and undeployable solar arrays. For TerraSAR-X (*left*) the solar panels are mounted on the upper right of the satellite. Since its follower TerraSAR-X 2 (*right*) will have higher power consumption, larger solar arrays are needed. Thus the panels have to be deployed after launch

temperatures during operation) and on the other hand the thermal behavior during warm-up and cool-down. Depending on actual heat dissipation, heat capacity, and operation phase, the warm-up of the units is largely linear with time whereas the cool-down in general shows an exponential temperature drop. This is caused by the reason that during heating up the conducted power (generated by a heater) is constant whereas during cool down the dissipation of heat depends on the temperature of the thermal environment within the satellite. The battery charge control model is driven by the total power over time. It includes much information about the charge behavior of the battery, the number of allowed charge cycles, and the maximum Depth of Discharge (DoD).[13]

6.3.2.2 LEOP Operations

After the launch it is necessary to bring the satellite into a nominal operating state as soon as possible in order to lower the risk of power problems. For many space missions, power is generated by the combination of solar panels and batteries (see Sect 6.3.1). Several units such as the OBC, sensors, or heaters are switched on shortly after separation. The power therefore is drained from the battery as long as the solar arrays do not yet generate power.

For this reason one of the first actions after separation is the deployment of the solar panels (if they are designed to be deployable; see Fig. 6.27). Right before that, the power and thermal subsystem engineer needs to evaluate the situation. Since solar panels are deployed by activating pyro cutter mechanisms which require a certain operating temperature, this and other preconditions need to be checked. After deployment is finished, the operations engineer has to confirm that the solar

[13] The index of utilization of the battery is the Depth of Discharge (DoD). This is the amount of charge drained from the battery expressed as a percentage of its rated capacity.

arrays are fully deployed and working properly. During sun illumination they have to generate enough power to operate all of the satellite equipment and to recharge the battery for eclipses or times with higher power consumption.

After this is done and the spacecraft generates enough power for operations, several units need to be activated. As soon as all activation preconditions are met, the nominal and redundant units are powered on by ground command to test their functionality. Also several on-board data configurations need to be checked and adapted: Since the real behavior of the vehicle in space cannot fully be predicted in the design phase, the trigger limits for autonomous on-board reactions have possibly to be adapted after launch.

Launch delays of a spacecraft by several months or even years have negative effects especially on the spacecraft battery. Even if the battery is not used before launch it degrades over the time and therefore the usable power budget decreases. The rate of degradation depends on the battery type. Li-Ion batteries, for example, degrade only slowly. Spacecraft batteries are already charged on ground regularly to compensate the self-discharge, which could harm or even destroy the battery cells by deep discharging. Nevertheless a launch delay of several months or more leads to a battery with a lower capacity as desired.

If the battery already degraded before launch it is desired to find possibilities to increase battery health after launch. This is called reconditioning. It depends on the battery and satellite design, whether it is possible to adapt the charge level in order to avoid temperatures or voltages which are outside design specifications or not.

The reconditioning after launch sometimes causes additional work for the power and thermal system team. Charging the battery slowly and creating a small overcharge prevents e.g. NiH_2 battery cells from degradation.

The task of the operations engineer is to derive a set of parameters which can be commanded to the satellite in order to optimize the battery charge process. Of course it can be necessary to adapt the parameters several times over the whole mission.

6.3.2.3 Routine Phase

Once the satellite is in an operational state the telemetry is continuously being monitored and the health status of the spacecraft equipment and the behavior is evaluated over large time spans. Trends are analyzed to achieve better predictions for the future subsystem behavior. For that it is indispensable to differentiate which effects are caused by e.g. seasonal variations or operations and which are induced by aging or degradation. An example for a suspicious behavior, which is actually explainable by normal effects, is given in Fig. 6.28.

In case telemetry readings or trends cannot be explained, it has to be clarified whether the observed behavior is critical and whether any action is required.

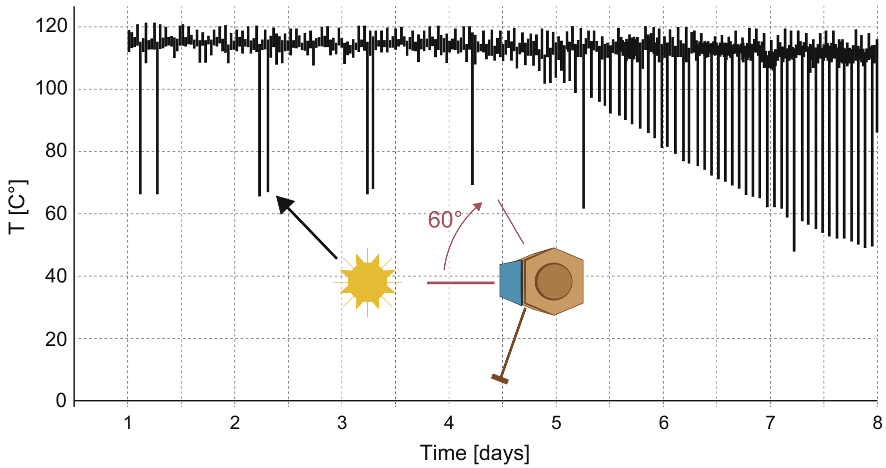

Fig. 6.28 Solar array temperature of TerraSAR-X over some days. Some irregular dips down to 65 °C can be seen as well as periodical dip intervals of 90 min with a constantly decreasing envelope. Contrary to first assumptions this is no indication for any problem or degradation, but just shows some operational and seasonal effects on the solar panels: The irregular dips from 115 °C to 65 °C are caused by so-called left-looking modes. The satellite is rotated by 67° around its *x*-axis and thus the solar arrays are no longer pointed directly towards the sun which results in a strong temperature decrease. The periodical temperature drops are a consequence of solar eclipses. Whenever the sun is shadowed by the Earth the temperature of the solar arrays decreases dependent on the duration of the eclipse

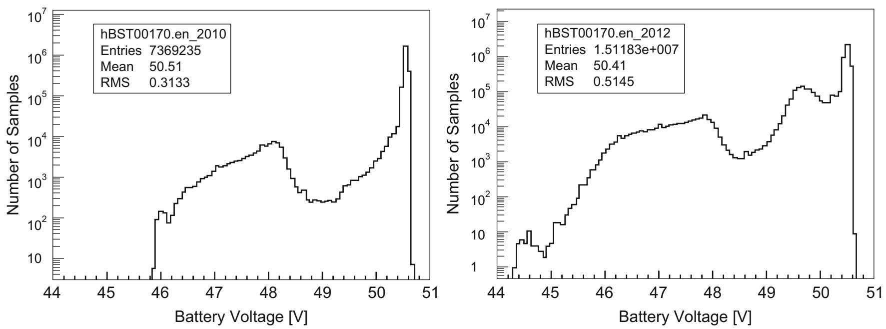

Fig. 6.29 These two histograms show the battery voltage distribution over 1 year for 2 subsequent years. In that time period a shift towards lower voltages can be observed; the minimum battery voltage decreased from 45.8 V to 44.2 V. The cause for this behavior can be attributed to the degradation of the battery

Battery Discharge Process

Probably the most sensitive and stressed unit within the power system is the battery. An example for its degradation is given in Fig. 6.29.

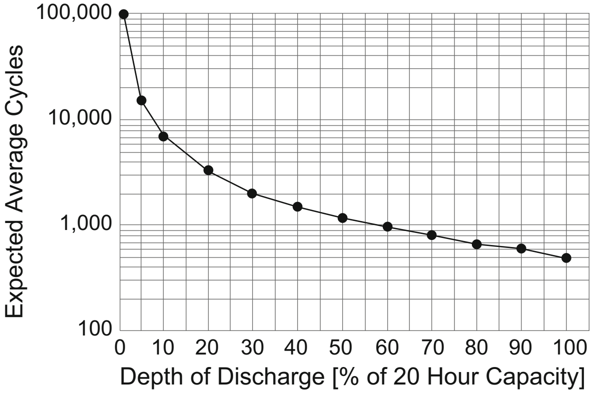

Fig. 6.30 The amount of expected charge cycles decreases with increasing Depth of Discharge (DoD). To achieve a certain amount of cycles it is therefore necessary to limit the DoD

Depending on the orbit and operational concept a battery sometimes has to go through thousands of charge cycles during its lifetime. Some experiments and payloads even need more power than the solar arrays generate. During operations they drain power which is stored in the battery and the battery is recharged once the operations have finished. For some missions the solar arrays generate power all the time, but payload operations are only performed over a certain amount of time (e.g., TerraSAR-X). For this type of mission the payload is allowed to drain more power as generated by the panels as the battery can be charged during periods without payload operations. For missions where payload operations are performed all the time [e.g., Gravity Recovery And Climate Experiment (GRACE)] it is necessary that the solar arrays produce more power than is used by the spacecraft.

The battery capacity is in fact the limiting resource for many missions, even more limited than propellant mass or available cycles of operations for thrusters or BUS[14] units (see Sect. 6.3.1). Here, the important characteristic is the number of charge/discharge cycles at a given DoD. The cycle lifetime increases as the DoD decreases (Nelson 1999). If the desired lifetime for a spacecraft is 5 years in LEO with 25,000 expected charge cycles, for example, it has to be ensured that the DoD is always below 15 %, assuming a behavior as seen in Fig. 6.30. This usually also is part of the power thermal model. Discharge can be caused by several things such as data takes, maneuvers, or eclipses. If analyses show that the DoD is already close to its limit for usual payload operations, the power and thermal engineer has to take care that no additional power-consuming operations (e.g., data takes) are done during an eclipse or a maneuver.

[14] A satellite bus or spacecraft bus is the general model on which multiple-production satellite spacecraft are often based. The bus is the infrastructure of a spacecraft, usually providing locations for the payload (typically space experiments or instruments).

Battery Charge Process

Also the charge process needs to be well defined. Charging the battery slowly is the preferred operations concept, but need to be balanced with other requirements, like having a fully charged battery available at a certain time.

A slower charging of the battery can be achieved by rotating the satellite's solar arrays away from the sun as the charge current depends on the effective illuminated area of the solar cells. A rotation of 30° decreases that area and therefore the charge current by approximately 14 %.[15] The smaller the charge current, the lesser heat will be produced during overcharging the battery. This makes it possible to charge the battery to a higher level of capacity without accepting an additional degradation.

Another possibility to influence the charge process is the so-called shunting of the solar array strings, which is essentially a partial deactivation of solar cells leading to a reduced current. As soon as a predefined charge level is reached shunting would be activated by the on-board software. When the state of charge drops again below the set DoD level shunting would be switched off and the charge process starts again.

The charge process can also be adapted to a decreased capacity by lowering the maximum charge level if it is adjustable per design[16]: Degradation of the battery can cause the effect that it is no longer possible to charge it to its Begin-of-Life (BoL) capacity—setting the maximum charge level to 100 % BoL capacity would consequently lead to an overcharge which would be converted into heat. Therefore the temperature within the battery would increase and possibly damage it. To avoid this, the power and thermal operations engineer has to adapt the maximum charge level to a reasonable level.

All of the scenarios mentioned above decrease the charge current which leads to the generation of less heat. Therefore, battery health can be preserved or even increased.

Battery Temperature and Performance

As already mentioned above an important factor for battery performance is temperature. The hotter the battery, the faster chemical reactions will occur. The desired chemical reactions on which the battery depends are usually accompanied by unwanted chemical reactions which consume some of the active chemicals or impede their reactions. Even if the cell's active chemicals remain unaffected over

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[15] Since $\cos(30°) = 0.86$.

[16] Example: BOL Capacity = 20 Ah, degradation = 5 %, so the battery can only be charged up to 95 % = 19 Ah. So, one could set the maximum capacity level to 95 % to avoid overcharge and therefore thermal effects. It should be noted that for Nickel–Hydrogen batteries, a slight overcharge is actually desired.

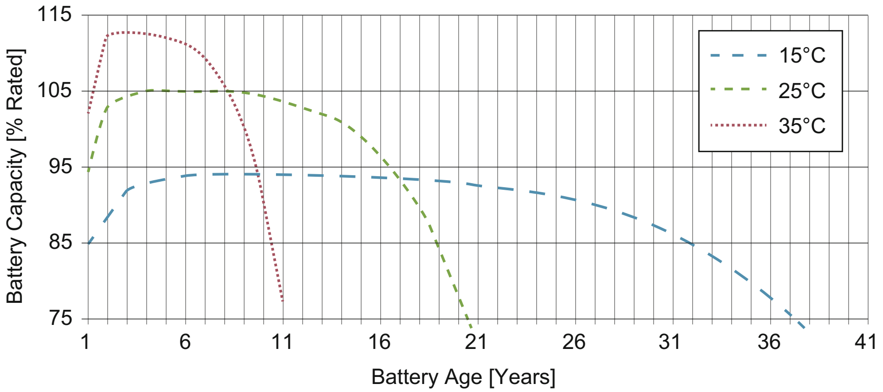

Fig. 6.31 The image shows the battery age dependency on the temperature. It can be seen that high temperatures decrease the battery lifetime significantly. Low temperatures decrease the maximum capacity of the battery

time, cells can fail because unwanted chemical or physical changes to the seals keeping the electrolyte in place. As a rule of thumb, for every 10 °C increase in temperature the reaction rate doubles. Thus, an hour at 35 °C for lead-acid batteries is equivalent in battery life to 2 h at 25 °C, as shown in Fig. 6.31. Heat is the enemy of the battery and as Arrhenius (Laidler 1987) shows, even small increases in temperature have a major influence on battery performance affecting both the desired and undesired chemical reactions.

Battery temperature has therefore to be monitored very carefully during the whole mission. It is necessary to reach a compromise between the temperature being low enough to afford a long life time and being warm enough to provide good performance. It is the duty of the power and thermal engineer in cooperation with battery experts to determine the time whenever it is necessary to regulate the battery charging and to perform the required actions.

Thermal Subsystem

Besides all the necessary operations concerning the power system there also exist a set of actions that have to be performed for the thermal system. Almost every unit of a satellite has specified limits between which it has to be operated. Among the temperature-sensitive units are star trackers, flow control valves for thrusters, and the propulsion system. Their temperatures have to be observed over the whole mission and specific actions have to be performed if necessary. If a unit impends to cool down below its operational limit it is necessary to switch on its heater if available. If the unit does not have heaters sometimes it is even necessary to create new operational concepts. This can, for example, be to rotate the satellite in a way that the units are pointed towards the Earth or the sun (if they are too cold) or to rotate them away (if they are too hot).

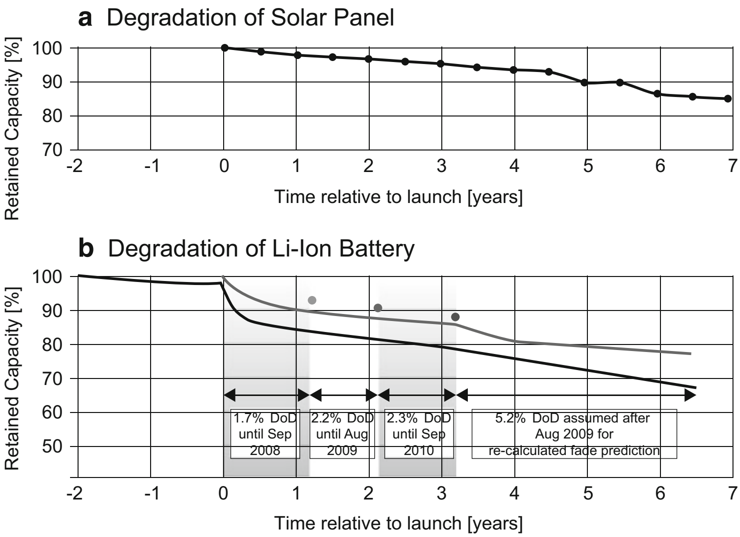

Fig. 6.32 The *upper image* (**a**) shows a typical degradation curve of solar panels. The *lower one* (**b**) describes the degradation of a Li-Ion battery. The *dark* and *light lines* show two predictions according to different models. The *dots* mark the real degradation

It is obvious that all those analyses and actions would be a huge amount of work if it all has to be done manually. To make operational life easier, automatic checks and corresponding actions can be implemented in the spacecraft. Several parameters such as battery voltage, temperatures, and power statuses are monitored by the on-board software. As soon as one of these parameters is out of limit for a certain amount of time, a specific action is performed automatically (e.g., switch on heater if temperature is too low).

The degradation of units has to be analyzed periodically by the operations team and the responsible experts. The observed behavior has to be compared with the modeled one and resulting actions have to be derived from this.

The two images in Fig. 6.32 show the degradation of solar panels and a satellite battery. The causes for the reduced performance (see Sect. 6.3.1) can be aging, radiation, or charge/discharge cycles. These effects have of course an impact on satellite operations. When solar arrays and battery degrade this means that all in all less power is available. This has to be taken into account for all activities, monitorings, and limits. Therefore some values and parameters (e.g., heater set points, battery limits, etc.) have to be adapted periodically during the mission and actions to recover the battery cells (described in Sect. 6.3.2.2) can be performed. Furthermore it has to be analyzed if payload operations have to be changed. This can result, for example, in decreased observation duration to avoid deep discharge and preserve the battery.

6.3.2.4 End of Life

Most of the satellite missions are extended beyond its originally planned lifetime. Once the degradation of the power equipment reaches a (mission) specific point, the satellite operations cannot be continued as before, especially during eclipse season when the satellite is mainly dependent on battery power. Also the solar panels degrade, what lengthens the battery recharge. With advancing age of batteries and solar panels, the daily operations become time-consuming and are ruled by avoiding low-power conditions.

The GRACE (Tapley et al. 2004) project shows very explicitly how end of life (EOL) operations use to be. During eclipse phases, which are experienced twice per year[17] no power can be generated by the solar panels and thus the energy needed is drained from the battery. The eclipse duration varies from some seconds at the beginning to more than half an hour in the middle of the eclipse phase. By now (11 years after launch) the battery capacity is below 10 % of its original value. As a consequence the batteries cannot provide enough power to all units during long eclipses.

There are some operational tricks to deal with already weakened batteries. If it is possible to adjust the end-of-charge voltage of the battery, this charge level can be raised at the end of an eclipse to allow an immediate maximum charging of the battery after leaving the Earth shadow. After a certain time in sun condition, the charge level has to be reduced again to avoid an overcharge of the battery. If the battery power is no more sufficient to maintain the nominal satellite operation during an eclipse, it can be helpful to heat up parts of the satellite such as the propellant tanks during sun condition. The heat dissipation can then be used afterwards to keep the single units of the satellite in its temperature limits. Additionally, heater power can be saved by reducing the heater set points to a minimum during eclipse. (For some units this cannot be done for the whole orbit duration because payload data quality is temperature dependent).

If none of these measures is successful, the only remaining option is to switch off single units or at worst the whole satellite. This implies a major impact on the mission and involves a lot of subsequent adjustments of units like the OBC. Potentially even software uploads are then necessary to restore the required configuration. This worst-case scenario should be avoided under all circumstances.

Full sun orbits with continuous power generation can also be a threat to weak battery cells because of missing discharge processes. This would then result in an irreversible cell collapse due to electrolyte bridging. Again it is the duty of the operations team to avoid these. For the already mentioned GRACE satellite it is unfortunately neither possible just to disable some of the solar array strings as described above nor can additional consumers be switched on. One possibility to force the battery's discharge is to generate some kind of artificial eclipse. This can be reached by rotating the satellite in a way that the solar panels are not irradiated

[17] For low-earth orbit satellites with polar orbit.

by the Sun for some minutes. First of all it has to be clarified how often such "eclipses" have to be generated and how long they should be. Therefore a close coordination with the satellite support team and especially with the battery experts is required in order to find the best solution. Also the AOCS engineers need to be involved: While the AOCS team calculates what kind of rotation should be made to save as much fuel as possible (at EOL most likely fuel budget is very low, too), it is very important to figure out if such a rotation poses a threat to the thermal system with respect to allowed unit temperatures. Usually it is required to point the solar panels towards the Sun whenever it is possible. Therefore the spacecraft is designed such that all units which should be operated at low temperatures (e.g., star cameras) are mounted on the opposite side of the vehicle. When rotating the spacecraft to a not nominal attitude it could be that some units get too hot, others which are usually pointed towards the sun could cool down. So a compromise between thermal and power-driven attitude requirements has to be found.

These actions have to be planned, prepared, and executed carefully by the operations team. Monitoring limits for temperatures or voltages during the time without payload operations have to be adapted because charge and temperature behavior change (e.g., the temperature decreases if less payloads are operated). The switching has to be timed exactly to avoid voltage or temperature variations that would violate their limits and as a consequence trigger any unwanted on board mechanisms.

At the end of the satellite's lifetime operations get harder and it is necessary to know the system very well to extend the mission as far as possible because the components within the power system are the limiting ones for most missions.

6.3.3 Contingency Operations

The power and thermal subsystem occupies a central position for the operability of a spacecraft because all systems need power and thermal control. When a PTS contingency occurs, it is not unusual that the whole spacecraft is affected. In a Multiple-Failure Scenario, the PTS subsystem engineers have to identify and to recover electrical failures first. In the case with several electrical failures, they have to begin with the failure most upstream. An example would be the failure of an on-board instrument and simultaneously voltage fluctuations of the relating power bus. Stabilizing the power bus would be in this case the precondition for a successful recovery of the failed instrument. Thermal failures allow in most cases to be solved subsequently because of the thermal inertia of the systems. Some spacecraft have dedicated modes for a failure of a cooling circuit or for a reduced heater activity in case of a low-power situation.

In some anomaly cases it is not directly obvious what caused the problem. If for example the output power of the solar panel is reduced there are a few different failures which can lead to a reduced output power displayed in the telemetry. As illustrated in Fig. 6.33, damage of a part of the solar panel (#1), cable break (#2),

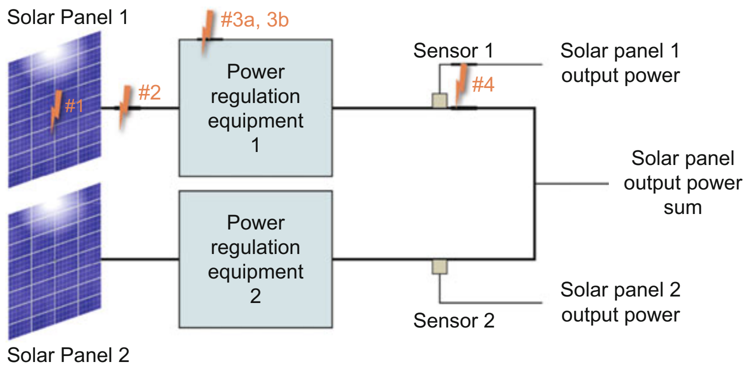

Fig. 6.33 Possible reasons causing a reduced solar panel output power

switch-off (#3a), or damage (#3b) of power regulation equipment or damage of a sensor (#4) could be the possible reasons.

As diverse the possible causes are as diverse are the effects on the mission. The loss of output power of the solar panel could be real (#1, 2, 3a, 3b) or not (#4) and possible to recover (#3a) or not (#1, 2, 3b).

In 2004, after about 3.5 years of the GRACE mission, first a DSHL (Disconnection of Supplementary Heater Lines) and afterwards a DNEL (Disconnection of Non-Essential Loads) occurred on one of the two satellites. For this kind of spacecraft there exist four stages of DNEL. A DNEL is caused by an undervoltage situation in the battery. The deeper the voltage drops the higher is the stage of DNEL and the more actions are performed automatically on board. In this case the spacecraft was found in thermal survival mode during ground station contact. In this mode all unnecessary heaters are switched off and the heater set points are lowered to a point where as little power as possible is needed to keep the satellite and its hardware alive. After analyzing the data it could be seen that a battery cell was lost and the battery voltage dropped below a defined threshold for more than 30 s, which triggered the first stage of DNEL. In addition to the change of the heater configuration several other on-board reactions are triggered by a DNEL. As the spacecraft is commanded to a coarse pointing mode which leads to a higher gas consumption, several hardware units are switched off which are not necessary for the safety of the satellite. The transmitter is switched to low rate to save power during ground station contact and all commands on board, which were saved in the timeline, are deleted to avoid power-consuming activities.

As one can see, there is a huge amount of actions necessary to get the satellite back operational. First of all, the EOC level settings (described in Sect. 6.3.1) have to be changed in order to compensate for the low-power situation. After this event in 2004, the EOC level was raised to get the battery stabilized. Additionally the spacecraft was commanded back to thermal nominal mode to reestablish the desired heater configuration. As the satellite transmitter was switched to low rate by an on-board action, a blind acquisition had to be made. This is because the spacecraft was configured to low rate while the ground station was configured to high rate. The reconfiguration on board was in this case faster than one on the ground. This means that the subsystem engineer had to prepare a procedure to switch the transmitter back to high rate to see telemetry. As the complete timeline on board the spacecraft was empty, he also had to ensure that the transmitter will be switched off again at the end of the ground station contact to save power. The timeline had to be updated and reloaded. The payload equipment had to be switched on and set up again as soon as enough power was available. The telemetry had to be analyzed to find out which autonomous on-board actions were executed because an on-board action is disabled directly after triggering to avoid a continuous execution. All executed actions had to be re-enabled after the recovery to ensure satellite safety in case of a similar problem in the future. For example, the voltage drop could occur every eclipse because the battery does not gain enough power. If one does not re-enable all necessary on-board reaction after setting the heater configuration or switching on some units, no on-board actions will be triggered during the next low voltage situation. This could lead to a voltage drop below a limit acceptable for the OBC unit. In the case of the GRACE mission the switch-off of the OBC would be equal to the loss of the mission because neither an autonomous reboot of the OBC unit is foreseen nor the execution of any on-board telecommands.

6.4 Propulsion Subsystem Operations

Franck Chatel

6.4.1 Principle of Propulsion

Satellite or rocket propulsion systems are based on the action–reaction principle: thrust arises in reaction to the expulsion of mass. The same principle is at work when a cannon is fired and moves backwards after the shot.

Some simple physics allows quantifying the force created by the expulsion of mass. Consider a satellite of mass m moving through space with a velocity vector $\boldsymbol{V}$, as shown on Fig. 6.34. The propellant mass dm to be expelled is bound to the satellite and moves with the same velocity vector. An instant $\mathrm{d}t$ later, the mass is expelled with an exhaust velocity $\boldsymbol{V}_\mathrm{e}$ relatively to the satellite body, which receives a velocity increment $\mathrm{d}\boldsymbol{V}$. Newton's second law of motion relates the change in impulse of the considered system to the sum of the external forces $\boldsymbol{F}$ acting on this system.

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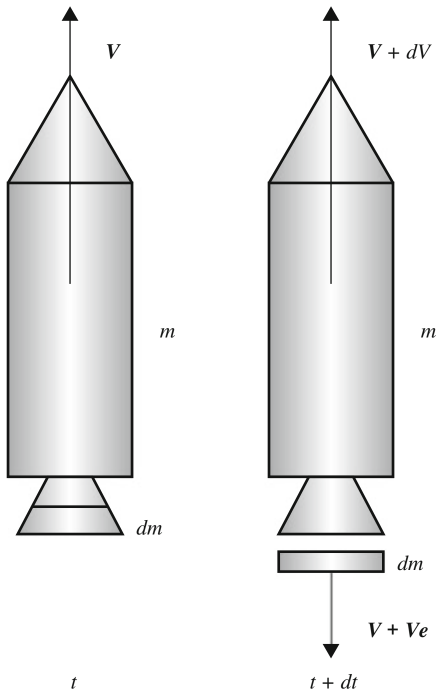

Fig. 6.34 Kinematical states of the spacecraft prior and after the expulsion of mass

In the present case, the system composed of the satellite is open since its mass varies due to the expulsion of propellant mass. Newton's second law, however, applies to closed systems so that one has to consider the system composed by both the satellite and the expelled propellant mass. Before the expulsion of the propellant mass, the impulse of the system is $\boldsymbol{p}_1 = (m + \mathrm{d}m)\boldsymbol{V}$ whereas after the expulsion, it becomes $\boldsymbol{p}_2 = m(\boldsymbol{V} + \mathrm{d}\boldsymbol{V}) + \mathrm{d}m(\boldsymbol{V} + \boldsymbol{V}_\mathrm{e})$. Newton's second law then yields:

$$\boldsymbol{F} = \frac{\mathrm{d}\boldsymbol{p}}{\mathrm{d}t} = \frac{\boldsymbol{p}_2 - \boldsymbol{p}_1}{\mathrm{d}t} = m\frac{\mathrm{d}\boldsymbol{V}}{\mathrm{d}t} + \frac{\mathrm{d}m}{\mathrm{d}t}\boldsymbol{V}_\mathrm{e}.$$

The change in impulse is different from the usual product of the mass by the acceleration vector: an additional term appears because of the mass variation. This relation can be reordered as follows:

$$m\frac{\mathrm{d}\boldsymbol{V}}{\mathrm{d}t} = \boldsymbol{F} + \boldsymbol{T}$$

with

$$\boldsymbol{T} = -\frac{\mathrm{d}m}{\mathrm{d}t}\boldsymbol{V}_{\mathrm{e}}. \tag{6.5}$$

This reordering shows that the situation can be seen as equivalent to a usual closed system (term on the left-hand side), except that a new force $\boldsymbol{T}$, the thrust, must be introduced. Of course, one must keep in mind that the satellite mass m is variable. However, it can often be considered as quasi-constant when the expelled mass is small with respect to the satellite mass.

The expression obtained previously shows different features of the thrust:

- Its direction is opposed to the one of the exhaust velocity, which reflects that this force is a reaction
- It is proportional to the exhaust mass flow rate $\mathrm{d}m/\mathrm{d}t$
- It is proportional to the exhaust velocity

The norm of the exhaust velocity vector is often written as the product of the standard gravity g_0[18] by the specific impulse (noted I_{sp} and expressed in seconds). One can then write

$$\boldsymbol{V}_{\mathrm{e}} = g_0 I_{\mathrm{sp}} \boldsymbol{u}_{\mathrm{e}} \tag{6.6}$$

where $\boldsymbol{u}_{\mathrm{e}}$ is a unitary vector aligned with the direction of the exhaust velocity. Consequently, the expression of the thrust becomes:

$$\boldsymbol{T} = -\left(g_0 I_{\mathrm{sp}} \frac{\mathrm{d}m}{\mathrm{d}t}\right)\boldsymbol{u}_{\mathrm{e}}. \tag{6.7}$$

From this expression, one can say that the higher the specific impulse, the less the mass flow rate required to produce the same thrust level. This expression is quite general and applies to all kind of engines, even if part of the propellant is not stored on board as it is the case for airplanes.

Specific impulse is used to distinguish classes between different propulsion systems. Chemical systems use a chemical reaction to produce gases, which are subsequently expelled through a nozzle. For such systems, the mass flow rate is rather high because the expelled gases are quite heavy. However, the exhaust velocity is rather low, leading to specific impulses on the order of 250–500 s. On the other hand, an electric system can use high-power currents to produce ions that are then accelerated by a magnetic field before being expelled. In this case, the mass flow rate is rather low, but the exhaust velocity is very high, leading to specific impulses in the order of 1,500–3,000 s.

Finally the equation of motion can be integrated under some assumptions into the so-called ideal rocket equation.[19] If one assumes that all external forces are

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[18] g_0=9.80665 m/s^2.

[19] Also named after Konstantin Tsiolkovsky, a Russian scientist who derived this relation at the end of the nineteenth century.

negligible with respect to the thrust and that the exhaust velocity vector is constant over the maneuver duration then one can write:

$$m\frac{\mathrm{d}\boldsymbol{V}}{\mathrm{d}t} = -\frac{\mathrm{d}m}{\mathrm{d}t}\boldsymbol{V}_{\mathrm{e}} = -\left(g_0 I_{\mathrm{sp}}\frac{\mathrm{d}m}{\mathrm{d}t}\right)\boldsymbol{u}_{\mathrm{e}}. \tag{6.8}$$

Denoting the initial and final conditions with subscripts "i" and "f" respectively, one gets

$$\int_{V_{\mathrm{i}}}^{V_{\mathrm{f}}} \mathrm{d}V = \left(g_0 I_{\mathrm{sp}} \int_{m_{\mathrm{i}}}^{m_{\mathrm{f}}} \frac{\mathrm{d}m}{m}\right)\boldsymbol{u}_{\mathrm{e}}.$$

The terms are rearranged to give the usual form of the ideal rocket equation:

$$\boldsymbol{V}_{\mathrm{f}} - \boldsymbol{V}_{\mathrm{i}} = \Delta V \boldsymbol{u}_{\mathrm{e}}$$

with

$$\Delta V = g_0 I_{\mathrm{sp}} \ln\left(\frac{m_{\mathrm{i}}}{m_{\mathrm{f}}}\right). \tag{6.9}$$

6.4.2 *Configurations of Propulsion System*

Having understood the physics behind propulsion, we will use the example of a bi-propellant propulsion system to introduce a typical propulsion system layout and its operational configurations.

Bi-propellant systems are not the only propulsion system used for spacecraft. A mono-propellant system is very similar, except that only one propellant is used for the chemical reaction. This is for example the case when using hydrazine with a catalyst to activate the decomposition into gaseous products. Propulsion systems based on cold gas or electrical equipment are also in use. Their operation has some specialties, especially for electrical propulsion. The reader may refer to (Fortescue 2011) for a more in depth presentation.

6.4.2.1 Layout of a Bi-Propellant Propulsion System

In this kind of propulsion system, the thrust is obtained through the expulsion of the gas produced by a chemical reaction. In order to get a high exhaust velocity, the gas is expanded through a nozzle after the combustion chamber. Propellant liquids are stored in two separated propellant tanks and valves allow or block the flow towards the combustion chamber of the different engines so that the reaction can take place when commanded. Usually different engines are available, depending on the usage: apogee engine delivering around 400 N thrust for orbit maneuvers and reaction thrusters (between 1 and 10 N thrust) for attitude control or station keeping maneuvers.

Unfortunately a system built only on the previous elements could not work because the fluids are subjected to micro-gravity conditions in a spacecraft. A system is needed to press the fluids into the outlet of the tank. A tank filled with inert gas (typically helium) under high pressure coupled with a pressure regulator is connected to the propellant tanks. The inert gas can either be directly in contact with the propellant or separation is achieved via a bladder linked to the tank wall. The pressure regulator decrease the pressure of the inert gas to a pressure compatible with the propellant tanks characteristics (e.g., burst pressure, operational pressure) and allows controlling the fluid mass flow rate and ensures reproducible thrust conditions as long as it is in use.

All these elements are of course linked together through tubing so that fluids can flow between them. Apart from the already mentioned valve to control the flow to the combustion chambers, other valves are needed as well. Some valves are implemented to isolate parts of the system from the rest. Such valves can only be activated once (either to close or to open) and are composed of pyrotechnic devices. Release valves, that open when the pressure exceed a threshold, are used as safety to avoid high pressures in the system. Finally some valves are required to fill or drain the tanks and the tubing at the launch pad.

The observability of the system is ensured with pressure and temperature sensors together with the related electronics. The pressure regulator and the valves are commanded via the OBC. The resulting system is shown on Fig. 6.35.

6.4.2.2 Operational Configurations

After having shown an example of propulsion system layout, the first question that arises is how to operate it. Of course, operations depend on the layout itself and this chapter uses again the layout of a bi-propellant system as example.

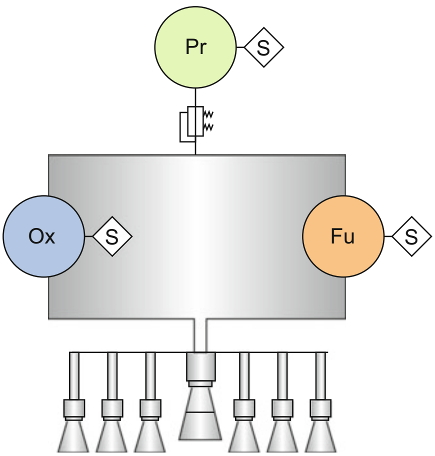

Fig. 6.35 Bi-propellant propulsion system

Before a spacecraft can be operated, it has to be launched. The environment encountered by the spacecraft during launch differs drastically (e.g., accelerations, vibrations, or temperature conditions) from the one experienced during the mission. The propulsion system is configured specifically for the launch in order to avoid degradations caused by this harsh environment: closed valves first isolate the inert gas tanks under high pressure (from 250 to 300 bar) from the propellant tanks pressurized around 18 to 20 bar. Propellant tanks are also isolated from the apogee engine and reaction thrusters by further closed valves in order to avoid an unwanted firing, which would be especially critical on top of a rocket! An inert gas under low pressure (around 3 bar) usually fills up the tubing between the previous valves and the engines to keep them under pressure and minimize the risk of degradation by the vibrations. The electronics controlling the propulsion system is usually switched off during the ascent as another security measure.

After separation from the launcher, the propulsion system shall be brought into an operational state (so-called priming) before the first orbit maneuver. The logic is to put the system under increasing pressure, beginning from the engines. The filler gas between the propellant tanks and the engine is vented by opening some thrusters. Then, after closing the thruster valves again, the valve isolating the propellant from the engine is opened. Finally the valve between the high pressure inert gas tanks and the propellant tanks is opened. The complete pressurization of the system can take up to 2 h, but it can already be used under non-nominal conditions if needed, for example, for attitude control purposes or in case of emergency. Once the priming is completed, the spacecraft retains its configuration until the drift orbit (see Sect. 4.1) for a geostationary satellite or possibly the end of the mission for a LEO satellite.

The insertion into the final orbit is a delicate operation. In case of a geostationary satellite, a box of $\pm 0.1°$ (around 150 km wide) around the nominal longitude is allocated and collocating more than one spacecraft at a position is common. It is thus necessary to avoid neighbor boxes as much as possible. Reaction thrusters are better suited for such maneuvers as well as for the further station keeping activities because their low thrust allows a finer control of the orbit. The apogee engine being not needed anymore, it is isolated from the propellant system by closing a pyrotechnical valve. Since thrusters require a lower mass flow than the apogee engine, the pressure regulator is not needed anymore. The inert gas tanks and pressure regulator are isolated from the propellant tanks by closing another pyrotechnical valve. All these measures are taken to ensure the safety of the spacecraft. Once these activities are completed, the so-called blow-down phase begins and the pressure within the propellant tanks will constantly decrease until the end of the mission. As a consequence, thrust performances will decrease with time. De-orbiting will be performed in this configuration.

6.4.3 *Real-Time Operations*

Knowledge from the two previous sections is gathered by the propulsion engineer from the spacecraft user manuals during the preparation phase preceding the launch. After launch comes the thrill of real-time operations!

6.4.3.1 Monitoring During Quiescent Periods

There is always something to do in operations, even when no activity is planned! The very first check is the limit checking. For each telemetry parameter, domains of values can be defined and flagged automatically by the telemetry processor. There are typically two types of limit domains: a warning range, flagged yellow, indicating that the values shall be monitored but that no immediate action is needed and an alarm range, flagged red, indicating a possible damage to some equipment and requiring an action. Warning and alarm domains are defined by the spacecraft manufacturer. Operations engineer, however, adapt these limits to the situation, sometimes setting them tight to be warned soon enough, sometimes loosening them when a situation is well understood and under control. Limit checking is the first easy check to get an overview of the system status. Under nominal conditions, no warning or alarm shall appear in the telemetry display system. When a warning or an alarm is raised, it is always necessary to analyze the situation and get a full understanding of what is happening before taking any action. Many automatisms on board, like FDIR, can cause warning or alarms flagging and necessitate a thorough analysis across many different subsystems.

The second monitoring covers the pressure within the entire propulsion system. When no firing occurs, the pressure should stay constant, except for the variations caused by temperature changes: as a matter of fact, gas pressure reacts to temperature changes according its state equation.[20] A decrease in pressure could indicate a leakage or a thruster valve that failed to close. Although a failed closing is rather obvious, leakages can be hard to detect if they are small. In this case, only an analysis over a long period of time might highlight the problem. Temperatures shall also be monitored, especially those of the thruster combustion chambers. An increase might indicate a reaction and possibly a leakage.

Another value to monitor is the time thrusters are in use (so-called on-time). This duration is usually produced by the OBC and is available in telemetry data. The more these values change, the more the thrusters are active. When an increase in the on-time values is monitored, it is always necessary to ensure the reason is understood. A leakage or a valve that failed to close can be the root cause.

[20] For example the state equation of an ideal gas reads $PV = nRT$, where P denotes the gas pressure [Pa], V its volume [m^3], n its number of moles [mol], and T its temperature [K]. $R = 8.314$ J mol^{-1} K^{-1} is the ideal gas constant.

Monitoring details depend of course on the spacecraft design but the previous examples give an overview of the monitoring philosophy. Whatever the combination of limit checking, long-term analysis or telemetry values, the goal is to understand what is happening within the system.

6.4.3.2 Orbit Maneuvers

Orbit maneuvers are the most important activities performed by the propulsion system. It is carefully prepared off-line (see Sect. 6.4.4.1) and its execution is considered as a critical event.

Figure 6.36 gives an overview of the pressure evolution during the second Apogee Maneuver Firing (AMF) of a geostationary mission whereas the corresponding temperature evolution is shown in Fig. 6.37. Both figures highlight the main events happening during an orbit maneuver: the maneuver starts by opening the apogee engine valves, which result in a pressure drop in the propellant tanks. This drop in pressure is caused by propellant flow towards the engine

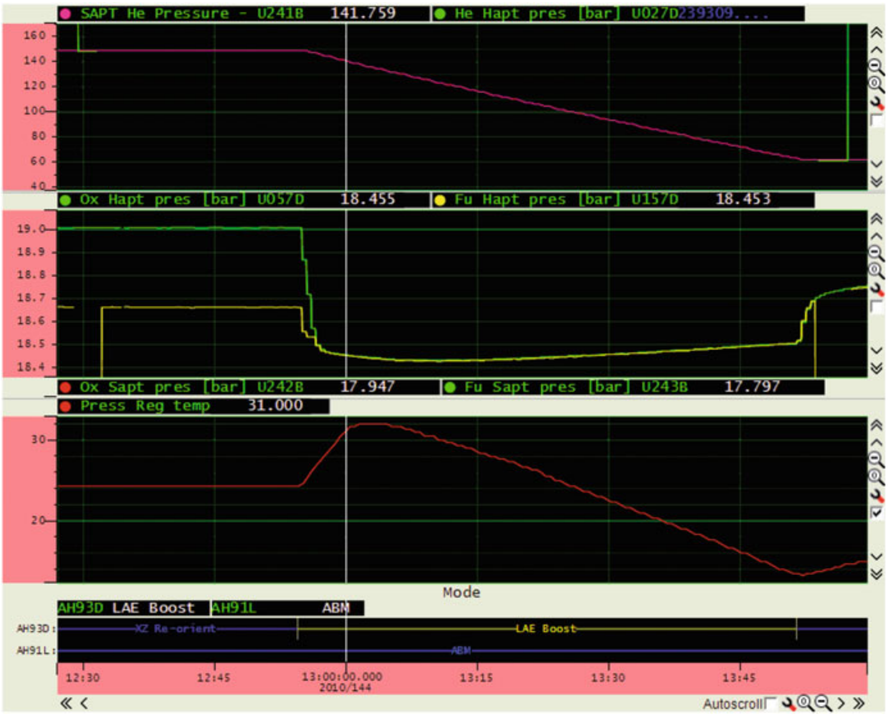

Fig. 6.36 Pressure evolution during AMF2. First plot on *top* displays the pressure in the helium tank. The second plot shows the pressures in both oxidizer (U057D in *green*) and fuel (U157D in *yellow*) tanks. The third plot shows the pressure regulator temperature and the last plot at the *bottom* depicts the state of the AOCS software, showing when the orbit maneuver takes place

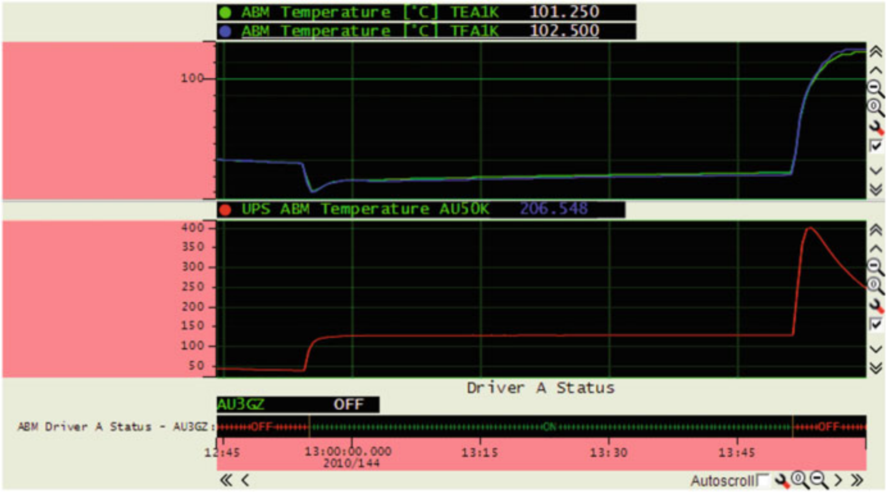

Fig. 6.37 Temperature evolution during AMF2. First plot on *top* shows the temperature at the inlet of the combustion chamber. The second plot displays the temperature measured closed to the combustion chamber. The last plot at the *bottom* gives the status of the engine valve, showing when the orbit maneuver takes place

combustion chamber, which increases the space available to the inert gas pressuring and thus decreases its pressure. At the same time the helium pressure starts decreasing as a result of the pressure regulator activity, which tries to compensate the pressure drop in the propellant tanks by pumping some gas into the propellant tanks. After a short period, the pressure in the propellant tanks reaches a quasi steady state as a balance between the two flows mentioned previously. The slight increase in propellant pressure reflects the fact that the pressure regulators pump helium into the propellant tanks at a higher rate than the chemical reaction burns propellant in combustion chamber. When the engine valve is closed at completion of the orbit maneuver, a sharp rise in pressure is observed in the propellant tanks. This results from the stop of the propellant flow while the pressure regulator is still pumping gas into the propellant tanks. Afterwards the pressure in the tanks will increase until the set point of the pressure regulator is reached.

The pressure regulator temperature increases at the beginning of the maneuver. This is due to the Joule–Thomson effect: helium has a negative Joule–Thomson coefficient at the operational temperature, which means that the gas warms up when expanded (from 150 to approximately 18.5 bar during this maneuver, as shown on Fig. 6.36) in the pressure regulator. Later on, the pressure regulator temperature decreases because of the pressure drop in the helium tank, which leads to a decrease of the helium temperature.

The temperature measured at the inlet of the combustion chamber first decreases because of the propellant flow and then reaches a steady state. As expected, the temperature close to the combustion chamber rises due to the strongly exothermic chemical reaction. More surprising is the sharp rise of temperature at completion of the orbit maneuver: during the maneuver, the exhaust flow takes away a part of the thermal energy produced by the chemical reaction by convection. When the engine valve is closed, the flow stops and the thermal energy can only be evacuated by radiation and conduction. Since the radiation depends on the temperature, the heat quantity evacuated by this mechanism remains relatively constant at first. Consequently the heat starts flowing back into the relatively cold spacecraft by conduction: this is called the "heat soak-back effect" leading to an important increase of temperature (more than 350 °C close to the combustion chamber) in the spacecraft around the apogee engine. A shield is usually installed around the engine to minimize the effect on the internal equipment.

Another phenomenon that can occur is propellant sloshing in the tanks. Sloshing depends on the propellant mass and the space being available to move freely. It is consequently reduced during AMF1 because the tanks are almost fully filled and after AMF2 because the propellant mass left is low. Sloshing effects are best monitored in the pointing error of the spacecraft at beginning of the maneuver.

Due to the criticality of an orbit maneuver, it is important for a propulsion engineer to assess quickly whether it is performed nominally or not. This is especially valid at the beginning and the end of the maneuver. Nominal start of maneuver is first detected in the attitude system, where the spacecraft rates reflect the engine start. The thermal system can also confirm correct start by monitoring the temperatures. Finally the propulsion engineer also confirms the start by looking at the valve status. The end of the maneuver is more critical because a failure at engine valve closure could lead to a premature end of mission. The same systems as previously can confirm a nominal end of maneuver, the heat soak-back and the rate stabilization being the major indicators. The propulsion engineer shall, however, monitor closely the pressure in the system for a certain duration to ensure no leakage has buildup.

The effects described in this section are quite patent for engines producing a high amount of thrust like the apogee engine. For reaction thrusters, some effects might become so tiny that they cannot be observed.

For other technologies, a preparation time is sometimes necessary and shall be taken into account when planning the orbit maneuvers. It is for example necessary to preheat the catalyst of a mono-propellant system with hydrazine to avoid degradation. The same preheating time is necessary for the neutralizer of electrical engines. In case of electrical propulsion, the power balance and the battery status are to be checked carefully before starting the maneuver.

6.4.3.3 Isolation of Apogee Engine (GEO)

As mentioned in the section about the operational configurations, the apogee engine is isolated from the rest of the system for safety reasons after reaching the drift orbit. Pyrotechnical valves can only be moved once and irreversibly so it is necessary to ensure the apogee engine will not be used anymore. This confirmation is gained from a precise orbit determination, which shall demonstrate that the remaining ΔV required to park the spacecraft into its box can be achieved by the using reaction thrusters only. GPS are currently not used on-board geostationary satellite because their orbit is above the GPS constellation, leading to difficulties in acquiring constant signals. Orbit determination must then rely on ranging data (see Sect. 4.1, especially Figs. 4.17 and 4.18). Moreover Doppler data is not of much help because the drift orbit is close to circular and the radial velocity small. In order to improve the orbit determination, ranging data is acquired alternatively from two ground stations with enough separation in longitude.

The isolation closes the supply of inert gas into the propellant tanks. From this point on, the inert gas pressure in the propellant tanks will decrease until the end of the mission. Sometimes, the spacecraft manufacturer may decide to optimize slightly the thruster performances, which depend directly on the pressure of the inert gas within the tanks. This is achieved by switching off the tank heaters to cool down the propellant and the inert gas. The propellant volume does not vary much during this operation so that the inert gas volume remains almost constant. Consequently a decrease in temperature results in a decrease of pressure, which is compensated by the pressure regulator: this trick allows getting a higher pressure of the inert gas within the propellant tanks after reactivating the thermal control. It is finally important to record carefully pressure and temperatures within the entire propulsion system just after isolation for the purpose of mass calculation and lifetime estimation (see both sections under off-line operations).

6.4.3.4 Autonomous Operations

Orbit maneuvers, either performed with the apogee engine or the reaction thrusters, are performed by directly commanding the opening and closing time of the engine valves. Reaction thrusters create a force on the spacecraft, which can be used for attitude control when its direction does not go through the current center of mass. If this is the case, the thruster also generates an external torque, which can be used to control the spacecraft attitude or its angular momentum.

Due to the complexity of the attitude motion (see Sect. 4.2.5), the Attitude and Orbit Control System (AOCS) usually controls autonomously the thruster firings. In case the AOCS controller aims at rotating the spacecraft, the thruster firings are combined so as to result in a torque around the desired axis, with the desired magnitude. The torque obtained with reaction thrusters can be used to control the spacecraft angular momentum. This is especially interesting when a reaction wheel assembly is used to control the attitude. As a matter of fact, this assembly stores the angular momentum from external perturbations, which cause the wheel to reach their saturation speed. Reaction thrusters help desaturating the wheels the same way magnetic torquers do. This activity is usefully combined within station keeping maneuvers to take advantage of the resulting force on the spacecraft.

During such autonomous operations, monitoring is more qualitative and based on physical sense. For example the run-up of a momentum wheel is counteracted by thruster firing to keep the platform orientation steady. The propulsion engineer shall check whether the thruster activity displayed by telemetry data actually corresponds to a torque along the axis of the momentum wheel.

6.4.4 Off-line Operations

The real-time operations described previously require some preparation work and post-performance analyses. Such activities are referred to as “off-line” as opposed to the actual operations.

6.4.4.1 Preparation and Calibration of Orbit Maneuvers

The preparation of orbit maneuver involves three parties: a flight dynamics engineer who specifies the required ΔV and the center of burn (time of apogee crossing from orbit determination), the propulsion engineer who provides the spacecraft mass, and the propulsion expert of the satellite manufacturer who provides corrections based on flight experience.

The thrust direction is tangential to the orbit at the apogee, which can be translated into a spacecraft attitude during the orbit maneuver. Flight dynamics experts provide the commands for the reorientation as required by the spacecraft AOCS. The ΔV and the center of burn depend on the other purely on the current and final orbits. These quantities shall be translated into start and stop times by taking the amount of thrust the apogee engine can actually deliver. Neglecting the gravity with respect to the thrust and assuming a constant specific impulse, the ideal rocket equation [see Eq. (6.9)] yields:

$$m_{\mathrm{f}} = m_{\mathrm{i}} - \Delta m = m_{\mathrm{i}} e^{-\frac{\Delta V}{g_0 I_{\mathrm{sp}}}}$$

of

$$\Delta m = m_{\mathrm{i}}\left(1 - e^{-\frac{\Delta V}{g_0 I_{\mathrm{sp}}}}\right). \tag{6.10}$$

Knowing the initial mass, provided by the propulsion engineer from its mass calculation records (see Sect. 6.4.4.2 further), and the specific impulse one can compute how much propellant mass will be used during the maneuver. The knowledge of the expected mass flow rate $\frac{\mathrm{d}m}{\mathrm{d}t}$ allows ultimately computing how long the burn shall last. Both the mass flow rate and the specific impulse depend on the pressure and temperature within the propulsion system. Their mathematical expression is fitted to on-ground tests results, for example, as a polynomial or in form of tables, and is given in the spacecraft user manual. The propulsion engineer can provide the current pressure and temperature values and thus help computing an estimate of the burn duration. However, this estimate would not be accurate because the pressure and temperature values needed are the not current ones, but the ones *during the maneuver*! Moreover Fig. 6.36 shows that the pressure in the tank drops during the maneuver and this pressure drop varies with the tank filling state. Finally the pressure does not remain constant during the maneuver but increases slightly over time as shown on Fig. 6.36. At this point, the propulsion expert of the spacecraft manufacturer can help by providing more accurate data based on flight experience, usually in the form of a mean thrust level to be expected. Once the maneuver duration has been computed, the start and stop times are easily deduced from the center of burn time.

Previous computations are additionally complicated by the fact that the reaction thrusters are also used for attitude control. The thrusters are activated autonomously by the AOCS control to create torques required to keep the apogee engine aligned with the desired thrust direction. These torques result from forces not going through the center of mass, but these forces also contribute to the ΔV in an unpredictable way! The gravity, which has been neglected in the previous computations, also affects the maneuver performance as it curbs the orbit (see gravitation losses), especially for long maneuvers. It is consequently necessary to calibrate the orbit maneuver after its performance. Precise orbit determination yields the actually achieved ΔV: it is then possible to compute a correction factor (for example on the mean thrust level) that will be taken into account for the preparation of the next maneuver.

6.4.4.2 Propellant Mass Calculation

The previous section showed clearly the need to know the spacecraft mass. It can be divided into the dry mass covering the spacecraft equipments and the wet mass that comprises the propellant and different gases used in the propulsion system. The dry mass is measured accurately on ground before launch. The wet mass on the other hand is known at launch from the filling of the tanks. The wet mass decreases as propellant is burnt and this mass cannot be easily measured directly. As a matter of fact, a direct measure of the remaining propellant mass is complicated by its distribution within the tank, which is governed by the propellant surface tension under microgravity conditions. This is the reason why the remaining propellant mass is usually estimated by two complementary methods.

The first method is based on the thermodynamic properties of the inert gas used to pressurize the propellant tanks. It uses pressure and temperature measurements available in telemetry data to compute the volume occupied by the gas from its state equation. A simple subtraction from the tank volume yields the propellant volume and the propellant density ultimately allows computing the mass. These quantities explain the name of PVT (Pressure–Volume–Temperature) given to this method. Although the principle is straightforward, different effects shall be taken into account to get a correct accuracy. First of all ideal gas law does not hold accurately at pressures above 5–10 bars and it is necessary to resort to more complicated state equations. A practical way is to introduce a correction factor $Z(P, T)$ such that the state equation now reads $PV = ZnRT$. Values of the correction factors are then provided for the operational range of pressure and temperature, either as a function or as table. A liquid evaporates and its vapor mixes with the gas used to pressure the propellant, leading to a change in the tank pressure. The effect of this gas mixture on the measured tank pressure is sometimes taken into account. Moreover gases dissolve themselves into liquids and shall be accounted for when computing the number of moles n. One can finally mention the dependency of the propellant density with the temperature and the expansion of the tanks under pressure. All these effects may seem so small they could be neglected, but it will be seen that the accuracy of the thermodynamic method is closely related to EOL, which justifies taking these effects into account. Another pitfall of the PVT method is the activation of relief valves when the pressure in the system becomes too high. If this happens, it is necessary to estimate how much gas escaped from the system.

The second method is based on the knowledge of the propellant mass flow rate inside all engines. As for the correction factor introduced previously for real gas, the mass flow rate of an engine is measured on ground and provided to the propulsion engineer as a function of pressure and temperature whose coefficients are specified or as a table. In any case, the knowledge of the mass flow rate allows computing the propellant mass used by the engine by integrating it over the maneuver time. The remaining propellant mass is then simply computed by subtracting the mass used during engine activation to the mass present before the activation. Due to this accounting of all the mass used during the actuations, this method is called "bookkeeping."

Having two methods is a good way to cross-check computations. They, however, rarely agree and the difference can be in the range of 10 kg at EOL! The bookkeeping method is more accurate than the thermodynamic method on the short term. The latter has an accuracy determined by the accuracies of the involved sensors and the underlying thermodynamic model. The bookkeeping method on the other hand is applied to every maneuver and inaccuracies add up over time. Which of both methods has the better accuracy, especially at EOL shall be carefully analyzed by the satellite manufacturer.

Another method based on measuring the thermal capacity of a propellant tank has emerged in the last decade. The thermal capacity depends on how much propellant is left in the tank and can be estimated by measuring the tank temperature response to a controlled heating. This estimation, however, requires modeling the propellant tank, the location of the probes, and the location of the temperature sensors. This method has the advantage to be more accurate when the remaining propellant mass decreases.

6.4.4.3 Center of Gravity Calibration

Mass calculation is closely related to the position of the spacecraft mass center and the inertia tensor. Both are used in the attitude control algorithms and are parameters loaded in the SGM of the satellite. Due to the mass consumption by the propulsion system, it is often necessary to update the values of these parameters.

Some spacecrafts, like the spacebus 3000, cope with this issue by setting the value of a predefined index once per year. This index corresponds to a predefined position of the center of mass and is used as input by the AOCS. Values are chosen such that it approximates the nominal propellant consumption over a year, but it might be necessary to choose the value differently, in case the spacecraft had an anomaly or needed to be repositioned at another orbital position for example.

The GRACE mission has particularly high requirements about the knowledge on the center of mass position. This mission is composed of two satellites separated by around 220 km. A microwave link between the two satellites allows observing the earth gravity field. It is, however, necessary to filter out nongravitational accelerations, which is achieved with an accelerometer. In order to minimize accelerations resulting from satellite rotation, the center of mass is kept fixed with respect to the spacecraft structure by moving small masses around the satellite. Due to the required accuracy, dedicated attitude maneuvers around the three axes of the spacecraft are commanded regularly to accurately calibrate the center of mass position. After an off-line analysis of the results, the positions of the small masses are commanded such that the center of mass remains at its allocated position.

6.4.4.4 Lifetime Estimation

A mission is very often put to an end not because the payload is not functioning properly, but rather because propellant runs out! Since debris are a growing concern, spacecraft operators are more and more required to leave useful orbits (LEO or GEO) and put their property either into graveyard orbits or de-orbit it into the atmosphere. The propulsion engineer plays an important role in this decision because his/her mass estimation, taking into account de-orbiting maneuvers, is an important input to call the end of mission. It adds to the necessity to know which of the mass calculation methods shall be trusted for this decision. However, pressure is high to postpone the EOL and "push the envelope" as much as possible.

De-orbiting maneuvers are computed by flight dynamics experts. Due to almost full depletion of the propellant tanks or postponing the EOL more than expected, it can be difficult to predict the amount of thrust that will be produced. As a matter of fact, bubbles of pressuring gas can degrade the combustion and result into non-nominal performances.

6.5 Attitude and Orbit Control Subsystem Operations

Ralf Faller

This chapter will give a general overview of the attitude and orbit control subsystem, its components, and the operational tasks to be performed by the ground control personnel. It will complement the basics of attitude and orbit control explained in Sect. 4.1.

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After a short introduction and an outline of its relevance for different needs coming from other subsystems and the payload, an overview of the subsystem is given in Sect. 6.5.2, where its components and most common applications will be described and the basic functions of the on-board control will be explained. With this background, Sect. 6.5.3 will summarize the tasks for the ground control team during different mission phases. Some lifetime experiences in different missions will be reported in Sect. 6.5.4. The AOCS topic is finally summarized in Sect. 6.5.5.

6.5.1 Introduction and Overview

The AOCS is one of the main subsystems responsible for providing both favorable conditions for other satellite subsystems and the payload with respect to attitude and orbit.

Attitude comprises two aspects: The first is the orientation of the spacecraft with respect to reference objects or directions, e.g., earth, sun, stars, flight direction, etc. The attitude accuracy requirements encompasses the full range of levels, from a rough orientation of a satellite towards the sun for sufficient power supply to the order of some degrees, to high precision orientation tasks down to the order of arc seconds, e.g., for orientation of an astronomical telescope towards dedicated targets. The second aspect of attitude control is its dynamic. Some satellites need to change their orientation from time to time, have to rotate about dedicated axes (e.g., daily pitch rotation of geostationary satellites), or need to stabilize their orientation after separation from the launcher. Thus rotation and rotation rate control are tasks of the attitude control.

Orbit control comprises all aspects of maintaining or acquiring a target orbit. Geostationary communication satellites have to be positioned on the GEO and need to be kept in their control boxes during the whole lifetime for adequate payload support. In the same way, all kinds of LEO satellite missions need to acquire and maintain their target orbits, have to maintain formations, or need to change orbits depending on the mission purpose. Another aspect is the avoidance of collisions with other objects. In times with a more and more crowded space, this topic is relevant for both manned and unmanned spaceflight missions.

Altogether, maintaining attitude and orbit are tasks which are provided by the corresponding control system. The on-board control itself is realized by three main instances. The data about the current attitude and orbit are received by sensors in form of aspect angles, spin rates, or position measurements. These data are collected by a control unit, which derives corresponding attitude and orbit-related parameter. By comparison with nominal values, deviations are determined and correction (control-) demands are generated. These control demands are sent to

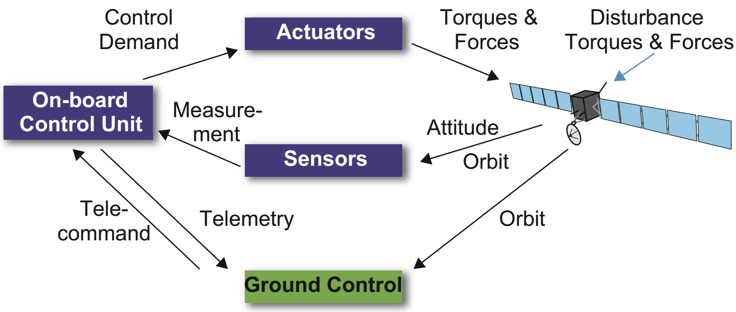

Fig. 6.38 Subsystem components and control cycles

Actuators, like thrusters or other equipment generating control torques and forces, which correct the attitude or orbit. Figure 6.38 illustrates these main components of the on-board control cycle.

Attitude control is done via a classic on-board closed-loop control. The ground can adjust control settings and margins or can change control strategies (see attitude modes), but the control loop requires short round-trip times of the control signals and, thus, the loop is always closed on board.

Orbit control works partly different. Orbit correction maneuvers are normally done at discrete times and not permanently. The current orbit is measured and calculated, either on board using global positioning data (e.g., GPS or Galileo) or classically on ground using station tracking information. The decision for and the preparation of an orbit maneuver is done on ground, then commanded to the satellite, and finally executed by the S/C at the dedicated maneuver time. There are normally no real-time requirements for the control loop. Only missions with permanent orbit control demand (e.g., rendezvous and docking situations) need to have an on-board closed-loop control.

6.5.2 Subsystem Description

With the basic understanding of the attitude and orbit control described in Sect. 6.5.1, this chapter will give an overview about most common sensors and actuators, and will describe the typical functions of the control unit. The descriptions only focus on the main aspects.

6.5.2.1 Sensors

The main task of sensors is the measurement of S/C position, orientation, and dynamics mainly w.r.t. celestial or other reference objects. The output of these sensors might be S/C position and velocity vectors, attitude aspect angles, rotational

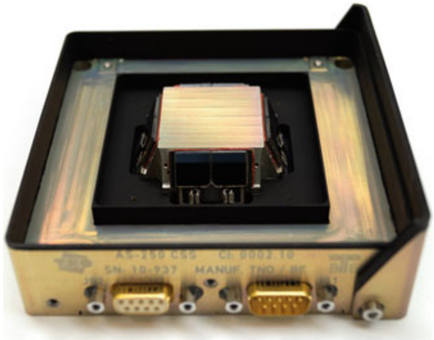

Fig. 6.39 Coarse sun sensor (Image: Bradford-Space.com)

rates, or inertial orientations. Whilst orbit-realted data are given with respect to earth-related coordinate systems (e.g., ECEF), attitude measurements are typically provided with respect to reference axes of the sensor system or the spacecraft body. Depending upon which kind of information is used, attitude information about one, two, or all three axes is provided. The following list of typical sensors gives a brief overview.

Sun Sensor

This group of optical sensors is most commonly used for spaceflight applications. The brightest object in our solar system represents an unambiguous orientation source. Different technical solutions are available to provide sun direction data with an accuracy between a few degrees in case of simple safe mode sensors and better than 0.05°. Another main characteristic is the large field of view of a single sensor head, ranging in the order of ± 60–90°. Shading of the sunlight by other objects (e.g., eclipse phases by earth or moon) need to be operationally considered. For some sun sensor models, the earth albedo might reduce the attainable accuracy. Since the sun is rotation symmetric for more or less all sensor applications, only 2-axes attitude information can be obtained (Fig. 6.39).

Earth Sensor

Classic earth sensors are providing direction information by scanning the infrared disk of the earth. They are commonly used for earth-related missions with medium pointing accuracy requirements, like geostationary communication satellites. These sensors could be disturbed by other infrared objects like the sun or the fully illuminated moon, which need to be respected by the flight operations.

Other earth sensor applications are available using imaging systems (cameras). In addition, combinations of sun and earth sensors have been developed. Such

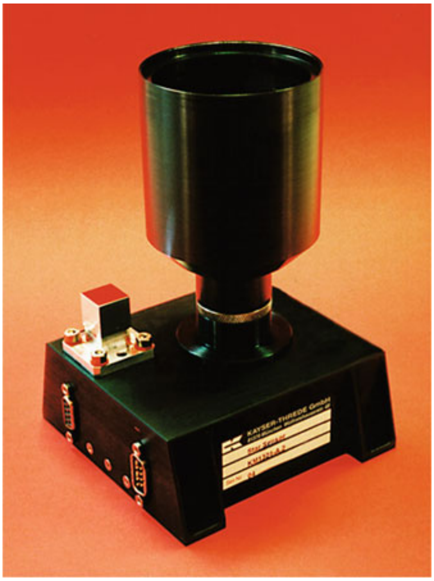

Fig. 6.40 Star sensor KM1301 (Image: Kayser-Threde GmbH)

sensors consist of thermistor elements providing coarse direction data of both celestial bodies.

Star Sensors

This class of optical sensors is using stars as orientation source. Different to earth or sun sensors, star sensors provide an inertial attitude reference about all three axes. The attainable accuracy is in the order of arc seconds. Disturbance by sun or moon blinding is possible and is normally respected per satellite design by implementing 2–3 sensors in different directions (Fig. 6.40). See also Sect. 4.2.

Magnetometer

The earth magnetic field can also be used as an orientation source. Magnetometers are measuring the direction of the local magnetic field lines and, thus, provide a fair 2-axes attitude reference. Fairly accurate mathematical models are available, but the earth magnetic field is influenced by sun activities and, thus, magnetometer measurements are seldom input for attitude determination calculations. Nevertheless, magnetometers are commonly used in combination with magnetic torque rods to provide attitude control torques.

Gyroscopes

Gyroscopes are measuring spacecraft rotation rates. Compared to the sensors described so far, gyros are providing relative attitude information. They are mainly used to get data about attitude changes and, therefore, allow to cover phases with reduced absolute sensor data availability (e.g., during eclipse phases or slews from one orientation to the next). All kinds of gyros have a drift, which cause accumulative rate integration errors over time.

Space-Based Satellite Navigation Systems

Satellite navigation systems, like the Global Positioning Systems GPS (others: GLONASS, GALILEO), can be used to gather position (and velocity) data of LEO missions. These data are input for on-board navigation systems, which perform orbit determination tasks in space. In addition, specially equipped GPS systems can be used for attitude determination, preferable for larger spacecraft.

Other Sensors

Besides these common sensor systems described above, a few other sensor concepts might be in use. As an example, Radio Frequency Beacon sensors are mentioned here. They are using RF sources on ground as orientation reference with an accuracy of order 1 arc minute.

6.5.2.2 Actuators

The change of the spacecraft's velocity or its rotational rates can be realized by different actuator systems generating the required control torques and forces.

Thrusters

Thrusters are the classic equipment for both attitude and orbit control. Depending on the spacecraft's complexity, they are sometimes part of a separate subsystem as described in detail in Sect. 6.4, but activity demands are always generated by the AOCS subsystem. Since the amount of fuel aboard a spacecraft is limited, the availability of all thrusters systems is limited correspondingly.

Fig. 6.41 *Left*: Magnetic torquer VMT-35 (Image: vectronic-aerospace.com); *right*: COMPASS-2 Cubesat air coil (Image: FH Aachen)

Magnetic Torquer

These actuators are using the effect of the interaction with the magnetic fields. Electric magnets aboard a spacecraft, so-called magnetic torquer rods, can generate control torques by interacting with the earth magnetic field. By mounting 3 torquer rods in the 3 body axes of the spacecraft, control torques about 2 axes can be achieved rectangular to the local magnetic field lines only. However, this can be compensated using the change of the earth magnetic field vector over an orbit. Different to thrusters, magnetic torquer is available as long as electric power is available. They are provided in two different applications, torque rods with an iron core and air coils (Fig. 6.41).

Wheels

In principle, a reaction wheel consists of a rotating mass attached to an electric motor. By increasing or decreasing its rotational speed, the angular momentum of the mass increases or decreases. Due to the law of conservation of momentum, the total angular momentum of the spacecraft remains constant as long as no external torque is applied. As a consequence of speeding up the motor of the reaction wheel and the corresponding increase of its angular momentum, a reverse-directed momentum is applied to the satellite's structure. This effect can be used for the attitude control. Different to thrusters or magnetic torquer, the angular momentum of the whole system is not changed. Angular momentum is only shifted between the spacecraft body and the wheel.

Two different types of wheels are available, reaction wheels with nominally zero speed and momentum wheels, so-called fly-wheels, running on a high speed. The latter kind of wheels is also used to store a portion of angular momentum in the spacecraft in order to provide some dynamic stiffness against external disturbance torques.

Fig. 6.42 Left: Reaction wheels on Lunar Reconnaissance Orbiter, NASA GSFC (Image: NASA); *right*: Reaction wheel unit (Image: bradford-space.com)

Due to friction effects or systematic external disturbance torques, wheels need to be de-saturated from time to time when their maximum speeds are reached. This is called "wheel unloading" or "wheel desaturation." Therefore, purposeful torques are generated using thrusters or torquer, which can be compensated by the wheel by reducing its speed (Fig. 6.42).

6.5.2.3 On-Board Control Unit

The on-board control unit is the central element of the AOCS. It can be a separate piece of hardware or a dedicated process running on the board computer. The main tasks of the AOCS control unit are:

Calculation of Current Attitude and Prediction

All sensor measurements are used to calculate the current S/C attitude, rotation rates, and their deviations from nominal values typically by means of numeric filtering (e.g., Kalman filter). In addition, the development of the parameters for the next control grid point (see Sect. 4.2.3) is also predicted.

Examples for such attitude parameter are the sun aspect angle for sun-pointing spacecraft or errors in roll, pitch, and yaw in case of an earth-oriented geostationary satellite.

Generation of Control Arguments for Attitude Correction/Adjustment

Using the predicted attitude deviations, control demands are generated, like speeding up or down of a corresponding reaction wheel, on/off switching of magnetic torquer, or firing of selected thrusters in order to get a desired attitude change.

Wheel Unloading

As mentioned above, wheels need to be saturated from time to time, when they reach its maximum speed (typically a few thousand RPM). Continuous wheel unloading is also possible.

This is process is completely controlled by the AOCS without ground support.

Calculation and Prediction of Orbit-Related Data

Spacecraft equipped with an on-board navigation system are calculating the orbit using GPS measurements.

Error Detection and Reaction

Early detection of problems within the AOCS and autonomous reaction of the spacecraft without immediate activities by the ground control is a mandatory function for all kinds of spacecraft. This is realized by respective FDIR mechanism.

AOCS Modes

More or less all spacecraft have different operational states, so-called operational modes. By selecting a specific mode, different settings and preselection are used for the attitude control. The most relevant settings are:

Selection of the Control Strategy

For example switching into a sun-pointing mode sets the sun as main orientation reference; the spacecraft will orient itself relative to this direction.

Setting of Control Ranges and Limits

For modes with high attitude accuracy requirements, the control margins are kept narrow, whilst for modes like a safe mode, only a coarse attitude is needed and the control margins can be set to a larger interval.

Preselection of Needed Sensors and Actuators

The equipment required for the corresponding mode is activated, others are switched off. For instance, a star sensor would be switched on for a mode with high pointing accuracy, but deactivated in safe mode.

Selection of FDIR Strategies

This means enabling or disabling of predefined reactions on potential malfunctions. Table 6.5 shows some example AOCS modes and corresponding settings.

Table 6.5 Example AOCS mode

Mode	Orientation strategy	Control limits	Sensors/ actuators
Safe mode (sun pointing)	z-axis towards sun slow spin around z (2-axes stabilization)	0.0° ± 0.3 in both axes 0.4°/s ± 0.01 about z	Sun sensor Gyros Thrusters
Inertial target pointing	Inertial target orientation (3-axes stabilization)	±0.002° in all 3 axes	Star camera Gyros Magnetometer Reaction wheels Magnetic torquer
Earth pointing	z-axis oriented to earth, y-axis perpendicular (south) to the orbit plane, x-axis quasi in flight direction (3-axes stabilization)	0° ± 0.01 in roll and pitch 0° ± 0.2 in yaw	Sun sensor Earth sensor Gyros Thrusters Wheels

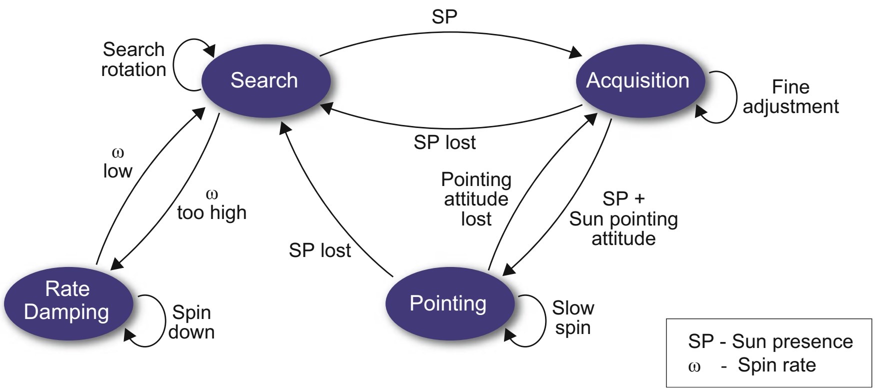

Fig. 6.43 Sub modes and control laws for a sun-pointing mode

Sub-Modes

Each mode might have different states, so-called sub-modes, to allow tiered activities and working levels. As an example, Fig. 6.43 shows sub-modes of a simple sun-pointing mode. The lowest one is a rate damping state, where only the rotational rates of the spacecraft are dumped to low levels in order to provide adequate conditions for the attitude measurement. Being the default entry state after invoking the sun-pointing mode, this sub-mode is mainly defined for the first sun orientation after separation from the launcher, when the spacecraft was put into some high spin rotation. When the rates are low enough, the on-board control changes to the search sub-mode. Here, the mode logic checks, if the sun is already in the sun sensors field of

Table 6.6 Examples of possible error levels and adequate FDIR reactions

Error level	Error/problem	FDIR reaction	Impact on payload
1	Sensor failure, indicated by equipment status bit	Switchover to redundant sensor	None Payload activities are continued Ground control has time to analyze the problem
2	Attitude error exceeds limits	Switchover to redundant set of equipment	Possible
3	Loss of attitude reference (e.g., no attitude solution available)	Switch to safe mode Switchover to redundant set of equipment	Yes Payload is shut down Ground control needs to analyze and react, until payload operations can be resumed

view. (Of course this sub-mode does not make sense for spacecraft with an overall field of view). If the sun sensors do not provide a sun presence (SP) signal, the spacecraft starts rotation about 1 or 2 body axes to bring the sun into sight. When a valid sun presence signal is available, the acquisition sub-mode orients the dedicated spacecraft body axis towards sun. As soon as this has been accomplished adequately, the highest sub-mode is invoked. Here, a slow spin about the sun orientation is applied providing some passive attitude stabilization.

Fault Detection, Isolation, and Recovery

FDIR routines and functions are a key element of the AOCS. Whilst other subsystems normally have longer response periods until a problem becomes severe, failures in the AOCS control cycle might immediately lead to loss of the correct attitude and possibly to bad thermal and power conditions or to a loss of the communication link. Therefore, autonomous reaction strategies of the on-board control are implemented for various error cases. A key element of these FDIR routines is the adequacy of the reaction with respect to the occurred problem. For instance, a minor glitch in a sensor system might not need to send the complete spacecraft into safe mode, which would cause a payload shutdown and a major disruption of the current mission activities. A simple switchover to the redundant sensor would be more suitable and would allow maintaining the current activities. For adequate reactions, different error levels and corresponding reactions are defined. Table 6.6 gives some exemplary error levels and corresponding reactions for a scientific LEO satellite mission.

Table 6.7 Possible sensor/actuator combinations

	Mission		
Equipment	Commercial TV geostationary	Scientific near-earth	Scientific deep-space
Sun sensor	++	++	+
Earth sensor	++	+/−	−
Star tracker	+	++	++
Magnetometer	−	+	−
Gyros	++	++	++
Thruster	++	++/−	++
Wheels	++	++	+
Magnetic torquer	−	++	−

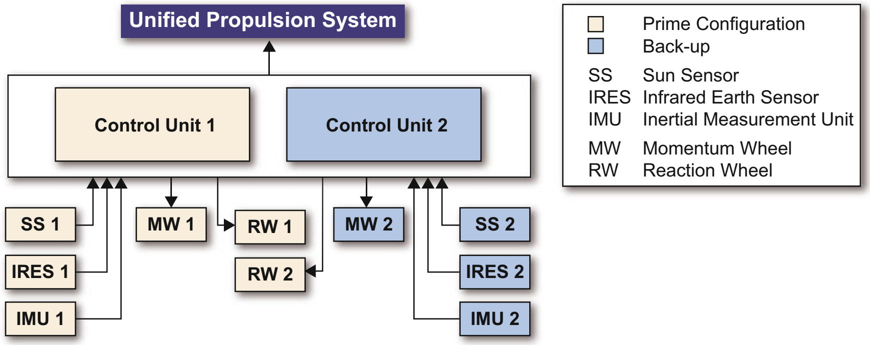

Fig. 6.44 Functional AOCS block diagram for a GEO satellite

6.5.2.4 AOCS Equipment Combinations and Redundancy

As described in the previous sections, different sensors and actuators are available with various technical realizations and performance levels. Table 6.7 shows sample combinations of sensors and actuators. For three different kinds of missions, the equipment is labeled as most suitable (++), suitable (+), or not suitable (−), but other combinations are possible too and also depending on mission constraints, the project budget, and other S/C design drivers. For instance, thrusters are the main actuator for orbit control, but could not be used for scientific mission with sensitive sensor systems, since the exhausted gases of the thrusters would produce some kind of pollution of the space around the spacecraft.

Besides the selection of adequate sensors and actuators, the number of units per category is another relevant aspect for the operations. As for other unmanned spaceflight applications, a single failure tolerance is normally achieved by using two pieces of equipment, one being active and in control, and the other one switched off and available as cold redundancy. Fig. 6.44 shows an example of a functional AOCS block diagram for a geostationary communication satellite. For each kind of sensor, a redundant one is available. Also available is a second control unit. In case of a loss of one of the wheels, a redundant fly-wheel is available. In this example, the thrusters are covered by a separate subsystem.

6.5.3 AOCS-Related Ground Operations

6.5.3.1 Basic AOCS Ground Activities

This section describes common tasks and responsibilities of an AOCS engineer during all mission phases. As for all other satellite subsystems, flight procedures shall be available for the AOCS in order to provide a reliable basis for flight operations.

Monitoring

The most common ground activities are all kinds of monitoring and analyzing of the subsystem functions, equipment, and performances. In more detail, AOCS monitoring comprises the following items:

Check of Current Status

The subsystem status is provided by the AOCS mode and sub-mode. Error words or flags, triggered FDIR actions, often in combination with spacecraft event messages, are an indication of problems detected by the AOCS. In case of anomalies or FDIR activities, routine activities change to contingency operations (see Sect. 6.5.3.3).

Verification of Equipment Health and Performance

These more specific checks comprise on/off settings, status, and health information of sensors and actuators. Direct readings of sensors and duty cycles of the actuators, like wheel speeds, magnetic torquer activities, and thruster on-times provide a first guess of the AOCS performance.

Current Attitude

This expression means the attitude solution, as calculated by the on-board control, which respects all sensor inputs. Major deviations between calculated and expected attitude are a direct indication of attitude control problems, if not already triggered by FDIR mechanisms.

Monitoring of Orbit

For spacecraft equipped with satellite navigation systems (e.g., GPS), the current orbit measurements are monitored. Typical parameters to be checked are number

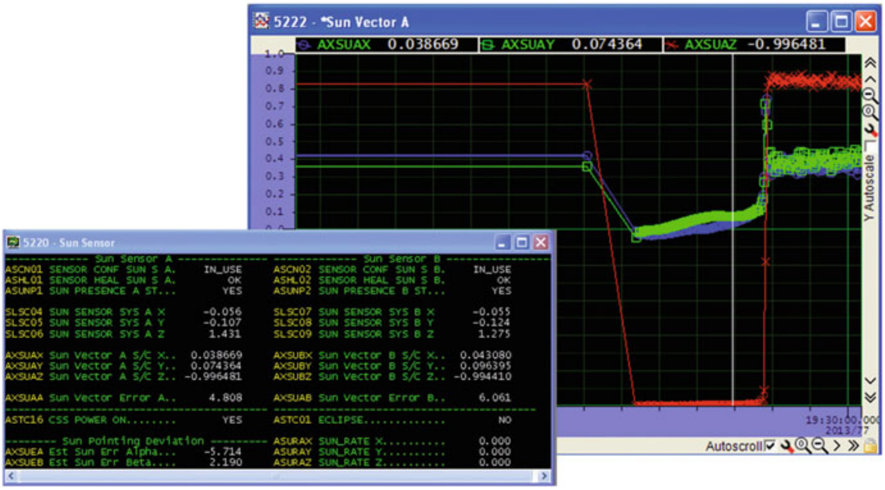

Fig. 6.45 Classic alphanumeric and graphical display pages for AOCS. Whilst alphanumeric pages only give a snapshot of a status, plots allow observing parameter behavior over time, which is most useful for dynamic AOCS parameter, like angles, rates, momentum, etc.

of tracked satellites and availability of position and velocity solutions. If on-board routines for further processing of these data are available, their functioning is observed.

These checks and analysis mentioned above are performed typically during a contact with the spacecraft by observation of incoming real-time telemetry. Main tool for real-time monitoring is the display system providing corresponding display pages. Figure 6.45 is an example of common sensor-specific telemetry.

Another valuable monitoring method is the analysis of recorded (off-line) data. Off-line checks and analysis allow monitoring of the subsystem over longer periods. In addition, long-time analysis of equipment telemetry might indicate evolving problems or anomalies (e.g., increase of wheel friction torques due to degradation or aging effects, change in data noise of mechanical gyros, etc.).

Support of Orbit and Attitude Maneuvers

The AOCS engineer supports all kinds of orbit and attitude maneuvers. For maneuver preparation (e.g., station keeping maneuver for geostationary satellites or orbit correction maneuver for LEO missions), he works in close contact with the flight dynamics engineer. During routine phases, maneuvers and corresponding command lists are mainly planned and prepared by mission planning systems or flight dynamics tools.

If real-time telemetry is available, the maneuver execution and the corresponding performance of the AOCS are monitored. Otherwise, off-line telemetry is analyzed afterwards.

Miscellaneous Activities

Depending on the mission and spacecraft design, additional tasks are in responsibility of the AOCS engineer. Some typical tasks are mentioned here:

Routine Update of Reference Data

On-board navigation systems for orbit determination sometimes require periodic updates of reference data (e.g., two-line elements) as fallback option in case of missing GPS measurements.

Sensor Interferences

For LEO missions, sensor interferences normally do not require special activities by the ground personnel, but for some GEO satellites, interferences of the earth sensor by sun or moon need to be handled. The standard measure to avoid interferences is to disable the corresponding sensor head, when sun or moon comes into its field of view. Such head switching is done either manually via time-tagged commands (in case of moon events), or automatically by on-board routines (in case of sun events). In both cases, the ground personnel monitor these events.

Eclipse Phases

As in case of sensor interferences, eclipse phases are normally only a topic for GEO missions. Some GEO satellites need to be configured for each eclipse phase and will be monitored by ground personnel during the event.

6.5.3.2 LEOP and IOT

The LEOP phase is the most intense phase for AOCS engineers. The engineer has to perform all of the tasks, as described in the basic activities section. Beyond, he is also responsible for:

First Attitude Estimation After Separation from Launcher

Receiving the first telemetry from the satellite, the AOCS engineer performs a quick subsystem checkout. LEO satellites normally invoke their attitude control shortly after board computer start-up and it depends on the first ground station visibility, if this can be monitored in real time by ground personnel. For GEO missions, even when they are put into a GTO first, real-time telemetry is often available shortly after separation. So, the AOCS engineer can check the separation attitude. The AOCS activation and all further steps are done ground controlled.

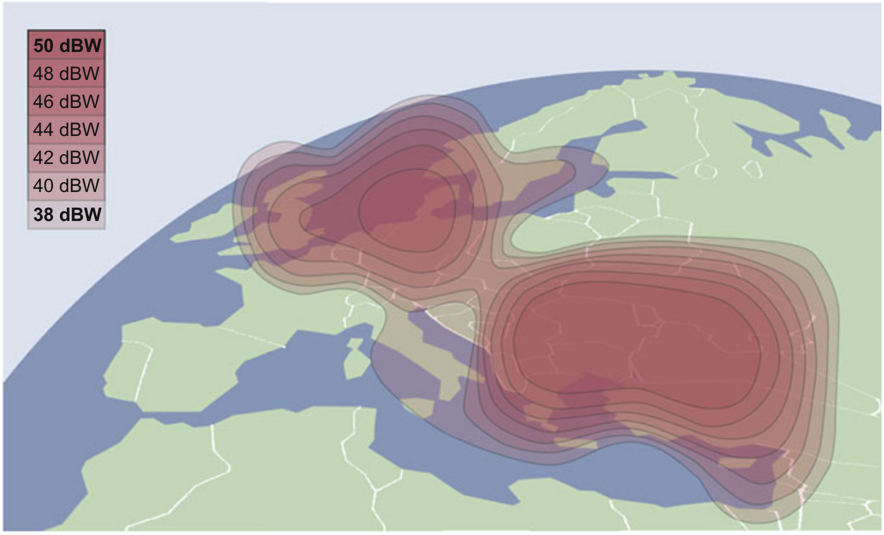

Fig. 6.46 Example of a TV satellite downlink coverage

Support of Orbit and Attitude Maneuvers

All orbit and attitude maneuvers, performed the first times, are prepared thoroughly, monitored in real time, and analyzed and calibrated afterwards. For GEO missions starting in a GTO, this comprises also the execution of apogee boost maneuvers.

In-Orbit Tests

Before the mission goes into its routine phase, the spacecraft and its payload are tested and calibrated beforehand. Tests of the AOCS comprise checkouts of sensor orientation and field-of-view measurements, calibration of sensors (e.g., drift compensation of gyroscopes), tests of redundant equipment, and actuator checkouts. It depends on the mission, what kinds of tests are foreseen. Customers who ordered a satellite delivered in space often request testing all redundant equipment (all sensors and redundant actuators) in order to get the confirmation that all components, they pay for, are available. Other missions prefer leaving redundant components off until they are really needed.

Payload Checkouts

Checkouts and IOTs are also supported by the AOCS engineer if required. Communication satellites often perform an antenna mapping during the IOT, where the ground "footprint" of the payload antennas is measured. Therefore, a single ground station is measuring the downlink power, whilst the whole GEO satellite is turned systematically by a few degrees about roll and pitch axes, covering the area of the ground footprint. Figure 6.46 shows an example of a TV satellite's footprint.

6.5.3.3 Contingency Operations

This kind of operations generally comprises the handling of abnormal behavior or malfunctions of the AOCS. Such anomalies are either detected by the spacecraft autonomously via FDIR tasks or discovered by the ground personnel. The most common tasks of the AOCS engineer in contingency situations are:

Error Detection and Analysis

Most contingency situations start with analysis of error bits/flags, event messages, and triggered FDIRs. Since the root cause of a problem is not always clear right from the start, evaluation of real-time and history telemetry might be needed. Corresponding contingency procedures should be available to support such analysis.

Reconfiguration After FDIR Cases

If a problem is sufficiently analyzed, it might be needed to reset equipment selections to nominal settings (e.g., after triggered error and automatic switch to redundancy), when corresponding equipment was declared as o.k. and operational. In addition, error counters and FDIR mechanisms are reset again.

Change to Redundant Equipment

Manual switching of sensors or actuators to the redundant equipment might be needed, when long-time analysis and specialists appraisal recommend changing it preventively, before a real problem occurs.

AOCS Recovery

In case of major anomalies, when a spacecraft was put into a safe mode, it might require extensive recovery activities, until payload operations can be resumed. Geostationary satellites are good examples for these recoveries. When they were put into a sun-pointing safe mode, it needs hours of operational work and a considerable amount of fuel, until the spacecraft is back in 3-axes stabilized orientation.

AOCS Software Maintenance

In some cases, a problem solution might include an update of the on-board software. The loading of S/W upgrades and patches itself is normally not a critical task, but the activation of it often requires full attention of the ground control team. Typically, some follow-on testing sessions are mandatory.

6.5.3.4 Support Tools for AOCS

For AOCS flight operations, a multiplicity of tools and software modules might be developed in order to support the engineer's daily business. The scope of these tools ranges from simple tools for quick solutions or estimations to complex and fully developed software systems. The technical realization is depending on the mission, spacecraft design aspects, and operational needs and constraints. Some typical AOCS tools are listed here:

Reference Parameter Calculation for Attitude Maneuvers

These tools are used to calculate parameter for upcoming attitude maneuvers, which are input for corresponding flight procedures. A typical example for GEO missions is the transition from sun pointing to earth pointing orientation. Therefore, some simplified orbit parameters for the epoch of the planned mode transition are calculated on ground and then sent to the satellite. For some LEO mission, lists of attitude quaternions for a time period are calculated to provide a dedicated attitude profile.

Gyro Calibration

For the drift estimation and compensation of gyroscopes, a ground calculation might be needed. Therefore, gyro measurements over a corresponding period are collected and a drift is estimated on ground.

Attitude Determination

A classic application for AOCS engineers are attitude determination tools, which either allow a rough order of magnitude estimation of the current spacecraft orientation or perform a high accuracy attitude determination by processing larger amount of telemetry data.

6.5.4 Experience from Previous Missions

This section illustrates some examples of an AOCS engineer's experience.

6.5.4.1 AOCS Degradation

The very successful multinational (USA, UK, Germany) ROSAT project is a good example, how long successful mission operations could be conducted in spite of

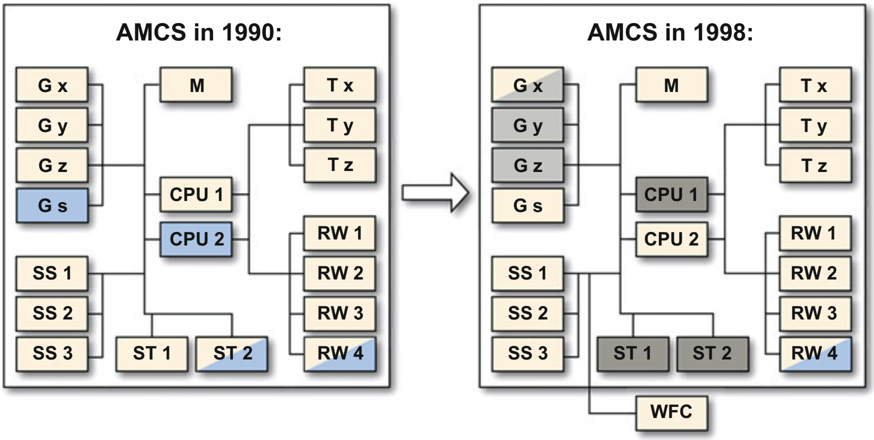

Fig. 6.47 Layout of the ROSAT AMCS in 1990 and 1998. At launch in 1990, the AMCS was equipped with two board computers (CPU), four gyroscopes (G), a magnetometer (M), three sun sensor heads (SS), two star tracker cameras (ST), three magnetic torquer (T), and four reaction wheels (RW). ROSAT had no propulsion system. Until end of the mission, some equipment was lost. In addition, a part of the payload, a star tracker of the Wide Field Camera (WFC), could be used successfully to replace the lost AMCS star trackers. Legend: prime equipment (*light grey*), cold redundant (*dark grey*), hot redundant (*light grey/dark grey*)

degradation and loss of equipment. Figure 6.47 shows the layout of the attitude measurement and control system (AMCS) at begin of the mission and at the end.

After a short LEOP and commissioning phase, the mission started in 1990 with a systematic mapping of the sky for X-ray sources, followed by a pointing phase where selected objects were observed. Over the years, some sensors degraded and failed at the end, but all of these losses could be managed by updating and adjusting the AMCS software. Even after loss of both star tracker cameras, the mission was not over. It could be realized to bring a part of the payload instruments into the attitude control loop. Therefore, also parts of the satellite OBDH needed to be changed. After implementation of the change, some successful data takes could be done. Of course, the performance of the system was not as good as after launch, but it could be impressively demonstrated, how flexible both satellite configurations and ground operations could be adapted to encounter problems.

6.5.4.2 Strange Wheel Behavior

This example illustrates how environmental effects might influence the attitude control. This situation occurred during the positioning of a geostationary communication satellite. At the last day of the LEOP, all daily activities had been performed, which included the last apogee motor firing, the solar array full deployment, the deployment of both antenna dishes, and the run-up of the fly-wheel a

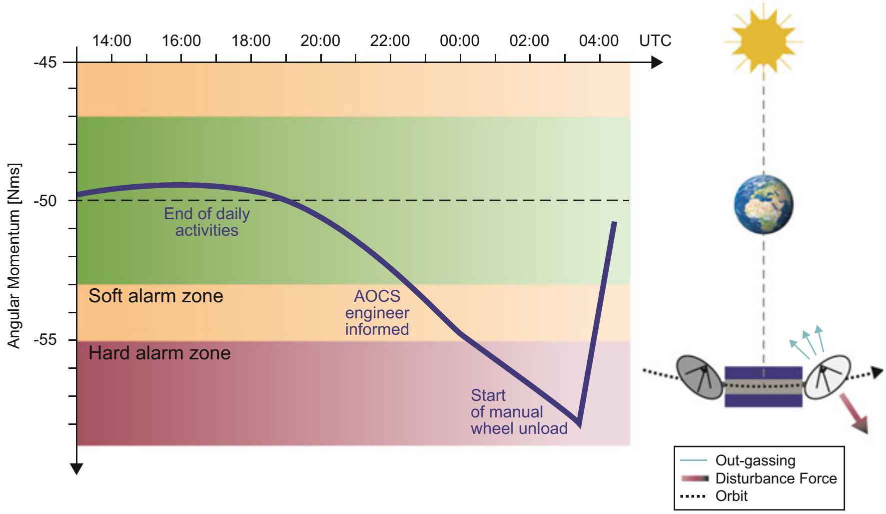

Fig. 6.48 Strange wheel behavior due to outgassing effects. The *left figure* shows the angular momentum telemetry of the pitch axis. The *colored zones* are marking the soft and hard alarm zones. The *right figure* is a view on orbital plane of the satellite. The sun (*yellow*) was shining directly into the dishes, which generated a weak disturbance torque by outgassing

reaction wheels. The satellite was in a nominal configuration in the afternoon on that day. The angular momentum, stored in the fly-wheel, was nominally at around 50 Nms, but through the upcoming night, it developed unexpectedly, as shown in Fig. 6.48. The angular momentum and the speed of the fly-wheel were steadily increasing. Analysis of the AOCS and other subsystems did not lead to clear explanations. The effect was indicating that the attitude control was working against a weak, but consistent disturbance torque about the spacecraft pitch axis (north–south axis). Fuel leakages, which might cause such an effect, could not be identified.

The satellite had a corresponding FDIR mechanism, which would have reacted on this by performing an autonomous wheel unload maneuver, but in order not to interfere the next day's flight operations activities, it was decided to perform a manual wheel unload procedure. During the night, the operations team performed it and the wheel speed was set back to a nominal value. Thus, the accumulated momentum could be removed, but the root cause was not yet understood.

The explanation was provided by the satellite's manufacturer on the next morning. The satellite was equipped with two big payload dishes about the same size, but made of different material. One dish was made metal, but the other one consisted of composite, which tended to soak water from the air humidity at the launch site. After the deployment of the dishes, the sun was shining into them for the first time since launch and heated them up correspondingly. As a consequence, the water was evaporating from the dishes, a so-called outgassing (see Chap. 1). This created a weak but constant pressure on the dish. The disturbance torque was

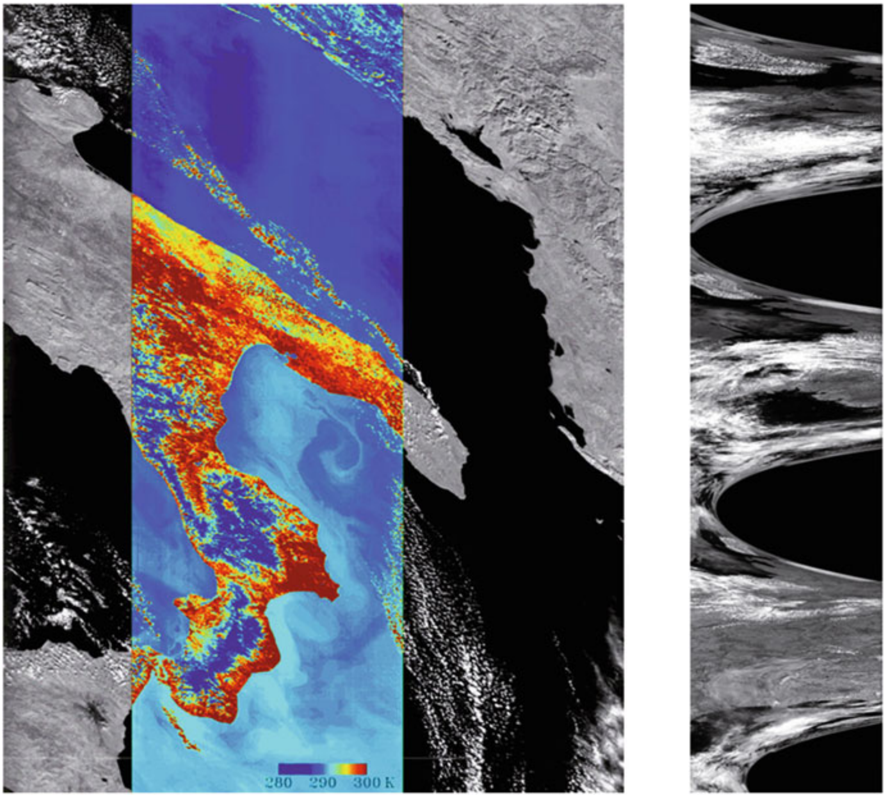

Fig. 6.49 *Left picture*: Early BIRD data take showing a merged infrared picture of southern Italy. BIRD was doing the data takes with single line camera detectors. By keeping a stable attitude, the camera detectors were scanning the earth over 6–8 min and generating such pictures. *Right pictures*: After loss of some ACS equipment, a test data take is shown. The spacecraft had stabilized in a high spin cone like motion, which caused a quite artificial photo of the earth

compensated by increasing the fly-wheel speed. So, there was no leakage and no AOCS failure, everything went fine. The effect was a natural one and disappeared after 2 days.

6.5.4.3 Undocumented AOCS Feature

BIRD was a new approach in the design of a small satellite mission dedicated to hot spot detection (forest fires, volcanic activities, burning oil wells, or coal seams), and evaluation. Payload and satellite were designed, built, and operated by DLR. The subsystem components for BIRD were chosen with respect to a low-cost mission for a planned lifetime of 1 year. The Attitude Control Subsystem ACS consisted of sun sensors, gyroscopes, magnetometer, star sensors, magnetic torquer, and reaction wheels. BIRD had only sun-pointing modes, which could be biased to realize a nadir orientation during the data takes.

The mission started in 2001 and could be operated successfully for 2 years, where pictures of the earth in different wave lengths could be gathered. One of the first shots is shown in Fig. 6.49 (left picture). In 2003, the gyro package failed and in the following hours, three of four reaction wheels were destroyed by overstraining, before the ground team had a chance to react. The contact with BIRD was lost for 5 days. It looked like the mission was over, but on day 6, telemetry could be received again. First checks of the ACS showed BIRD in a stabile orientation, spinning with around the sun direction with 4°/s and an offset angle of 40°.

A first test of a payload camera after this incident is shown in Fig. 6.49 (right picture).

This strange mode remained stable over weeks, but it was never designed so. Nevertheless, it gave the ground team enough time to update the ACS software and to proceed with degraded BIRD operations.

6.5.5 Summary

The AOCS is one of the most complex subsystems of a spacecraft. Therefore, the ground operations team and the AOCS engineers have to perform various tasks, ranging from simple subsystem monitoring to intense maneuver preparations and executions.

It is expected that future spacecraft will have more autonomy. More and more, higher control functions, like orbit maneuver planning tasks, can be implemented in the on-board control logic, e.g., the Autonomous Formation Control TAFF on board of the satellites TerraSAR-X and TanDEM-X. Another subject is the use of artificial intelligence AI methods. Precursor missions (e.g., NASA's Deep Space 1) have already been flown.

6.6 Repeater Operations

Jürgen Letschnik

The Sputnik 1 is well known as the first satellite, launched on October 4, 1957. It was equipped with an on-board radio-transmitter that worked on two frequencies: 20,005 and 40,002 MHz. The first satellite which could be used for communication

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Fig. 6.50 Basic functions of the repeater payload

activities was the project SCORE in 1958 of the Americas. This satellite was equipped with a tape recorder to store and forward voice messages and it was used to send a Christmas greeting to the world from U.S. President Dwight D. Eisenhower. After that mission, NASA launched the satellite Echo 1 in 1960. The satellite consisted only of a 30.5 m diameter metalized polyethylene film balloon with a thickness of 12.7 μm. It was used to redirect transcontinental and intercontinental communication like telephone, radio, and television (Wikipedia http://en.wikipedia.org/wiki/Sputnik_1 2014).

Telstar 1 was launched in July 1962 into an elliptical orbit. It was the first active, direct relay communications satellite and the era of a telecommunication got its kickoff. Telstar successfully relayed television pictures, telephone calls, and fax images through space and provided the first live transatlantic television feed (Wikipedia http://en.wikipedia.org/wiki/Telstar 2014).

Syncom 1, launched on February 14, 1963 with the Delta B #16, was the first satellite in a geosynchronous orbit. Seconds after the apogee kick motor for circularizing the orbit was fired, the spacecraft fell silent which led to an electronics failure and therefore the satellite was lost. By using telescopic observation measurements, it could be verified that the satellite was in an orbit with a period of almost 24 h at a 33° inclination (Wikipedia http://en.wikipedia.org/wiki/Syncom 2014).

Then on July 26, 1963, the successor Syncom 2 became the first successful geosynchronous communication satellite. During the first year of Syncom 2 operations, NASA conducted voice, teletype, and facsimile tests, as well as 110 public demonstrations to show the capabilities of this satellite and invite feedback. In August 1963, President John F. Kennedy in Washington, DC, telephoned Nigerian Prime Minister Abubakar Balewa aboard USNS Kingsport docked in Lagos Harbor: the first live two-way call between heads of state by satellite. The Kingsport acted as a control station and uplink station (Wikipedia http://en.wikipedia.org/wiki/Syncom 2014).

6.6.1 Repeater Subsystem

This chapter is devoted to a description of the satellite communication payload (repeater) with the emphasis on design principles, characteristic parameters, and the technologies used for the equipment.

On most commercial communication satellites the payload consists of two distinct parts with well-defined interfaces—the repeaters and the antennas (Berlin 2005; Bostian-Allnutt 2003).

The repeater usually encompasses several channels (also called transponders) which are individually dedicated to sub-bands within the overall payload frequency band. The generic functionality and characteristic parameters of a repeater payload are presented in the following.

6.6.1.1 Functions of a Repeater Payload

The main functions of the communications repeater payload can be split up in four major functionalities; see Fig. 6.50. A detailed description of each functionality will be given in the upcoming paragraphs.

Receive The repeater will get a signal which is received from the antenna in a given frequency band and with a given polarization, from one or more earth stations. The stations are situated within a given region (service zone) on earth and are seen from the satellite within an angle that determines the necessary angular width of the satellite antenna beam. The intersection of the satellite antenna beam with the surface of the earth defines the receive coverage.

Amplify The signal strength received from the earth station via the antenna system of the satellite is extremely low, around -190 dBW (equalling 10^{-19} W). A direct reflection of the received signal like it was done on ECHO 1 would lead to a total loss of the whole signal. It is therefore mandatory to amplify the signal on board of the spacecraft to get a useable signal at the receiving ground terminal. To maximize the level of amplification without generating any distortion is therefore the aspired goal.

During the amplification process, the frequency from the received signal will also be converted to a different frequency to prevent disturbances between the up- and the downlink signal. At the moment, a typical and commonly used frequency range is the Ku-Band, using the 14 GHz band for uplink and the 11–12 GHz band for downlink signals (Handbook on Satellite Communications 2002; Maral and Bousquet 2002).

Route Modern satellite systems are able to route signals between a variety of receive and transmit antenna systems on the satellite. The routing can take place on radio frequency (RF) level or on the base band level of the signal (with or without the carrier frequency as described in Sects. 1.3.2 and 1.3.3).

If the received signals are demodulated to base band level on the satellite, the repeaters are called *regenerative repeaters*. Based on the on-board routing configuration the data streams can be routed to different spot beams. For transmitting the signal, the stream will be modulated on an RF carrier again. Such systems are already available, for example, on EUTELSAT 13A (HB6), EUTELSAT 7A (W3A), and on several other satellites.

If the received RF signal is routed without any demodulation or modulation process, the repeater is called *transparent repeater*.

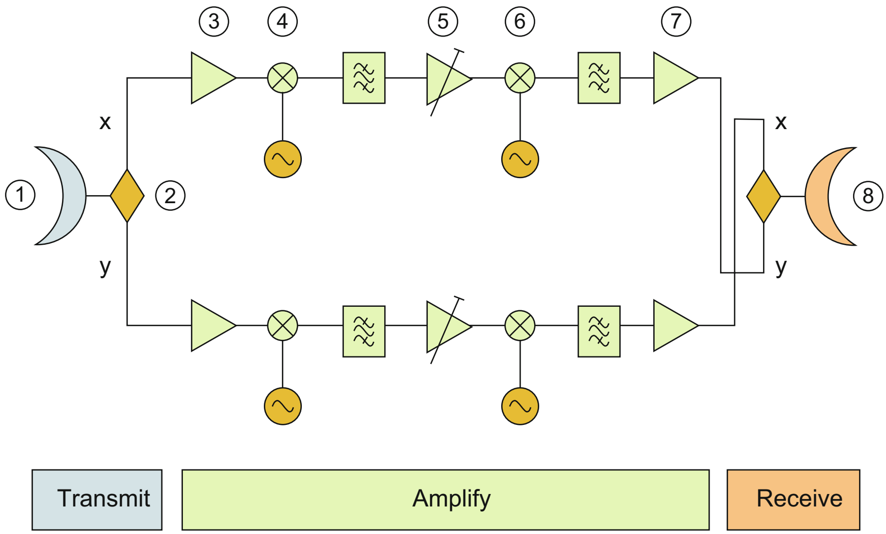

Fig. 6.51 Simplified transponder/repeater layout

Transmit The amplified and possibly also regenerated signal will now be transmitted via a transmit antenna system. Such antenna systems can be realized as single beam or multi-beam antennas depending on the application scenarios of the satellite.

6.6.1.2 Overview and Layout of a Repeater/Transponder

One of the simplest transponder configurations (without a routing functionality) of a dual polarization transponder [HLP—Horizontal Linear Polarization (X), VLP—Vertical Linear Polarization (Y)] can be seen in Fig. 6.51 and refers to the addressed functions above. The key equipment of this simple transponder will be described in this paragraph.

1. *Receive Antenna* The typical realization of a spacecraft antenna is the parabolic dish especially for higher frequencies (e.g., X-Band, Ku-Band to Ka-Band). For lower frequencies like P-Band (400 MHz) Helix antennas will be used. Sizes of such antennas are in the range from around 1 m up to 8 m, e.g., EUTELSAT 10A (W2A) is using an 8 m deployable antenna in S-Band).

   The antenna system can be realized as fix-mounted antennas or as steerable antennas or as a combination of both, depending on the services of the satellite provider. The usage of multi-feed antennas is becoming more and more popular; their high flux density and the possibility of frequency reuse are the driving arguments.
2. *OMT* The OMT (Orthomode Transducer) is an important element of a satellite transponder system. The OMT is a wave-guide element used to split up the

received polarizations HLP and VLP into two separate paths. The usage of different polarizations on the same frequency allows an increase of transmitted information capacity. A detailed description will be given in paragraph 8.

3. *LNA* LNA stands for Low Noise Amplifier. As the name implies, this device is characterized by having a very low noise level while providing a high-gain level. In all cases, even with high-gain antennas, an LNA is necessary to get a good signal-to-noise ratio of the received signal.

   The LNA is a broadband RF equipment and therefore able to amplify the whole receive spectrum of a dedicated frequency band (e.g., 27.5–30.0 GHz, Uplink Ka-Band). In a typical communications satellite system it is followed in the signal path by an input multiplexer that separates the channels.
4. *Downconverter* The frequency of the received signal will be reduced to an intermediate frequency (IF) in order to reduce the losses and to make components simpler in the components following in the signal chain. On the ESA satellite ARTEMIS for example the IF is 5 GHz.
5. *Channel Amplifier* The channel signal will then again be amplified with an adjustable channel amplifier (CAMP). Depending on the detailed design of the satellite it is possible to adjust the gain of each amplifier individually. The performance of the overall link (ground–satellite–ground) is controlled with this setting. Some satellites also have a so-called ALC amplifier (ALC—Automatic Level Control). This module adjusts the gain of the channel amplifier as a function of the signal input level coming from the antenna.
6. *Upconverter* The IF that was used for amplification and routing will then be transformed to the higher frequency of the downlink. That frequency is coordinated with the ITU in the design phase to the satellite. Typically the D/L frequency band is on a lower frequency than the U/L frequency band. Filters located after the converter modules eliminate frequencies that are generated by the mixing process itself.
7. *High-Power Amplifier* The high-power amplifier (HPA) plays an important role on a communication satellite system. To get an adequate signal level on ground it is necessary to generate a high RF (radio frequency) power level at the output of the satellite. Generating a signal with a high RF output level leads to a high electrical power consumption which has to be provided by the satellite platform. Traveling wave tube amplifiers (TWTA) are the most common used amplifiers on communication satellites. An efficiency of about 60 % can be expected and the higher bandwidth compared to solid-state power amplifiers (SSPA), efficiency of 10–20 %, is a positive effect. The disadvantage of a TWTA can be seen in its nonlinearity transfer curve, which plays a role later on in the operations aspect.
8. *Transmit Antenna* The high-power RF signal will now be guided to the transmit antenna. Hereby the polarizations are swapped in comparison to the receive antenna system to maximize the decoupling of up- and downlink signal. Therefore the received X-polarization becomes the Y-polarization in the downlink, and vice versa the Y-polarization in the uplink becomes an X-polarization in the downlink. Figure 6.52 shows an illustrated example of the principle.

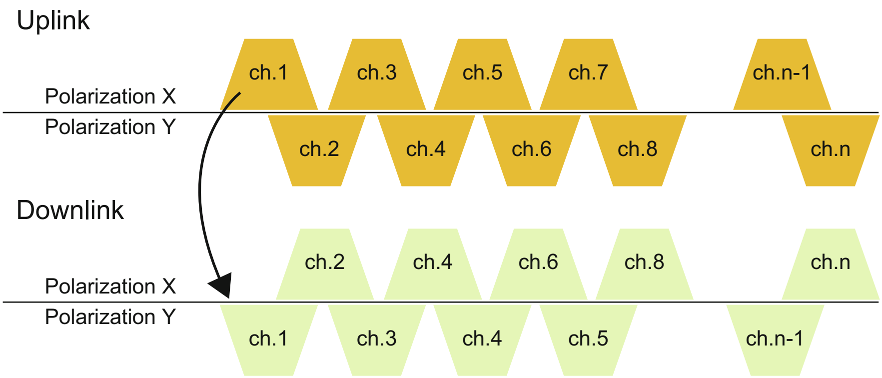

Fig. 6.52 Simplified illustration of a U/L and D/L channel allocation on a repeater subsystem for a communication satellite

6.6.2 Repeater Operations

The general environment to operate the repeaters of a geostationary communication satellite is very similar to other space missions. Infrastructure elements like ground station, spacecraft control center, and the data network between both are necessary. The main difference is the necessity of a ground station with equipment to measure the in-orbit performance of the repeater payload. Such a ground station is called IOT station. It is equipped with special RF (radio frequency) components, like power sensors, inject-pilot systems, and an ASSC (automatic satellite saturation control) unit, to guarantee high precision measurements of the repeater subsystem.

Repeater operations in general can be split as shown in Fig. 6.53 in communications operations and spacecraft operations subdivided in mission phases of LEOP, IOT, and routine and contingency operations.

Communication Operations The most well-known form of satellite communication is the direct-to-home broadcast of television programs via satellite. Here, the payload of the satellite (repeater) is used to spread television programs over a larger area. The satellite is only one element in the overall communication path. The care for the spacecraft activities is not in the remit of the user, because the connection time is often rented only. Typical users e.g. Play out Centers of TV stations or Satellite News Gathering (SNG) stations are coordinated and instructed by the Communication Service Center (CSC) of the satellite provider e.g. SES ASTRA, EUTELSAT, INTELSAT in configuring the uplink channels to the satellite. The user itself has no access to the repeater configuration on board.

Spacecraft Operations The housekeeping management of communications satellites as described here is sometimes done by a separate team. The main task is to guarantee a continuous monitoring of all important parameters (housekeeping data)

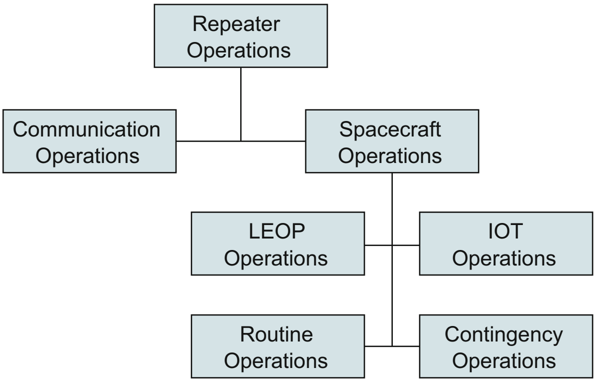

Fig. 6.53 Illustration of the structured repeater operations phases

of the satellite which are required for proper operating of the repeater subsystem. Typical satellite parameters to be monitored will be discussed in detail in the following chapters.

6.6.2.1 LEOP Operations

During the LEOP operations of a communication satellite, the main operations focus is on getting the satellite on station and on operating the platform subsystems. Only few tasks are forseen for payload operations during that period. A typical repeater operations task is the deployment of the antenna systems. Mostly the antennas have to be in a so-called "stow position" during the launch. In many cases they are folded down because of the limited space in the shroud of the launch vehicle and to protect the antenna structure against damage due to vibrations during the launch.

Before and during the antenna deployment, engineers have to carefully monitor the temperatures of the hold-down mechanism pyros, motor currents of the antenna mechanism, the temperatures of the RF cables, and other parameters defined by the satellite manufacturer.

6.6.2.2 IOT Operations

Besides the IOT of the satellite platform, the payload IOT phase can be seen as a single phase in itself, because of its duration and complexity. The repeater subsystem is the main focus in this time. From the customer perspective, the communications payload is the most important satellite equipment. Money shall be earned by providing a guaranteed, defined service to end users.

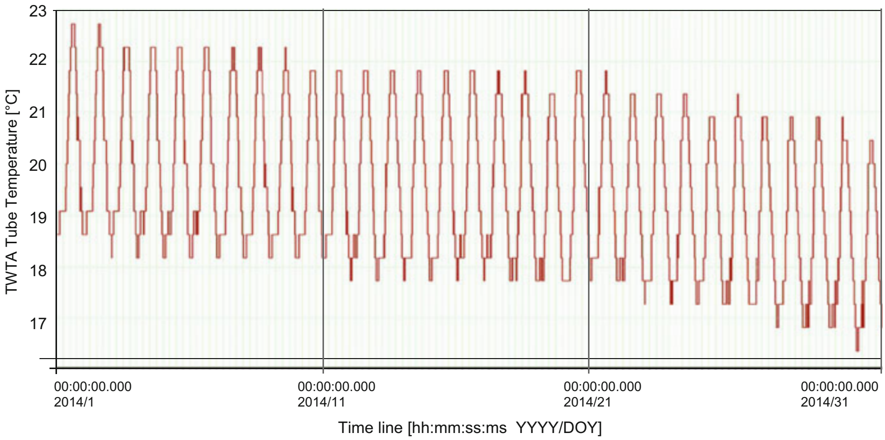

Fig. 6.54 Monitoring the tube temperature of a TWTA during 1 month

The focus and the sensibility to perform a correct and precise IOT of the repeater subsystem is very high on the customer and satellite manufacturer side. To manage a payload IOT campaign, all platform operations activities on the satellite have to be coordinated with the RF measurement campaign, e.g., no orbital maneuvers shall take place in that time. Typically two groups of engineers are involved in this process: On the one hand the IOT ground station test engineers, mostly situated at the ground station facility, and on the other hand the spacecraft operations engineers located at the spacecraft control center (SCC). The baseline for both groups is the IOT test plan for the repeater subsystem. This plan includes a schedule of all activities which will be done during the test campaign. All satellite configurations and required changes are part of this plan. The main exercise for the SCC group is to translate these configuration requirements into operational flight procedures. Depending on the complexity of the repeater subsystem, the IOT typically takes approximately 2–3 weeks.

6.6.2.3 Routine Operations

Routine Operations of the repeater subsystem are driven by the needs of the customers who are using the repeaters for their data transmissions, an activity with low working load on the payload subsystem engineers and operators. Their main activities in this phase are monitoring and observation. Figure 6.54 shows a typical telemetry plot (tube temperature of a TWTA) from a commercial repeater payload over a period of one month, which should be monitored continuously.

In nominal and routine operations the repeater itself does not need elaborate analysis of the telemetry. Typical activities in routine operations on the repeater subsystem are:

- Configuring the on-board routing between different channels and different antenna spot beams
- Change the pointing of the antenna reflectors
- Adjust the gain step settings of the channel amplifier
- Change the conversion frequency of the on-board up-/down converters
- Operations with strict observance of "Repeater Flight Rules"

On-board Routing The functionality to route received channels to different transmit channels is one of the most interesting activities on a telecommunication satellite. One technical example of a routing base system would be the SKYPLEX technology developed by ESA and EUTELSAT. This system allows to split and recombine DVB channels transmitted from different ground stations to one signal DVB stream on board of the satellite.

Antenna Pointing Some satellites are equipped with steerable antenna systems. With this antenna system it is possible to readjust the pointing of the generated spot beam on earth. This functionality increases the flexibility in providing communication link services, especially point-to-point connectivity services.

The adjustment of a mechanical system in space is always attended by a higher risk compared to electrically based adjustments. The manufacturer of the antenna pointing mechanism and the electronics typically defines a number of parameters which have to be within the nominal range, e.g., temperature of motors and the mechanism, to be ready for the adjustments. If heaters have to be switched ON to reach this nominal range, it could take a quite long time to reach the necessary values. This leads to a pre-adjustment period that has to be taken into account in the planning.

Gain Step Adjustment Channel amplifiers as described in the first chapter are normally equipped with the functionality of gain adjustment. This helps to optimize the communication link for one or more customers. Each channel amplifier can be adjusted separately, which leads to a high number of gain step values which have to be monitored by the SCC. In addition to the gain steps of the channel amplifiers, the HPAs can also be equipped with a gain step adjustment functionality, which results in a lot of additional parameters to be monitored.

Modern amplifier modules are very resistant again cosmic radiation; nevertheless it can happen that a gain steps setting "flips" in its value, which has a direct and important impact on the communication link, and therefore needs the immediate attention of the engineers in the SCC, especially for the re-establishment of the correct gain step value.

Conversion Frequency Some satellites open the possibility to change the conversion frequency of the up- and down converter. The important thing when changing this frequency is to be aware of the fact that all frequencies are coordinated for each geostationary communication satellite via the ITU. A misadjustment of the frequency can lead to interferences and disturbances on frequency bands that are used by other users.

Table 6.8 Overview of typical flight rules including the recommended operational activities for a repeater subsystem

Rule	Recommendation or activity
Switching of coaxial and wave-guide switches	Do not switch a coaxial or wave-guide switch during an RF signal is ON, no "hot-switching"
Helix current of the Traveling Wave Tube Amplifier (TWTA)	Permanent monitoring of the helix current of the Traveling Wave Tube Amplifier (TWTA). A current out of the nominal range could be an indicator of a faulty tube or faulty power supply. Initiate an error detection and backup switchover sequence
Monitoring of isolator temperatures	A temperature out of the nominal range could be an indicator of abnormal RF reflection from a high-power equipment. Initiate an error detection and backup switchover sequence
Preheating phases	Monitoring of the preheating phases of equipment's like converters, amplifiers which could take from 2 to 20 min. No impact on spacecraft operations

Repeater Flight Rules Flight Rules are generalized regulations and recommendations on how to operate the satellite. This is in contrast to the flight procedures that are very specific. An overview of typical repeater flight rules is shown in Table 6.8.

6.6.2.4 Contingency Operations

In general, contingency operations are done hand in hand with the satellite or payload manufacturer. Other very important stakeholders are the end users of the repeater service who are transmitting their signals via the satellite. It is highly recommended to involve that group in contingency operations on the repeater subsystem.

Typically all these communication business services are managed, as discussed above in "Communication Operations," by those special departments, "Communication Management Center" (CMC) or "CSC."

A lot of contingency cases have already been investigated by the manufacturer before the launch. Therefore flight procedures to quickly recover the system are provided. While the execution of the recovery might only take a few hours, nevertheless the whole process, starting from error detection and analysis, can take several days.

An example of a typical redundancy switching of an HPA module (TWTA) is described in the following. Based on a drop of the anode voltage (see Fig. 6.55), the administrative and technical process with all stakeholders is triggered. Investigations and analysis lead to the decision to execute the switch-over to the redundant backup high-power amplifier module. In Fig. 6.56 both configurations before and

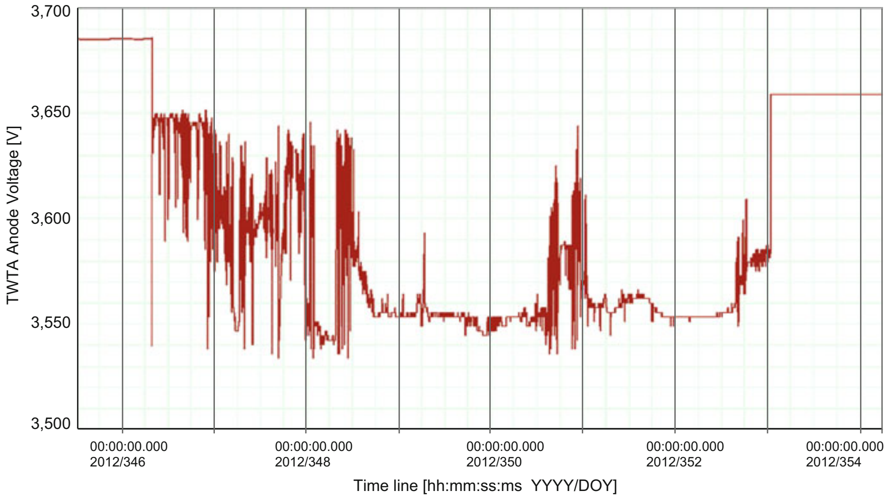

Fig. 6.55 Anode voltage drop of an traveling wave tube and self-recovery

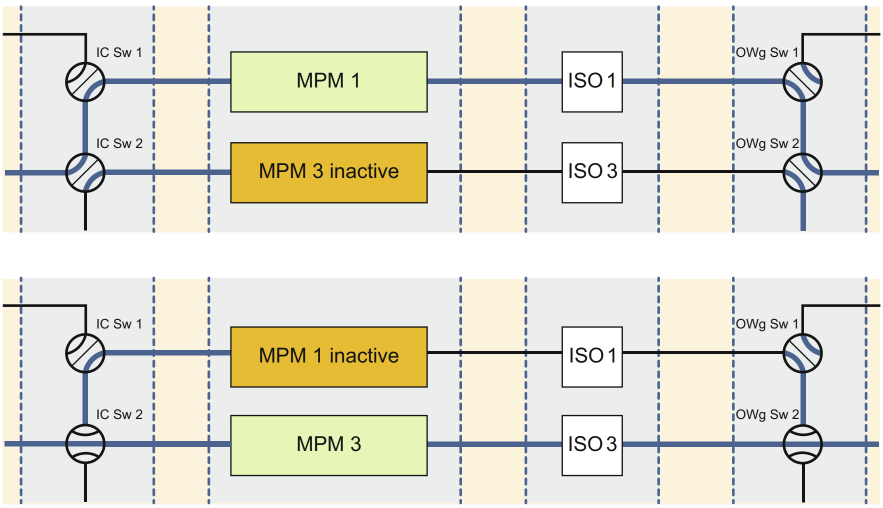

Fig. 6.56 (**a**) Configuration of the nominal repeater configuration. (**b**) Configuration of the repeater subsystem after the backup switch over

after the backup reconfiguration are illustrated. For this reconfiguration the following equipments have to be commanded:

- Power OFF of nominal high-power amplifier module (MPM 1)
- Input coax switch 2 (IC Sw 2) from Position 2 to Position 3

- Output wave-guide switch (OWg SW 2) Position 1 to Position 4
- Power ON of redundant high-power amplifier module (MPM 3)
- Readjustment of the gain steps of the HPA module

All activities have to be performed with respect to the flight rules and are based on the flight procedures which have been delivered by the manufacturer. They have to be coordinated with CSC or CMC to take care of involved communication services. This switch-over leads on the one hand to a total drop of all communication carriers of this faulty amplifier, but reestablishes on the other hand the required QOS.

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Chapter 7
Special Topics

7.1 Human Spaceflight Operations

Jérôme Campan, Thomas Uhlig, and Dennis Herrmann

7.1.1 Introduction

Human Spaceflight operations all started with Yuri Gagarin—he was the very first human in history heading towards space with his famous word "Поехали!" (Let's go!). Since that historic moment with the start of the Vostok program all the way to the current International Space Station (ISS) program and through several missions during the last decades such as Gemini, Mercury, Apollo, MIR, and STS Space Shuttle, Human Spaceflight operations concept has evolved to become what we know today. This chapter introduces the concept for Human Spaceflight operations currently in use for the International Space Station Program which gathers, as an international project, all the experience made in Human Spaceflight Operations so far.

Currently between three and six persons are traveling at an average speed of 27,000 km/h at an altitude of 400 km, in order to perform science in a microgravity environment. From an operational perspective, it is interesting to ask how manned space missions differ from unmanned ones and which assets an astronaut can bring to space missions.

This chapter focuses on the deltas to what has already been presented in previous chapters with regards to unmanned missions. It will lay out what is so special about Human Spaceflight and what are the big challenges, not just operating a vehicle in

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which astronauts are living, but driving an entire mission to its success. Special processes, functions, and behaviors are required to successfully complete such a manned mission and get the astronauts safely back on Earth.

7.1.2 Manned and Unmanned Missions

With the development of the technology, the intention is to automate everything as much as possible. Systems have to be flawless, more efficient, less time consuming, cheaper, always giving the expected outcome. However, with the actual technological state, some tasks which shall be performed in space cannot be automated. As a consequence, there are two kinds of space missions, the unmanned ones going where the human might never be able to go to and manned ones performing tasks only a human can do. Whenever an activity is too complex so that current technology cannot achieve the objective within an accepted probability, having a crew member involved makes these tasks doable. This is especially true for troubleshooting activities, reaction to the unexpected, and implementation of workarounds.

7.1.3 From a Satellite to a Living Place

7.1.3.1 Enhanced Satellite Subsystems

All the subsystems which were presented in previous chapters for unmanned vehicles are also used for manned missions but with additional features, which are briefly introduced below (Figs. 7.1 and 7.3).

Thermal Control Subsystem

This subsystem does not only provide thermal cooling to instruments and other components but also needs to provide a heat sink for the conditioning of the cabin air. The metabolic heat produced by human beings on board is putting an additional load onto this subsystem. The allowed temperature range for equipment accessible by the crew is also limited to prevent touch temperature hazards for the astronauts.

Electrical Power Distribution Subsystem

Besides generating and distributing power for the other subsystems, electric current is also required to provide light within the space station and also energy for the various electronic devices astronauts operate on board. Having electrical current in an oxygen enriched environment also constitutes a fire hazard, which is not present

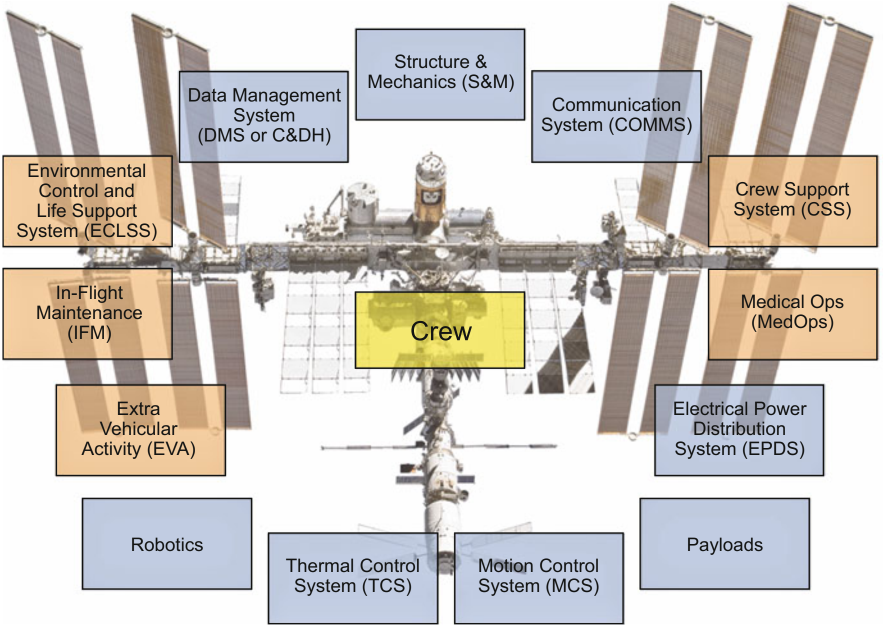

Fig. 7.1 Not only the conventional subsystems of a satellite (*blue*) need to be adapted but also new subsystems need to be added (*orange*) to allow the presence of human beings (*yellow*) on board of a spacecraft

in an unmanned vehicle. Open electrical connectors are an electrical shock hazard for the crew itself.

Data Management Subsystem

For crew autonomy purposes, a manned spacecraft is not always only being commanded by a Flight Control Team (FCT) on ground, but can also be controlled by the astronauts on board, especially when it comes to critical functions. Therefore the Data Management System needs to provide a Human Machine Interface (HMI), which allows the crew to send commands to the vehicle as well as to check the essential data of the subsystems. On the ISS this is mainly implemented via laptops with dedicated software, which offer a graphical representation of the space station and allows insight in ISS data as well as synoptic commanding.

Communications Subsystem

In Human Spaceflight, the communication subsystem is not limited to "telemetry and telecommand" transmission but it also includes visual (video/image), verbal, and written communications.

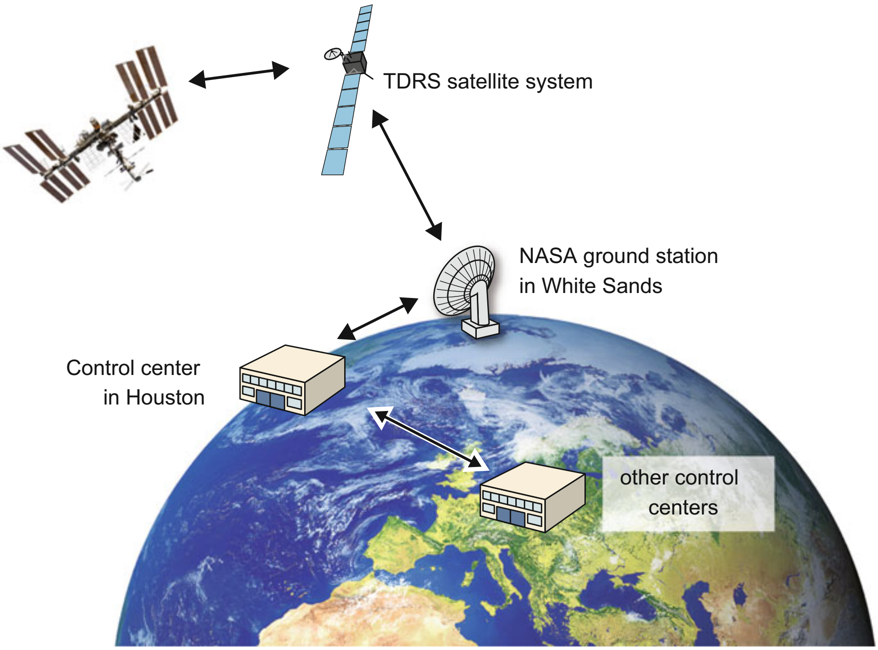

Fig. 7.2 The space station is connected via the Tracking and Data Relay Satellite system (TDRSS), which uses one ground station in White Sands/New Mexico with the Mission Control Center in Houston. All other control centers are connected to Houston

Each of the ISS control centers whether it is in Houston, Huntsville, Tsukuba, Moscow, or Munich has the capability to interact with the crew on board the ISS using this voice communication link commonly called Space to Ground (S/G).

For this link between ground and the ISS, the Tracking and Data Relay Satellite System (TDRSS), which allows an almost continuous contact, is used (Fig. 7.2).

Motion Control Subsystem

Attitude and orbit control are required for a manned spacecraft as well and does not differ from an unmanned vehicle. However, in order to ensure that the Motion Control System is available for years, as needed for the ISS, it is required to resupply consumable resources such as propellant. For the ISS, the reboosts used to counteract the continuous altitude decay due to breaking forces of the remaining atmosphere are usually performed by docked spacecrafts, as described in more detail in Sect. 7.1.3.3.

7.1.3.2 Environment Control and Life Support System

Not only modifications to the already introduced subsystems common for a satellite are required, but also at least one additional subsystem needs to be introduced to be able to put humans into space. A pressurized, habitable environment which can ensure life on board needs to be provided. Such functions are established by the Environment Control and Life Support System (ECLSS).

The ECLSS is a complex subsystem and is dedicated to ensure and protect life on board. It is comprised of the following primary functions.

Atmosphere Control and Supply

Cabin air—a combination of oxygen and nitrogen—is provided by the atmosphere control and supply (ACS). It ensures both components are supplied at the correct pressure and ratio so the astronaut can normally breathe on board without additional equipment. The ACS provides pressure equalization between the different modules as well as depressurization. Depressurization is considered as ultima ratio to extinguish a fire or to dump toxic atmosphere into space. In such situation the crew would be protected by closing relevant hatches in order to isolate the segment requiring the depressurization.

Atmosphere Revitalization

Whenever the astronauts are breathing they are producing carbon dioxide. The AR ensures the removal of CO_2 and allows to trace contaminants produced by the outgassing of the different materials on board as well as to continuously analyze the constituents of the ISS atmosphere. Due to its central function the AR subsystem is built up in a highly redundant way and various methods of CO_2 removal are used to ensure independency in failure situations.

Temperature and Humidity Control

The Temperature and Humidity Control (THC) has the goal to maintain the so-called “crew comfort box” defined by a parameter range of temperature (18–27 °C), humidity (25–70 %), and dew point (6–18 °C). The values in brackets are the ones used for the European module Columbus. The presence of humans in the space station leads to an increase of the humidity level in the cabin air and to an increase of temperature due to metabolic heat. The THC has therefore to remove humidity and regulate the temperature. For that dedicated air conditioning devices are used, in which the air is first cooled down below the dew point and hence humidity is removed via condensation. As a second step the cooled air is mixed with warm air to reach the desired cabin temperature. For the cooling of the air, the subsystem interfaces with the Thermal Control System (TCS), which provides the cool surfaces and removes the heat via the cooling water loop.

Water Recovery and Management

Water management is another central function of a space station, since human beings consume and produce water, which consequently needs to be provided as well as collected and processed, so it can be reused afterwards. This is the function of the Water Recovery and Management (WRM) subsystem. Water is collected from several sources such as the condense water from the cabin air, waste water from the urine, or return water after Extra Vehicular Activity (EVA) activities. This water is processed to either be used as drinking water for subsystems or it is simply vented over board making sure it would not impact external pieces of equipment/ payloads.

Fire Detection and Suppression

The Fire Detection and Suppression (FDS) function is also attributed to the ECLSS system. Fire requires three basic factors to start the combustion process: fuel, oxygen, and energy. Although it was attempted to use only inflammable materials on board the ISS, energy and oxygen are available—energy in the form of electrical power and oxygen as part of the cabin air. Therefore it is required to have a detection mechanism for potential fire plus the required means to suppress it. The detection function is implemented via multiple smoke detectors spread all over the station. These detectors are connected to the onboard software, which immediately triggers automatic reactions to prevent fire from spreading by shutting down the active air ventilation and removing any possible energy source by cutting the power to suspicious equipment. The FDS subsystem also provides the crew with equipment to actively fight a fire: fire extinguishers and breathing apparatus.

7.1.3.3 Visiting Vehicles - Cargo

A manned space station is operated for years, which therefore requires resupply capabilities for new experiments, replacement hardware, water, food, clothes, propellant, or even entire new modules. It is also required to bring up and down astronauts.

For the ISS this is ensured by an entire fleet of space vehicles (Fig. 7.3). The most famous one, now retired, was the Space Shuttle—also called Space Transportation System (STS)—which was able to deliver up to 24 t of payloads together with two to eight astronauts. The Space Shuttle was also able to return to the surface of the earth after its journey into space. It was capable to bring astronauts back as well as cargo (about 15 t).

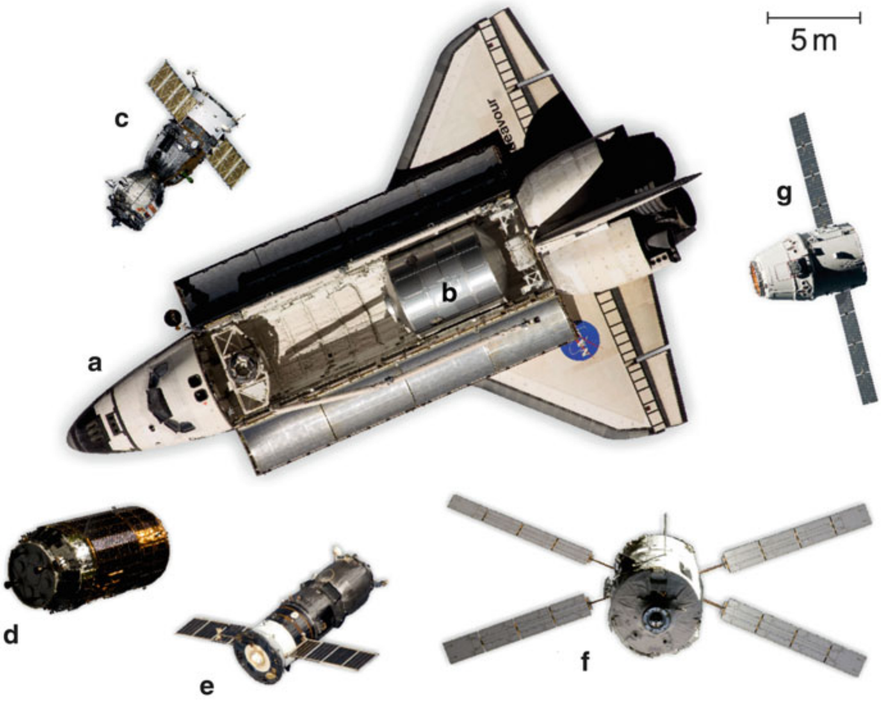

Fig. 7.3 The fleet of ISS supply spacecrafts: The Space Shuttle (**a**), during some missions equipped with the Multipurpose Logistic Module (MPLM, **b**) is retired in the mean time, the Soyuz capsule (**c**) is therefore the only vehicle capable to transport crew to or from the station. The HTV (**d**), the Progress (**e**), and the ATV (**f**) are one-way cargo ships, whereas the commercial Dragon capsule (**g**) can also return items to ground

The space vehicles currently in use for ISS operations are listed below.

- *Soyuz*: Developed by the Russian Space Agency (RSA). This is the oldest vehicle currently used for the ISS. The Soyuz is at the moment (status Aug 2013) the only means to send up and bring back astronauts. The capacity of the Soyuz is three passengers and few tens kg of cargo.
- *Progress*: Derived from the Soyuz by the Russian Space Agency. It is dedicated to fret transportation for about 2.3 t and docks automatically to the station. Once docked to the Station, the Progress is also used as a re-boost means in order to bring the ISS to a higher altitude. Once its mission on the ISS is finished, the Progress is filled with waste and burned in the atmosphere. A Progress is sent to the ISS two to three times a year.
- *ATV*: The Automated Transfer Vehicle is developed by the European Space Agency (ESA). It is dedicated to cargo transportation for up to 8 t of fret. Like the Progress, the ATV docks automatically to the Russian segment of the ISS. It is used for re-boosts and—before leaving the ISS—is filled with waste and then

destroyed during reentry. Four ATV missions have been flown between 2006 and 2013. The last one is planned for 2014.

- *HTV*: The H-II Transfer Vehicle is developed by the Japan Aerospace Exploration Agency (JAXA) and is also dedicated to cargo only operations. The main difference with the ATV and the Progress besides the capacity (6 t for HTV) is the way the HTV is attached to the ISS. The HTV is approaching the ISS until it reaches a distance of a few meters; then it is captured by the robotic arm of the space station which berths HTV to the ISS. Same as for the ATV and Progress, HTV is also filled with waste before being destroyed during its reentry.
- *Dragon*: While the previous vehicles are operated/managed by national space agencies, Dragon is the first capsule which was developed by a private company, the California-based Space-X corporation. The vehicle is the only cargo spacecraft which has the capability to return to ground. Its upload/download capacity is about 3 t.

7.1.3.4 Extra-Vehicular Activity

To assemble the International Space Station in space, a tremendous effort was required, involving many flights of the Space Shuttle or unmanned Russian vehicles. A robotic arm, controlled by an astronaut from inside the station, was used to put the modules together. However, and as mentioned before, sometime the technology is simply not good enough to handle all the tasks which need to be performed. Here human assistance is required even outside of the pressurized modules. Extra-vehicular activities (EVA) are the activities during which an astronaut works outside the space station using a space suit. These activities are not only for building or maintaining the station, they are also used sometime for installation of new experiments on external platforms.

The ISS program distinguishes between three kinds of EVAs, "scheduled" for EVAs planned well in advance and incorporated via the nominal planning, "unscheduled" for EVAs which are not part of the nominal plan and are meant to help to achieve mission objectives, and "contingency" which are also not scheduled in advance and executed in a contingency situation where crew safety or the vehicle are in danger.

In the past most scheduled US-led EVAs have been carried out during Space Shuttle docked phases with the Space Shuttle astronauts conducting them. That way the long duration ISS astronauts could spare the extensive EVA training.

During EVA, the astronaut can be considered as an independent spacecraft, which means that all subsystems already discussed need to be provided by the space suit: either by the Extravehicular Mobility Unit (EMU) or the Orlan suit, the first one being developed by NASA and the second one by the RSA. A space suit ensures a protection for the astronaut against the harsh space environment described in Sect. 1.1, and keeps him or her alive by providing oxygen and water. Although the astronauts always needs to be tethered to the station on their well-defined translation paths along the ISS, they also carry for contingency cases a rescue device with them, which could provide the required thrust and momentum to return to the structure of the station if need be. The longest EVA ever performed lasted 8 h and 56 min—and every “spacewalk” is much more than a walk, but back-breaking work for the astronauts, which requires not only lots of training and preparation on ground but also on orbit. For instance a so-called camp-out is facilitated before the EVA, which prevents decompression sickness for the EVA crew by having the astronauts breathing air at a lower pressure and with a significantly reduced amount of nitrogen to purge nitrogen from their blood system. This is very important since the EMUs only provide 1/3 of the air pressure inside the space station. This low pressure is required to reduce the stiffness of the space suit in the vacuum of space and it becomes easier for the astronaut to move while wearing the space suit outside of the ISS.

7.1.4 Crew: Another Subsystem to Operate

In the previous chapters the spacecraft was divided into subsystems, which can be treated separately, and which interact with each other via well-defined interfaces. This point of view can be kept, if the astronauts are now also considered, treated, and “operated” as one of the space vehicle subsystems.

7.1.4.1 Crew Safety

Whether we are on Earth or in Space, a human life is of an inestimable value and therefore needs to be protected as much as possible. This is also reflected in the basic rule of mission operations, which dictates that under any circumstance the priority order is “Crew—Vehicle—Mission”: The safety of the crew is the highest value to protect, the integrity of the ISS and its modules is second priority, and the mission goals and objectives only rank below.

Emergencies

The ISS program has defined three different emergency cases: Fire, Rapid Depress, and Toxic Atmosphere conditions. For all emergency scenarios there is a first common response, which the crew has memorized and know by heart before they run the specific responses. These are documented in emergency procedures. The operations concept is designed in a way that all emergency responses can be executed independently by crew only. However, the Flight Control Teams are trained to support the crew and take actions in a quicker way than crew could, one reason being that FCT comprises tens of persons whereas the crew are only six on board.

All emergency responses are following the same three-step approach.

1. Warn the other crew members. This includes triggering the ISS wide alarm system, which will create an audible tone all across the station as well as display the emergency condition on the station computers.
2. Gather in safe haven. The astronauts meet in a well-defined, safe area, close to their Soyuz escape vehicles. There, they also have access to the emergency procedures, the required equipment, and the station control computers. It is up to the crew to decide whether they attempt to combat the emergency situation or whether the ISS should be immediately evacuated.
3. Work Emergency procedure. Once the astronauts decided to fight the emergency, they start to work on the emergency procedures. These procedures are written in Russian and English and are available in printed versions to not rely on electronic tools.

Avoidance of Hazards

Already in the design of the station and its components, significant effort is spent on avoiding any potential hazards which could harm the crew during their stay in space. Dedicated safety documentation and review processes have to be followed in the development phase, which ensure that any imaginable danger for the astronauts is avoided. This includes exposure to hazardous material, rotating devices, extreme temperatures or powered equipment, control of pressurized systems, enrichment of oxygen, and release of shatterable material.

Any potential hazard which could be generated by a component is analyzed and proven that the crew is protected against it—for serious consequences of an exposure even multiple independent protections are mandated. Should the design of the component not be able to provide the safety requirements, then operational means of hazard control are requested by the safety community. The operational products, which are described in more detail in Sect. 7.1.5.2, need to contain corresponding controls of the hazard (e.g., the procedure sequence first ensures the power down of the corresponding electrical connectors, before the crew could get in contact with them).

7.1.4.2 Crew Health

The Spacecraft provides a habitable environment for human as described previously; however, the human body is adapted to the conditions on the surface of the earth. Therefore astronauts need to take care of their health while being in space, especially for long duration flight.

A Flight Surgeon or a Biomedical Engineer (BME) is responsible for the health of the crew before, during, and after the flight. They are aware of the medical history of the crew in order to best support them from ground whenever they have a problem on board. They are also responsible for scheduling physical exercises for the crew as well as attending private conferences such as Private Medical Conference (PMC) and

Private Psychological Conference (PPC). They keep track on the astronaut's daily working time so it does not exceed the agreed limit without proper rationales.

Astronauts on board have to train their muscles and the cardiovascular system on a daily basis to prevent degeneration due to the reduced gravity. Normally, 2.5 h of exercise are planned per crew member on one of the different exercise devices on board, which are part of the Crew Health Care System (CHeCS). This subsystem also provides all possible means for medical contingency support, for which the two Crew Medical Officers per Soyuz crew are trained.

7.1.4.3 Crew Communication

Communication between the crew and ground is a key element in human spaceflight operations, and different communication channels are facilitated for the ISS project.

Space to Ground (S/G) Audio Communications

Voice communication with the crew is called Space to Ground communications. Although the radio link to the ISS is, thanks to the TDRS system, almost continuous, the voice traffic is kept to a bare minimum for several reasons such as ensuring a crew can call the ground anytime without having to wait for a conversation to end. This becomes even more important in some specific situations like in an emergency. Of course the numerous public relation activities of the astronauts on board are an exception from this rule.

Usually for the ISS, two Daily Planning Conferences (DPC) per day during the week are conducted. In the morning the crew is, e.g., informed about deltas on experiments or procedures they will be performing throughout the day. In the evening the conference is used to sync up with the crew on the events of the day, the completion status of activities, and the current stowage situation. On Saturdays there might be in addition a so-called Weekly Planning Conference (WPC) scheduled, during which the crew is given an overview of the operations for the upcoming week.

Besides these planning conferences, the audio communication link is used either to assist the crew while they are working on a procedure or in case of off-nominal situations.

Written Communications via Operations Data File

Operations Data File (ODF) is the common name used in the ISS program for procedures. For each activity which needs to be performed, be it executed by the crew or commanded from ground, there is an associated ODF. Before being approved for operations, these ODFs have to follow a strict process of review and validation which involves experts for the experiment design, safety, representatives of the astronauts, and the Flight Control Teams. The ODFs are uplinked to the ISS and therefore serve as a written means of communications.

Supporting Material

On top of ODFs, the crew has access to other electronic documents which bring them additional information and which are uploaded to the ISS by the Flight Control Teams. For example, each day the crew receives a Daily Summary (DS) in which some valuable information (e.g., station configuration, potential targets for Earth observation) or some questions/answers are given. Another example could be a short overview of an experiment including pictures and information giving some context to the crew to better understand what they are about to do.

Timeline

The timeline is a way to communicate the daily working plan to the astronauts on board. Each of the upcoming day's schedule is available to the crew via the On Board Short Term Plan Viewer (OSTPV). This tool allows a graphical representation of the timeline. Designing such a timeline for each day is a challenge for which intensive international coordination is needed. The ISS planning process is described in more detail in Sect. 5.3.

7.1.5 Ground Support Operations

7.1.5.1 Mission Control Center

Spread over four continents, the ISS program is a tremendous international project including five major Mission Control Centers (MCC) working together to operate the ISS. These are the MCC-Houston (MCC-H) in Texas, the MCC-Moscow (MCC-M) in Russia, the Payload Operations and Integrations Center (POIC) in Huntsville/Alabama, the Space Station Integration and Promotion Center (SSIPC) in Tsukuba/Japan, and the Columbus Control Center (Col-CC) in Munich/Germany.

Flight Control Team

Flight operations are conducted by dedicated Flight Control Teams, which core positions are usually staffed or at least reachable 24/7. The team works under the authority of a Flight Director, who is not only responsible for his team but also for the operations taking place in the corresponding module.

Further the Flight Control Team comprises console positions for all different subsystems of the spacecraft, which were mentioned above. It depends on the focus of the corresponding team and the mission phase, whether the mapping between console positions and subsystems is 1:1 or whether several subsystems are merged into one position. Each control center has in addition a planning function, either as dedicated position (e.g., Ops Plan in Houston) or merged into another console position (e.g., COL OC in Munich, which also takes care of payload operation coordination). Each center also provides one crew communicator position, who takes care of all verbal communications with the astronauts and represents the crew's perspective in the team. In addition some ground controllers are required which take care of the complicated ground infrastructure including computers and networks.

Columbus Control Center

As an example for an ISS control center, the Col-CC located in Oberpfaffenhofen, close to Munich/Germany is briefly presented. Col-CC is in charge of Columbus subsystems operations, of providing the European ground segment for Human Spaceflight and of coordinating the European payloads. The actual payload operation is performed by dedicated small control centers, called User Support and Operations Centers (USOCs), which are spread all over Europe (B-USOC in Belgium, BIOTESC in Switzerland, DAMEC in Denmark, CADMOS in France, MARS in Italy, MUSC in Germany, N-USOC in Norway and S-USOC in Spain). This concept of distributed payload operations reflects the approach of the European Space Agency (ESA) to involve all nations funding the European Human Spaceflight efforts actively in the actual project. All these centers are coordinated at Col-CC.

With respect to NASA, Col-CC interfaces directly with the ISS lead control center in Houston, since the American segment provides Columbus with all required resources, and the NASA payload center POIC, which operates part of their payloads inside the European module.

The Flight Control Team at Col-CC is composed of three permanent positions (staffed 24/7) and two part-time positions. The three permanent positions are COL-FLIGHT who acts as Flight Director for Columbus and is in charge of the entire Columbus module, the STRATOS position introduced in 2013 by merging the Data Management Subsystem (DMS), Communications Subsystem (COMMS), TCS, Electrical Power Distribution Subsystem (EPDS) and ECLSS subsystem into one console, and the COL OC position responsible for the payload operation coordination with the abovementioned USOCs and the real-time planning function.

Amongst the non-permanent positions, there is the EUROCOM, in charge of Space to Ground voice communication for Columbus. This position is usually filled in by an astronaut presently on ground or by a crew trainer. The COSMO position is in charge of the on-orbit stowage, the onboard logistics, trash management, as well as up- and download coordination and all mechanical crew activities.

These five teams constitute the core Columbus Flight Control Team. They are supported by the Ground Control Team, which also works 24/7 and takes care of the complex European ground segment.

Also other teams support the Flight Control Team during office hours, like the European Planning Team (EPT) responsible for all non-real-time planning (further than 7 days ahead of execution day), an Engineering Support Team as gateway to the companies which built the module and its components, an ESA Mission Management Team providing programmatic guidance to the Flight Control Team or a Medical Operations Team.

7.1.5.2 Operations

Coordination

Being on four different continents does not ease coordination between all the ISS Mission Control Centers. There is a need to talk to each other and be able to exchange information.

In order to overcome this communication issue, a voice loop system is used. Almost each positions on ground has a "voice loop" assigned which can be compared to a conference line where people can either talk and listen or just listen to what is being said. Each Flight Controller can then discuss with another flight Controller using a headset/microphone and a screen on which all relevant loops for his or her position are available and can be selected. The Space to Ground channels which are used for communications with the ISS are also available via this voice loop system.

For quick, effective, and well-documented information exchange and decision making, a set of tools were designed. The Electronic Flight Note (EFN) system is for example used to exchange written information within the ground teams; this includes the capability to have a review and approval process for each Flight Note. Another example is the Planning Product Change Request (PPCR) tool, which is used to coordinate and agree upon changes to the schedule for upcoming days. Another important piece of software is the console log, which allows to document all decisions, events, or received/provided information in a shift diary. This is in particular helpful for the handover process between shifts.

Operations Products: OPS Products

In order to avoid conflicting approaches of the various control centers and positions, there is need to define in some documents how decisions and interactions are made within the project. The part of this documentation, which is applicable on console, is called Ops Documents and is binding for all centers for real-time operations and interactions/processes. They are briefly explained below.

- Flight Rule: A Flight Rule documents an agreement on how to react depending on a situation. They are designed to minimize the amount of justification and coordination when off-nominal situations occur during real-time operation. A Flight Rule is therefore a very important document and is applicable as "operational law" for all control centers and supersedes all the following documents in case of conflict.
- Payloads Regulation: They are kind of flight rules but dedicated to payloads. Each MCC can have its own set of Payload Regulations for its respective payloads which can be shared with other MCCs if relevant.
- Safety Documentation: Some reference material with regard to operational hazard control is provided here.
- Operations Data File: As explained in the crew communication section, an ODF is a step-by-step procedural description for all activities on board or via ground commanding.
- Operation Interface Procedure: On different levels (interactions within a control center, interactions within Europe, interactions between all international partners) the real-time operational interfaces and standard processes are described in different documents. Depending on the task a Flight Controller might have to consult three different books, if it involves interfaces on different levels.

All ops documents require a dedicated review and approval process to change them, which can require months from the initiation of the update to the final publication. Therefore, for most of the products, also "real-time change processes" are defined, which allow a quicker turnaround on console via Electronic Flight Notes with a final approval of the corresponding Flight Director.

7.1.6 Future

Technology is evolving at an incredible speed. No one can tell what our computers will be able to do in 20 years from now or how man will interact with machines in the future. And this opens up to a wide horizon of innovative and conceptual ideas. Like currently, NASA is working on an experiment on board the ISS called ROBONAUT which is a humanoid robot aiming at replacing man during some EVAs.

Also space tourism is booming: In the past RSA brought already a few space tourists on board the ISS, but the ticket was worth some tens of million dollars. In the meantime, several companies have their own space projects and successful tests are currently ongoing. Tickets for the first flights are already sold out—and approach affordable prizes. And of course, when talking about space travel, Mars is one of the first destinations we usually think of, directly linked to the dream man has to walk on the Martian ground, just as he did on the moon. It is true that current technology still contains some obstacles, but we are getting there. Improvements in so many fields are achieved everyday so that one day we will be able to put all the pieces together and be able to send someone there. We even already have candidates willing to take the risk to be the first one to try the dreamful trip.

All this new kind of missions will need to have their own way to deal with space operations, the basic concept will most likely remain comparable but deltas will be applied in order to fully respond to the need of such missions. Like for a Mars mission, we had in 2010–2011 the so-called Mars500 mission simulating during 520 days a trip to Mars with 6 persons living in a confined environment. The primary goal of this mission was to assess the physiological and psychological impacts on a human being, but the results will most likely also be used to define the operations concept of a future Mars mission. For instance, the delay in voice communication between the ISS and ground is in order of a fraction of a second whereas for Mars the delay would be in order of minutes, up to about 20 min depending on the planets position. This will definitively be a driver in the operations concept design.

Future is full of surprises for space explorers and we will for sure be able to go farer and beyond we ever imagined (International Space Station Familiarization 1998).

7.2 Operations of On-Orbit Servicing Missions

Florian Sellmaier

7.2.1 Introduction

7.2.1.1 What is On-Orbit Servicing?

On-Orbit Serving (OOS) is here understood as type of mission where a servicing satellite is giving service to another satellite or a structure in orbit. In principle three OOS service classes can be distinguished: Observation, motion, & manipulation (Table 7.1).

On-Orbit *Observation* of other spacecraft was so far mainly used for military purposes. In January 2009 a first deep space inspection of the failed missile warning satellite DSP 23 was conducted by two MITEX satellites in the Geostationary Earth Orbit (Sect. 7.2.2.3).

Motion of other spacecraft is planned to be used for fleet management of geostationary satellites, i.e., station keeping, relocation, or disposal activities after

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Table 7.1 OOS service classes

Service class	Kind of service	Examples
Observation	Remote inspection	Mitex (2009)
Motion	Station keeping	OLEV (study)
	Relocation	
	Disposal and de-orbiting	DEOS (planned for 2018)
Manipulation	Refueling	Orbital-Express (2007)
	Maintenance	
	Repair and retrofit	Shuttle/HST (1993)
	Docked inspection	

the spacecraft fuel has been depleted. This was planned with the Orbital Life Extension Vehicle OLEV (Sect. 7.2.2.2). Another application is the controlled de-orbiting of satellites from the low earth orbit in order to avoid damage on earth related to uncontrolled reentry or in the context of Space Debris Mitigation (SDM), a technology which will be demonstrated by DEOS (Sect. 7.2.2.4).

Manipulation of other spacecraft includes both maintenance and repair and is the most complex OOS activity. In 2007 the technology mission Orbital-Express successfully demonstrated the ability to maintain a client satellite including change of battery and exchange of fluids (refueling). The maintenance activities of the Hubble Space Telescope (HST) during 1990 and 2009 are an example of manned On-Orbit Servicing.

7.2.1.2 Motivations for OOS

As indicated in Table 7.1 there are several motivations for an On-Orbit Servicing mission. First, activities like "remote inspect," "repair," or "upgrade" are designed to help fix a malfunctioning spacecraft in orbit. Second, activities like "station keeping" or "refueling" plan to extend the lifetime of a spacecraft in orbit. All the activities mentioned so far are related to reestablishing and/or continuing the service of a client spacecraft by an OOS mission. The crucial question is here whether it is worthwhile to launch a quite complex spacecraft with the goal to repair another spacecraft. The answers will be only positive if the overall value of the OOS mission is higher than the costs. To date only very expensive mission like HST or the International Space Station (ISS) have been repaired or maintained by an OOS mission.

The motivation of "Disposal" or "De-Orbiting" of a spacecraft is either to avoid damage on earth due to uncontrolled de-orbiting or to remove a spacecraft from certain orbits to avoid further enhancement of space debris (e.g., discussed for ENVISAT). These activities are summarized with the expression Space *Debris Mitigation* (SDM) which can be seen as an important function of OOS.

7.2.1.3 Space Debris Mitigation

The existence of space debris becomes more and more relevant for operations. A certain percentage of the operational cost is spent in the context of collision avoidance programs. Additionally, frequent collision avoidance maneuvers decrease the lifetime of a mission. Hence, the process of SDM is becoming increasingly important (see also chapter 1.1.3 for space debris and Sect. 4.1.3.2 for collision avoidance maneuver).

Figure 7.4 shows the density of particles >10 cm peaks in altitudes between 800 and 1,000 km at high inclinations with the highest peak at the Sun-synchronous orbit between 97° and 100°. On one side these peaks are a result of most frequented orbits with a high number of inactive satellites in combination with an altitude where the rest atmosphere is thin enough to avoid self-cleaning through drag. And on the other side the highest collision risk is just in those regions which are most valuable for earth observation and other applications in the low earth orbit. Indeed, this is exactly the orbit where the 2009 satellite collision between Iridium 33 and Cosmos 2251 took place.

This situation already is quite alarming. However, in the future it will become even worse. Figure 7.5 shows a 200 years forecast for the effective LEO objects >10 cm before and after the 2009 satellite collision (solid and dashed thick lines) under the assumption that there are no more launches. The steady increase after 2050 is due to the so-called “Kessler effect”: collision fragments collide with other satellites multiplying the number of collision fragments which will again collide with other satellites, etc. This cascading effect will increase the density of space debris continuously. It shows also that in addition to ESA’s code of conduct active abatement measures are needed. The only way to limit this further increase is to actively remove inoperative satellites. Present forecasts predict that it is necessary to remove approximately 5 larger objects per year from LEO to flatten the increase of the collision fragments (c.f. also Fig. 1.9).

7.2.2 Examples of On-Orbit Servicing Missions

Manned OOS missions have a fairly long history and can be regarded as an integral part of human spaceflight. This began with the repair of the Skylab station and continued with several HST repair missions and servicing missions of the ISS. The most complex part of manned OOS missions is done by humans in space. Hence, the ground operation of this kind of missions is quite similar to the operation of human spaceflight in general (see also Sect. 2.1). Examples of manned OOS missions are described below in Sect. 7.2.2.1.

On the other hand, robotic OOS is still a very new field in space business; hence there are only a few missions as yet. It is quite a challenge to perform servicing

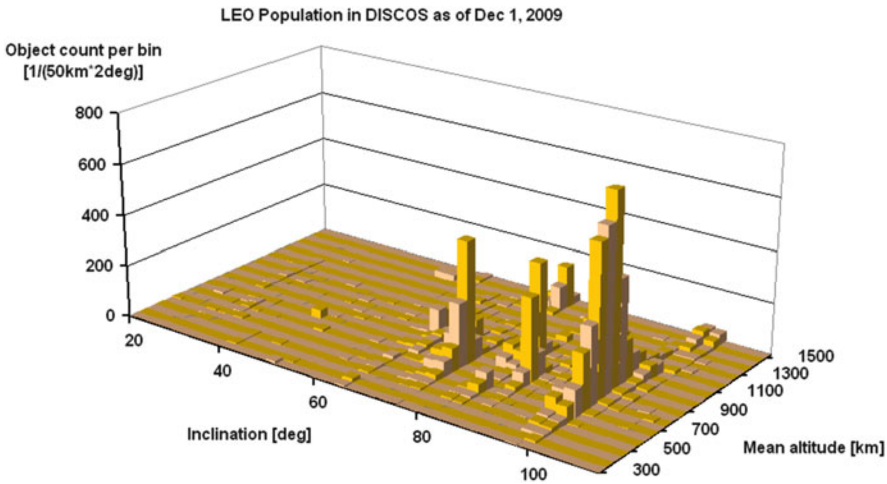

Fig. 7.4 Density of particles larger 10 cm vs. inclination vs. mean altitude (Klinkrad 2010)

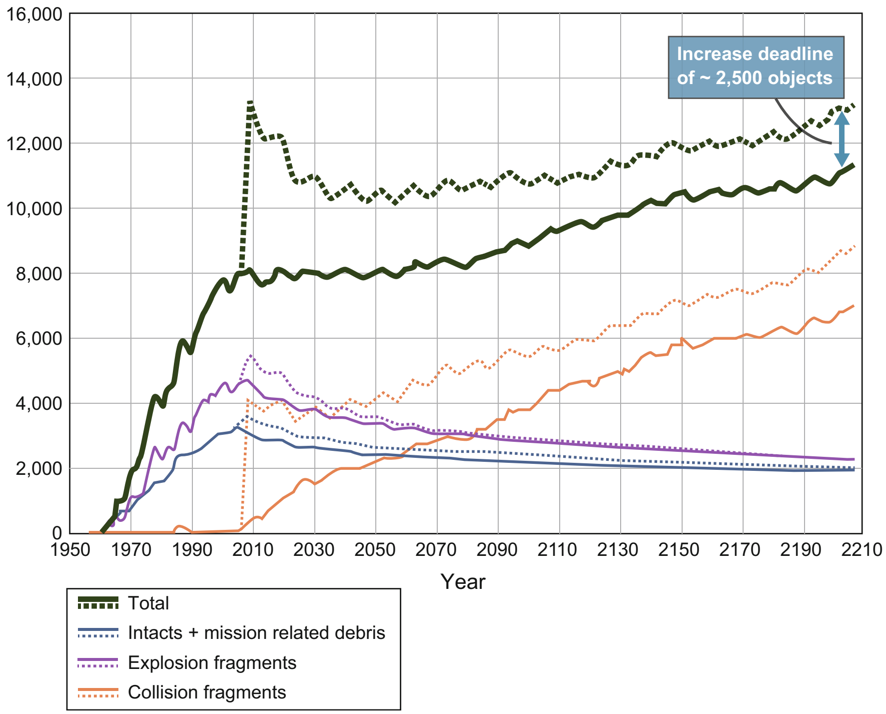

Fig. 7.5 200 years forecast for effective LEO objects >10 cm before (solid) & after the 2009 satellite collision (*dashed*) under the assumption that there are no more launches (credit: NASA)

activities by robots and even more if this activity is located in space. Examples of robotic OOS mission are described in Sect. 7.2.2.2–7.2.2.4.

7.2.2.1 Manned OOS Missions

Repair of Skylab and Hubble Space Telescope

The repair of NASA's Skylab station in 1973 is the first example of a manned On-Orbit Servicing mission. A minute after launch of Skylab 1 ground control received alarming telemetry signals. During launch and deployment Skylab 1 sustained severe damage which resulted in the loss of the micrometeroid & heat protection shield and one of the solar panels. Skylab 1 reached the planned orbit but was not usable. NASA decided to postpone the launch of Skylab 2 in order to analyze the situation. Finally, the crew of Skylab 2 and 3 was successful to repair the damage on the station.

The Hubble Space Telescope (HST) was launched in 1990. Within weeks after launch it became clear that the images were not as sharp as expected. The cause for this problem was that the primary mirror had the wrong shape. One of the first reactions was to rearrange the observation plan to push up less demanding observations like spectroscopy.

Between 1993 and 2009 five shuttle missions serviced and maintained the HST, beginning with a correction measure for the flawed mirror (Fig. 7.6) and including replacements of the Imaging Spectrograph, all gyroscopes, and the data handling unit. As a result of this servicing mission HST is still expected to be in operation until 2014 or even longer. For a mission in LEO this is a quite a respectable lifetime.

7.2.2.2 Studies for Commercial OOS Missions in GEO

From the Loss of TV-Sat 1 to ESS and OLEV

An early example of robotic OOS goes back to 1987 when TV-Sat 1 was launched. Unfortunately, the failure of one solar panel to deploy curtailed operations severely since the antenna could not be activated. After several attempts to repair the spacecraft from ground, it was finally placed in a graveyard orbit in 1989. This incident triggered the idea of a rescue satellite with TV-Sat 1 as an early candidate for OOS studies. DLR worked out the concept of an Experimental Servicing Satellite (ESS) (Settelmeyer et al. 1998) including the design of a capturing tool for the apogee engines of geostationary satellites (Fig. 7.7).

The concept of docking to the apogee engine of communication satellites was adopted by a commercial consortium: The Orbital Life Extension Vehicle (OLEV) aimed to prolong the lifetime of communication satellites whose fuel has almost been depleted. OLEV was designed to dock on such ComSats, take over station

Fig. 7.6 Servicing mission SM1 (STS-61) for Hubble Space Telescope installing corrective optics in December 1993 (credit: NASA)

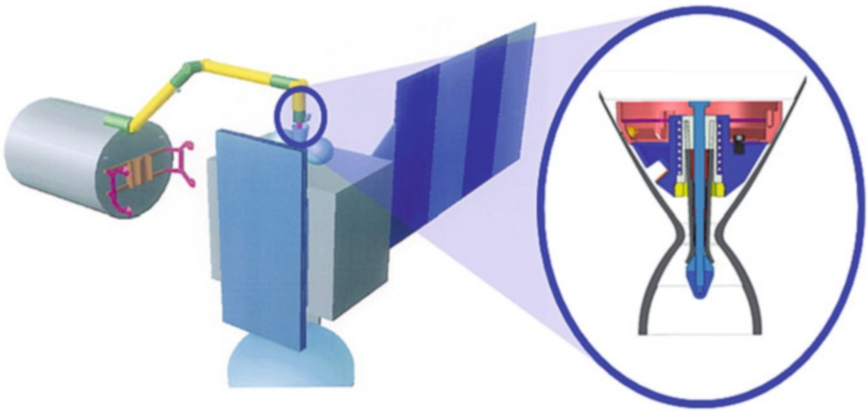

Fig. 7.7 Experimental Servicing Satellite (ESS) repairing TV-Sat 1. The study included the design of a capturing tool (credit: DLR)

keeping and attitude control for about 12 years, or to undock and fly to another client and to perform relocation and disposal maneuver (Fig. 7.8).

The OLEV study finished phase B with a Preliminary Design Review (PDR) in 2009 and there was a customer contract for general fleet management purposes. However, it was finally not possible to fund the nonrecurring development costs for

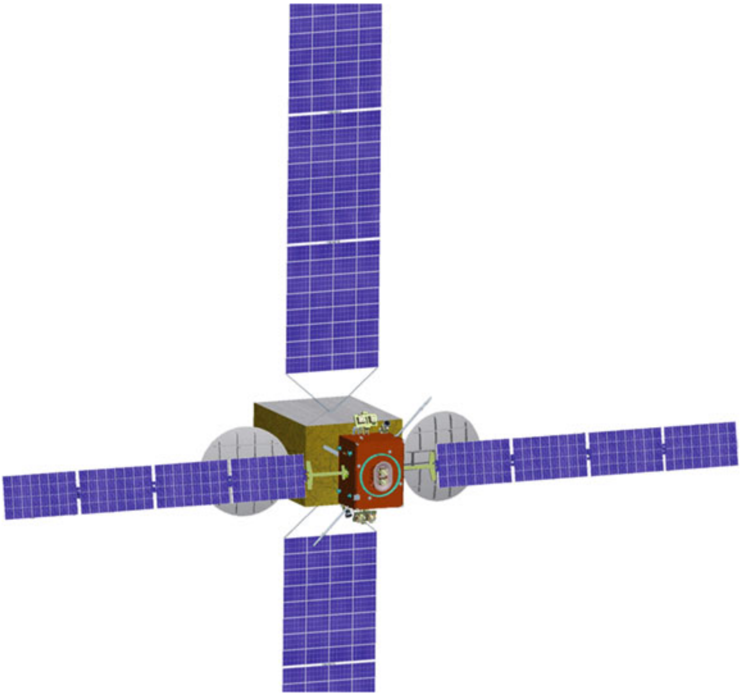

Fig. 7.8 Orbital Life Extension Vehicle (OLEV) with the horizontal solar panels docked on a communication satellite (credit: DLR)

the first mission. This example also demonstrates that especially commercial OOS will be always confronted with the question whether it is cheaper to repair or maintain an existing space system or to build and launch a new space system.

7.2.2.3 OOS Inspection Missions

MITEX

On January 14th, 2009 Spaceflight Now reported:

> *In a top secret operation, the U.S. Defense Dept. is conducting the first deep space inspection of a crippled U.S. military spacecraft. To do this, it is using sensors on two covert inspection satellites that have been prowling geosynchronous orbit for nearly three years.*

The failed satellite which was examined by the MITEX satellites was the Defense Support Program DSP23 missile warning satellite. Details of this inspection mission are not available but it was reported that the radio signatures of DSP23 and one of the MITEX satellites merged indicating a close distance between the two space crafts.

7.2.2.4 OOS Technology Demonstrations

Engineering Test Satellite ETS-VII

The Engineering Test Satellite No. 7 (ETS-VII) was one of the technology demonstrators in the context of OOS (Fig. 7.9). ETS-VII was developed by the National Space Development Agency of Japan (NASDA) and launched in 1997. It was the first satellite equipped with a robotic arm and the first unmanned spacecraft successfully performing autonomous Rendezvous & Docking operations. DLR performed experiments on ETS-VII which are described in Sect. 7.2.5.2 in the context of the satellite capture.

Demonstration for Autonomous Rendezvous Technology

Demonstration for Autonomous Rendezvous Technology (DART) was a NASA mission with the goal to demonstrate a fully automatic approach to the client satellite MUBLCOM. The satellite was launched in 2005 with LIDAR and camera-based sensors on board. DART approached its target as originally planned when suddenly the fuel consumption became much higher than foreseen. Finally, the mission was aborted after 11 h. The NASA investigation board declared the mission as a failure. DART had a collision shortly before mission abort. The target satellite MUBLCOM survived without apparent damage.

Orbital Express

A milestone for robotic OOS was set with the successful completion of DARPA’s Orbital Express (OE) mission in 2007 (Mulder 2008). It comprises of two satellites, ASTRO the servicing satellite and NextSat a prototype next generation serviceable satellite (client). Orbital Express successfully demonstrated the ability to autonomously perform Rendezvous & Docking (RvD) operations including maintenance activities like refueling and maintenance. Figure 7.10 shows the two satellites ASTRO and NextSat of the Orbital Express mission.

The mission philosophy of Orbital Express was to initially crawl, walk, and then run: Both satellites performed the necessary maintenance activities while still in a mated configuration. The transfer of liquids and components like a battery and a CPU—so-called Orbital Replacement Units (ORU)—has been demonstrated. After separation Rendezvous and Docking maneuver had been performed including approach navigation and fly around. Both satellites were left to decay naturally and have subsequently reentered the atmosphere.

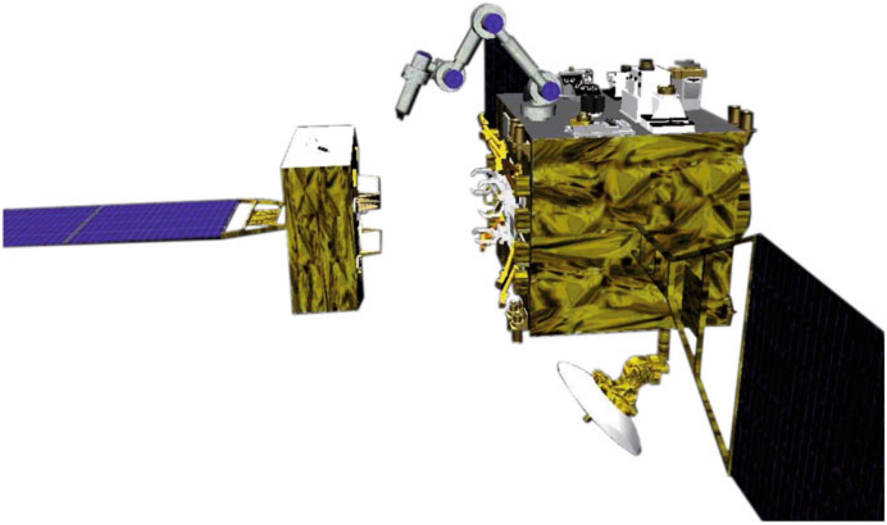

Fig. 7.9 Japanese ETS-VII servicer (*right*) captures target (*source*: NASDA)

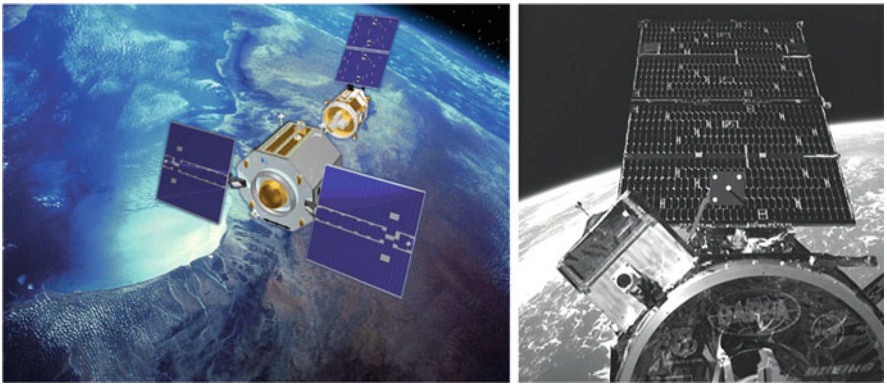

Fig. 7.10 Orbital express. *Left image*: ASTRO (with two solar panels) and NextSat autonomously perform operations while unmated (Credit: Boeing Image). *Right image*: NextSat photographed in space by ASTRO

Outlook: Technology Mission DEOS

The robotic OOS missions described above were designed to dock on clients which are still operative and which have specially prepared docking ports (except OLEV). The technology project DEOS ("Deutsche Orbitale Servicing Mission") wants to demonstrate the ability to capture and move a target spacecraft which is no longer operating and had not been specially prepared. A future application of DEOS will be Space Debris Mitigation. DLR plans to launch DEOS in 2017/18.

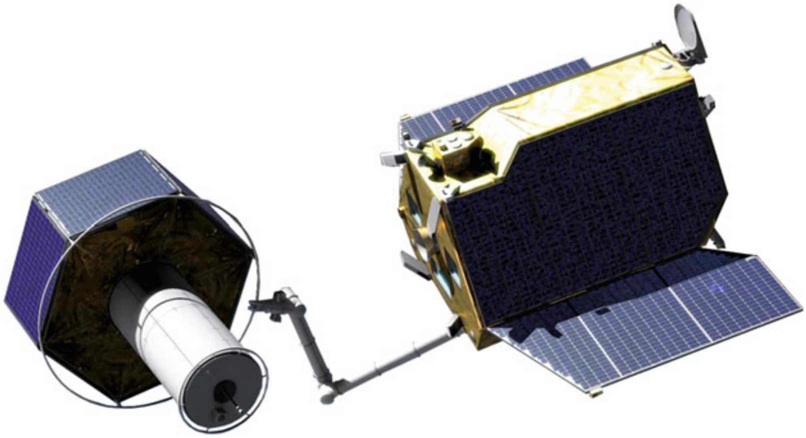

Fig. 7.11 DEOS client (*left*) and servicer (credit: Astrium)

DEOS is similar to Orbital Express in a sense that there are two satellites, one of which is a servicer and the other a client. However, the goals of the two missions differ. The primary goals of DEOS are (1) to capture a tumbling non-supportive client satellite with a servicer spacecraft and (2) to de-orbit the coupled configuration within a predefined orbit corridor at the end of mission. Secondary goals are to perform several rendezvous, capture, and docking scenarios as well as orbit maneuvers with the mated configuration. Therefore the servicer is equipped with an active Attitude and Orbit Control System (AOCS) and both a manipulator and a docking port (Fig. 7.11).

Since the initial experiment conditions such as the tumbling rate of the client have to be set several times, the client is provided with an active Attitude Control System (ACS). For DEOS the expression "non-supportive client" has to be understood in a sense that the client's ACS must not be used during the capture maneuver described above. Additionally, both servicer and client are equipped with Global Positioning System (GPS) and Relative Global Positioning System (RGPS) receivers. Again, GPS and RGPS are not used for nominal approach navigation and capture but as a reference for subsequent evaluation or to prevent a collision during the test. The sensor system used for the nominal approach navigation is a vision-based system using mono and stereo cameras.

Similar to Orbital Express the complexity is successively increased during the mission: Both spacecraft, client and servicer, will be injected together in an initial Low Earth Orbit (LEO). One of the challenges operating DEOS is the continuity of a communication link from ground to LEO. Therefore, the DEOS Servicer will be equipped with an inter-satellite link to a Geo-Relay as an alternative to direct space to ground communication.

7.2.3 Challenges Operating Robotic OOS Missions

The examples of OOS missions above indicate that there are quite some challenges operating OOS missions. We focus here on the challenges operating robotic OOS missions in earth orbit.

What makes the difference between a robotic OOS mission and a standard satellite mission? First of all, there are two spacecraft, one of them approaching the other. This means the approach navigation has to be handled by complexed flight dynamics. Approaching towards the capture point, the collision risk increases which introduces a much shorter reaction timescale compared to standard operations. A Collision Avoidance Maneuver (CAM) has to be performed within seconds whereas a standard command stack is usually time tagged hours in advanced. This has to be accounted for by either a high degree of autonomy or by an enhanced communication system including real-time commanding. And finally, for some missions a noncooperative target has to be captured. This requires high standards regarding robotics or docking technology.

The impact of OOS on the typical systems of a ground segment is summarized in the following. A more detailed discussion follows in the Sects. 7.2.4 and 7.2.5. For a description of the various components of the ground segment, please refer to Chaps. 2–4.

7.2.3.1 Flight Operation System

The fact that there are two spacecraft has to be mapped to the flight operations team, both in terms of operational functions and responsibility. Usually this means that there are two sub-teams preferably integrated within one large control room (Fig. 7.12). However, sometimes it might be convenient to have two separate control rooms for client and servicer. In the worst case, the operation for client and servicer is distributed across distinct control centers in which case the distance should be compensated by good communication links between the two control centers.

Also nonstandard to non-OOS mission is the integration of a Robotic Control System (RCS) within the flight operation system. In some cases a control device like an exoskeleton has to be integrated enabling the reception and transmission of data in real time. The technical requirements for so-called teleoperation are discussed in Sect. 7.2.5.1.

The control authority for a spacecraft is usually the command operator (CMD). This is also true for OOS mission for most of the time. However, during the capture of a client the control authority for the AOCS of the servicer might migrate from CMD to the RCS and back again. This situation has to be analyzed carefully also with respect to possible Collision Avoidance Maneuver (CAM). In some cases it might be appropriate to escape ahead, i.e., to try to capture instead of withdrawing backwards.

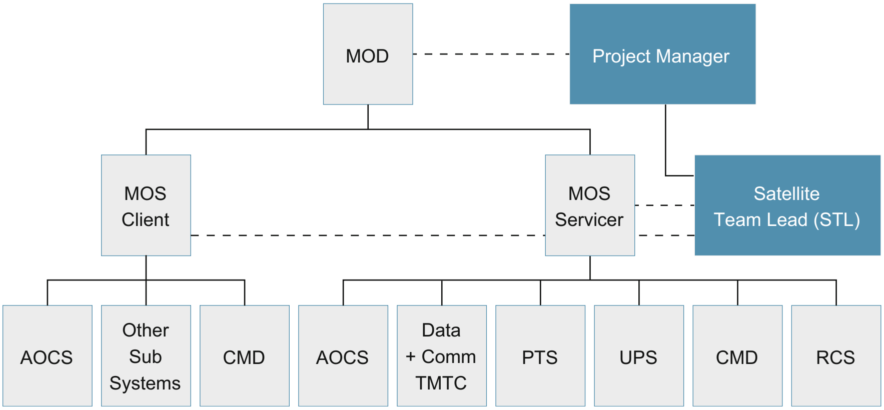

Fig. 7.12 Examples for an integrated flight operations team for both servicer and client. The positions are introduced in Chap. 2 (see also abbreviations)

7.2.3.2 Ground Data System

The Ground Data System (GDS) connects the Flight Operation System with the spacecraft. For standard spacecraft operation, the GDS has to assure a stable and redundant link from ground to space and return. Since most spacecrafts are designed to manage an order of 48 h autonomy, the usual delay time between 2 and 5 s is not a problem at all. The situation differs for a robotic OOS mission: robotic activities will eventually require a telepresence communication link with real-time response (cf. Sect. 7.2.5.1).

Additionally, the final approach including the stay on a hold point in the vicinity of the client and the capture process should be continuously monitored. This is no problem in GEO but LEO is usually limited to contact times below 10 min. For LEO mission the duration can be enhanced using a chain of ground stations (Fig. 7.23). An alternative would be the use of geo-relay such as TDRS and EDRS.

Potential solutions to the above requirements are discussed in Sect. 7.2.5.1.

7.2.3.3 Flight Dynamic System

Any OOS mission which includes Rendezvous & Docking (RvD) maneuver makes high demands on the Flight Dynamic System (FDS). The differences to standard flight dynamics for a single spacecraft is first, that there will be a hand over from absolute to relative navigation within the far range rendezvous phase (Fig. 7.18). The relative navigation will then be based on a sensor system on board the servicer eventually including systems on the client like RF transponder or reflectors for optical navigation. The relative navigation can be closed loop in space or it might have “ground in the loop” which has impacts on the requirements for the ground data system. Finally, the maneuver strategy from far range approach until contact is very demanding for the Flight Dynamic System on ground.

During the next three sections we follow the chronology of an OOS mission beginning with the approach from far range to the capture of the client.

7.2.4 Satellite Rendezvous

Approaching another spacecraft is quite different compared to the standard injection of an earth observation satellite into a low earth orbit. Regarding orbit mechanics there are some similarities with the approach to a parking box in the geostationary orbit. In both cases the maneuver strategy can by ideally discussed within the Local Orbital Frame (LOF) using an analytical solution the so-called Clohessy–Wiltshire Equations (see next subsection). Using the results of the Clohessy–Wiltshire equations for a few examples, we then describe the mission phases from launch to capture including a discussion of different maneuver strategies. Since the sensor system has an important role during the approach of another spacecraft, we describe the common rendezvous sensors in a subsequent section.

7.2.4.1 Orbit Mechanics in Local Orbital Frame

In the following we describe the orbital motion relative to the target within the Local Orbital Frame (LOF) (see Fig. 7.13). The LOF is a commoving coordinate system; the location of the target is defined as the center of LOF.

The distance to the coordinate center, i.e., the target is defined as vector $\boldsymbol{s}$ with:

$$\boldsymbol{s} \equiv \boldsymbol{r} - \boldsymbol{R} \tag{7.1}$$

The coordinates in the LOF are defined as

$$\boldsymbol{s} \equiv \begin{pmatrix} x \\ y \\ z \end{pmatrix} \tag{7.2}$$

This definition of (x, y, z) is commonly used for geostationary satellites. It relates to the alternative nomenclature V-bar, H-bar, and R-bar with

$$\begin{aligned} x &= \text{V-bar} \quad (\text{along track}) \\ y &= \text{H-bar} \quad (\text{south}) \\ z &= \text{R-bar} \quad (\text{to center of earth}) \end{aligned} \tag{7.3}$$

The advantage using the Local Orbital Frame (LOF) can be observed in Fig. 7.14. Whereas it is extremely difficult to track relative motion in the orbital plane frame, there is a clear trajectory visible in the LOF. The LOF is a co-rotating

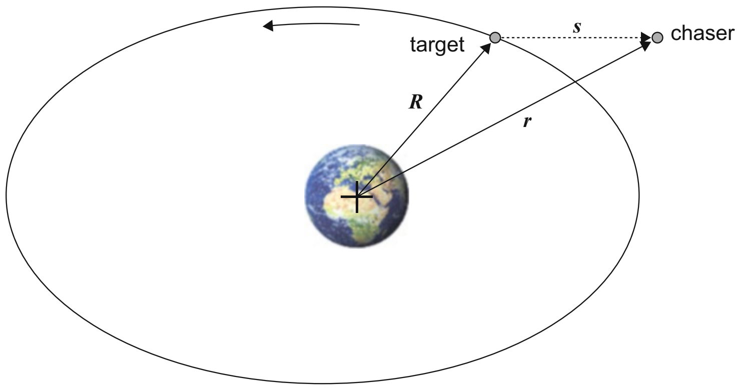

Fig. 7.13 Relative motion in the Local Orbital Frame LOF

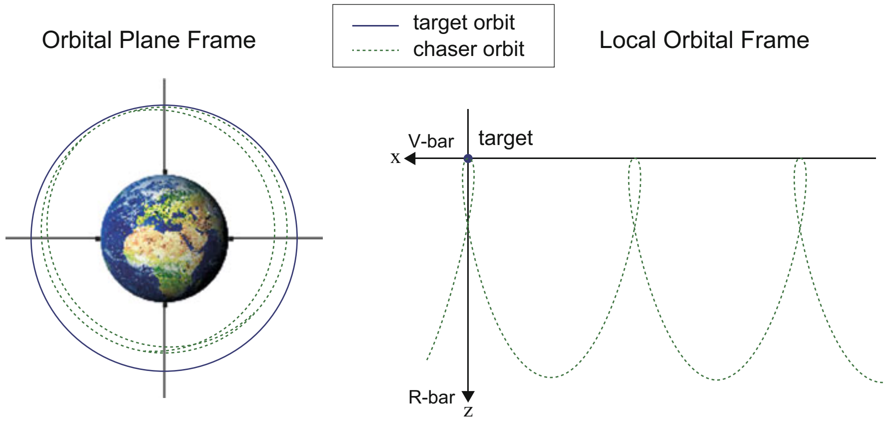

Fig. 7.14 Orbital plane frame (*left*) and Local Orbital Frame (*right*)

coordinate system which is right handed and orthogonal. However, it should be noted that it is not an inertial system; therefore additional pseudo forces like the "Coriolis Force" have to be regarded.

Clohessy–Wiltshire Equations

Using the above definition of the relative coordinates (7.1) and (7.2), an analytical set of equations can be derived based on Newton's law gravity for a central body and his second law of motion (Clohessy and Wiltshire 1960).

The following assumptions were made during the derivation of the Clohessy–Wiltshire equations:

- The distance to the target is much smaller than the distance to the center of gravity: $|\boldsymbol{s}| \ll |\boldsymbol{R}|$
- The target is on a circular Kepler orbit with a constant angular velocity ω.
- There are no external disturbances like drag, solar pressure, or higher moments of the earth gravitational field.

Using these assumptions analytical equation of motion, the so-called Clohessy–Wiltshire Equations, can be derived:

$$\begin{aligned}
x(t) &= \left(\frac{4\dot{x}_0}{\omega} - 6z_0\right)\sin\omega t - \frac{2\dot{z}_0}{\omega}\cos\omega t + (6\omega z_0 - 3\dot{x}_0)t + \left(x_0 + \frac{2\dot{z}_0}{\omega}\right)\\
y(t) &= y_0\cos\omega t + \frac{\dot{y}_0}{\omega}\sin\omega t\\
z(t) &= \left(\frac{2\dot{x}_0}{\omega} - 3z_0\right)\cos\omega t + \frac{\dot{z}_0}{\omega}\sin\omega t + \left(4z_0 - \frac{2\dot{x}_0}{\omega}\right)
\end{aligned} \tag{7.4}$$

The angular velocity is defined as

$$\omega = \frac{2\pi}{T}$$

with T as orbit period. Additionally, the Clohessy–Wiltshire Equations include the initial conditions in space and in velocity, i.e., the initial displacement to the coordinate center $\Delta s = (x_0, y_0, z_0)$ and an initial $\Delta \boldsymbol{v} = (\dot{x}_0, \dot{y}_0, \dot{z}_0)$, e.g., produced by an impulsive thrust.

These are the equations of motion for a spacecraft in the vicinity of a target spacecraft which is located in the center of the coordinate system (i.e., the Local Orbital Frame). Since additional effects like drag, solar pressure, and higher moments of the earth gravitational field are not considered, **these equations are not to be used for operative flight dynamics**. However, the Clohessy–Wiltshire equations are very useful to explain and interpret the most common elements of any approach strategy based on a centralized gravitational field and the effects of the transformation into a rotation coordinate system (Coriolis Effect).

Therefore, in the following, we analyze the orbit propagation of a body located with an offset to the center of the Local Orbital Frame.

(1) Displacement: Trajectories After Release at a Radial Distance

First we investigate the trajectory of a body which is released at a radial distance Z_0 relative to center of the coordinate system (i.e., the target).

The initial conditions for the radial displacement of this body are:

$$\begin{aligned} x_0 &= 0 & \dot{x}_0 &= 0 \\ y_0 &= 0 & \dot{y}_0 &= 0 \\ z_0 &= Z_0 & \dot{z}_0 &= 0 \end{aligned} \tag{7.5}$$

With Eq. (7.4) this yields to following equations of motion:

$$\begin{aligned} x\ (t) &= 6Z_0[\omega t - \sin\omega t] \\ y\ (t) &= 0 \\ z\ (t) &= Z_0[4 - 3\cos\omega t] \end{aligned} \tag{7.6}$$

Figure 7.15 shows the trajectory of a body which is released at a radial distance of $\pm Z_0$. This yields to a cyclic motion with an amplitude of $\pm 6\ Z_0$. Even more relevant, the distance in V-bar after one orbit will be $\pm\ 12\ \pi\ Z_0 \approx 38\ Z_0$. This along track propagation continues with every orbit.

The general lesson of the above consideration is that a displacement in the relative position can propagate by the factor of 38.

This was the reason for Fehse (2003) to postulate a 1 % rule for the accuracy of relative navigation during approach navigation:

> **The accuracy of any sensor system used for relative navigation should be better than 1% of the distance between target and chaser**.

This is especially required for missions in the low earth orbit (LEO) which do not have permanent contact and therefore should have a minimum autonomy of one orbit. This rule might be loosened for an approach in GEO where a permanent link is available or for closed loop Guidance, Navigation, and Control system in space.

(2) Displacement: Trajectories After Release at a Distance in V-Bar

The situation is different for a body which is released in an along-track distance. The initial conditions in this situation are:

$$\begin{aligned} x_0 &= X_0 & \dot{x}_0 &= 0 \\ y_0 &= 0 & \dot{y}_0 &= 0 \\ z_0 &= 0 & \dot{z}_0 &= 0 \end{aligned} \tag{7.7}$$

The Clohessy–Wiltshire equations (7.4) yield to

$$\begin{aligned} x\ (t) &= X_0 \\ y\ (t) &= 0 \\ z\ (t) &= 0 \end{aligned} \tag{7.8}$$

For approach navigation this means:

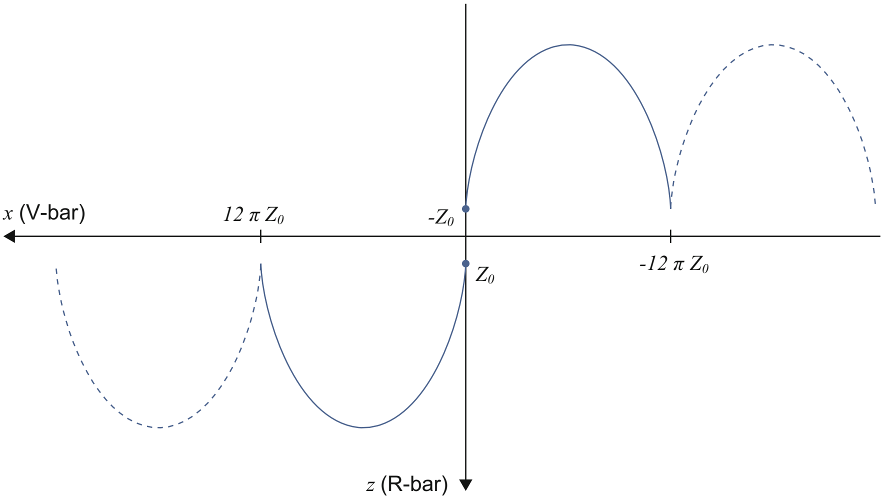

Fig. 7.15 Trajectory after release at a radial distance of $\pm Z_0$ (*dashed line* = second orbit)

> **Points on the target orbit are ideal waiting points with no drift relative to the target**.

Now we need to apply the Clohessy Wiltshire Equations to impulsive maneuvers. This is done by setting the resulting ΔV as an initial condition.

(3) Impulsive Maneuver: Delta-V in Orbital Direction (V-Bar Hop)

If an impulsive thrust is performed into orbital, i.e., along track direction the initial conditions are:

$$\begin{array}{ll} x_0 = 0 & \dot{x}_0 = \Delta V_x \\ y_0 = 0 & \dot{y}_0 = 0 \\ z_0 = 0 & \dot{z}_0 = 0 \end{array} \tag{7.9}$$

This results in following equations of motion:

$$x(t) = \frac{1}{\omega} \Delta V_x [4 \sin(\omega t) - 3\omega t]$$

$$y(t) = 0$$

$$z(t) = \frac{2}{\omega} \Delta V_x [\cos(\omega t) - 1] \tag{7.10}$$

Figure 7.16 shows the corresponding trajectories for both directions.

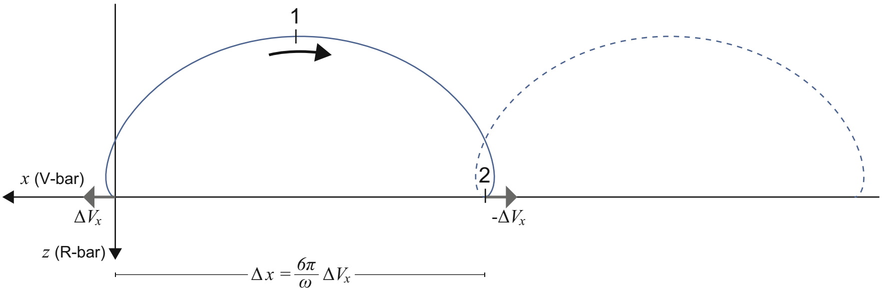

Fig. 7.16 Trajectory after Δv in orbital direction (V-bar hop)

Therefore, the result after one orbit is a propagation of

$$\Delta x = \frac{6\pi}{\omega}\Delta V_x$$

(position 2); this propagation continues in each orbit (dashed line). If the maneuver is stopped with an impulse of $-\Delta V$ after one orbit, it is called a V-bar hop. Compared to other maneuvers like the R-bar hop (see below) we find:

The v − bar hop is the most efficient maneuver to change position within the orbit.

If the propagation is stopped by an impulse of $-\Delta V$ after half an orbit (position 1), this is called a Hohmann transfer.

(4) Impulsive Maneuver: Delta-V in Radial Direction (R-Bar Hop)

An impulsive thrust in radial direction yields to following initial conditions:

$$\begin{array}{ll} x_0 = 0 & \dot{x}_0 = 0 \\ y_0 = 0 & \dot{y}_0 = 0 \\ z_0 = 0 & \dot{z}_0 = \Delta V_z \end{array} \tag{7.11}$$

This results in the equation of motion for an R-bar hop:

$$\begin{aligned} x(t) &= \frac{2}{\omega}\Delta V_z[1 - \cos(\omega t)] \\ y(t) &= 0 \\ z(t) &= \frac{1}{\omega}\Delta V_z \sin(\omega t) \end{aligned} \tag{7.12}$$

Figure 7.17 shows the resulting trajectory for an impulsive maneuver in radial direction.

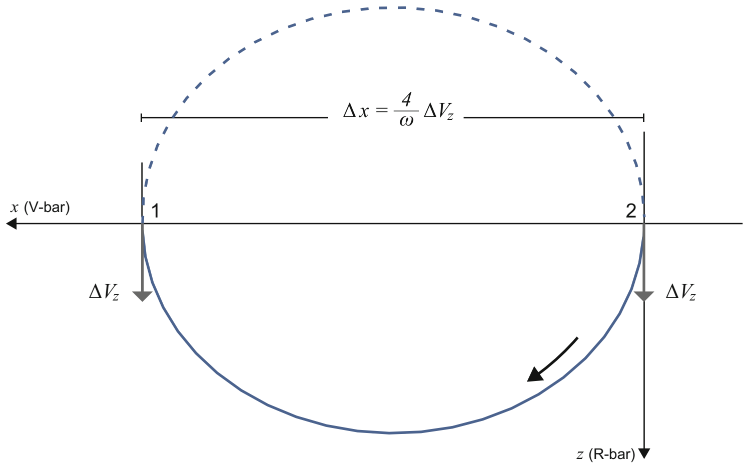

Fig. 7.17 Trajectory after Δv in radial direction (R-bar hop)

The efficiency of an R-bar hop is

$$\Delta x = \frac{4}{\omega} \Delta V_z$$

To stop this maneuver a second pulse ΔV_z is necessary after half an orbit (position 1). Compared to the efficiency of a V-bar hop this is by a factor

$$\frac{6\pi}{4} \approx 4,7$$

smaller.

If the maneuver isn't stopped at position 1, the R-bar hop continues to circle until it reaches position 2 again after one orbit (dashed line) which introduces a kind of passive safety in the vicinity of the target:

> **An R-bar hop is often used as the last two-pulse maneuver approaching a target due to its passive safety in case the 2nd stop maneuver fails**.

On the basis of these four examples described above, most of the maneuvers used during a typical approach can be explained or made plausible.

7.2.4.2 Mission Phases from Launch to Docking

The mission phases from Launch to Docking are thoroughly discussed in Fehse (2003).

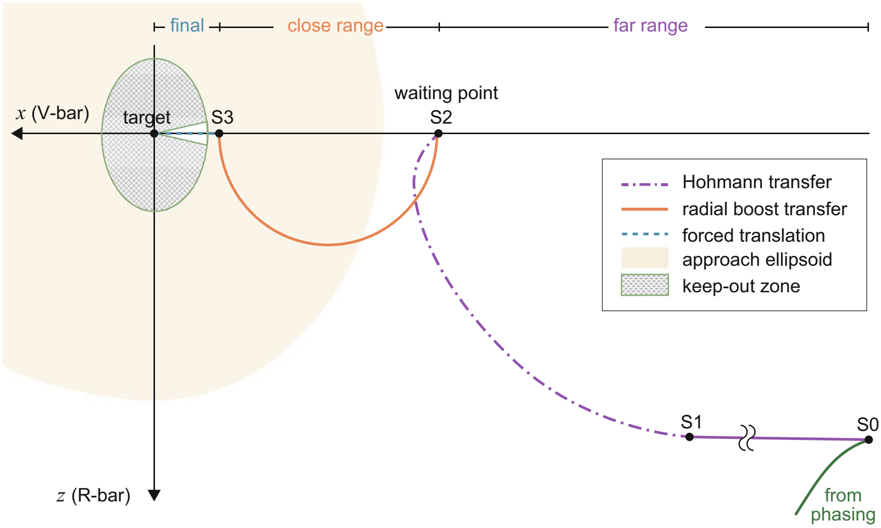

Fig. 7.18 Example for far and close range approach for a LEO mission (Fehse 2003)

(1) Typical Approach in LEO

After injection by the launcher into the orbital plane, the phase angle between chaser and target is reduced during the first approach phase. This phase is called “Phasing” and ends at the “First Aim Point” S0 (Fig. 7.18). This phase is followed by the “Far Range Rendezvous” from S0 to S2. It is the phase where usually absolute navigation is handed over to relative navigation (either closed loop in space or ground in the loop). From S2 to S3 “Close Range Rendezvous” follows as the last two-pulse maneuver in this example. The “Final Approach” is then realized by continuous thrust maneuver until contact and capture of the client.

The approach shown in Fig. 7.18 includes now most of the examples given in the Sect. 7.2.4.1 above:

- After a free drift from S0 to S1, the far range approach is realized as Hohmann transfer which is half of a V-bar hop shown in Fig. 7.16. This is the most efficient two-pulse maneuver to reach the target orbit at S2.
- As described in Eq. (7.8) a spacecraft in the target orbit does not underlie any orbit propagation relative to the target. Hence, S2 is an ideal waiting point.
- For the close range approach from S2 to S3, an radial boost transfer (=R-bar hop) was selected as a last two-pulse maneuver with the advantage that this maneuver introduces passive safety in case of a failure of the stop maneuver at S3 (cf. Fig. 7.17).

The advantages of this approach in LEO are (1) economical use of fuel due to efficient Hohmann transfer in the far range, (2) flexibility due to a waiting point in between, and (3) passive safety due to the R-bar hop in the close range.

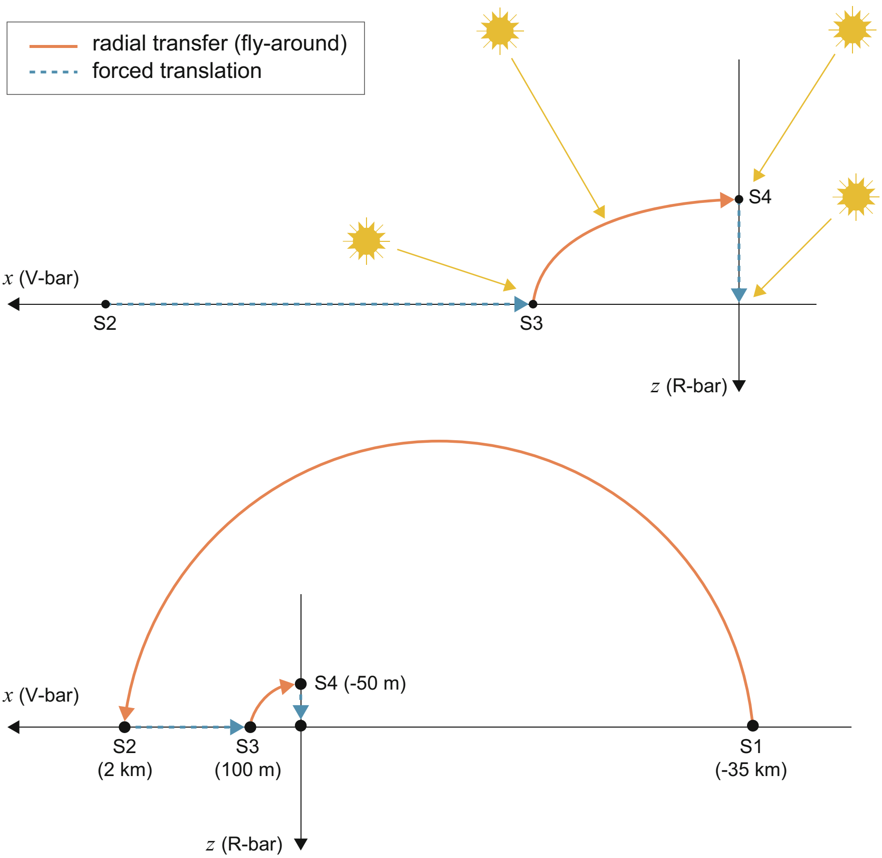

Fig. 7.19 Typical approach in geostationary orbit (example OLEV)

(2) Typical Approach in GEO

The principal differences between an approach in GEO and LEO are first the maneuver duration and second the required Δv.

The duration of a typical two-pulse maneuver is coupled to the orbit period. For the R-bar hop this means a duration of 12 h vs. 45 min. This means that an approach in GEO can be much slower than in LEO. Additionally, the approach can be synchronized to the illumination conditions (Fig. 7.19, top). This is the reason why OLEV was relying on purely optical relative navigation sensors.

Both, for V-bar and R-bar hop the efficiency of the maneuvers are proportional to the orbit period:

$$\Delta x \sim \frac{1}{\omega} \Delta v \sim T \Delta v \tag{7.13}$$

with

$$\frac{T_{\mathrm{GEO}}}{T_{\mathrm{LEO}}} \approx \frac{24\ \mathrm{h}}{1.5\ \mathrm{h}} = 16 \tag{7.14}$$

For a given Δx the required Δ*v* is in GEO approximately by a factor 16 smaller than in LEO.

The first part of an approach in GEO is usually the transfer from the Geostationary Transfer Orbit GTO to the geo-stationary drift orbit. In this example the spacecraft is positioned at a waiting point S1 in a distance of 35 km to the target. The following approach strategy is shown in Fig. 7.19 (bottom):

- From S1 to S2 it includes a semi-fly around, i.e., an R-bar hop at 2 km distance. During this fly around the camera-based navigation system can be calibrated for the use of Angles Only Navigation (see Sect. 7.2.4.3).
- From S2 to S3 a forced motion, continuous thrust maneuver follows. The forced motion maneuver could be interrupted any time since any point on the target orbit is a waiting point.
- From S3 to S4 a second quarter fly around reduces the distance from 100 m to 50 m and brings the servicer in line with the docking axis of the client.
- From S4 to the client a last forced motion maneuver follows in radial direction since this is the docking axis for OLEV docking on the client's apogees engine which is always facing zenith.

The advantages of the approach shown in Fig. 7.19 are the possibility to align the approach with good illumination conditions, to calibrate angles only navigation during the first fly around form S1 to S2, and to use the second fly around from S3 to S4 to inspect the client.

7.2.4.3 Rendezvous Sensors

For a thorough description of the rendezvous sensors, it is recommended to refer to Fehse (2003). In principle rendezvous sensors can be distinguished into absolute navigation sensors giving the absolute position of both client and servicer and into relative navigation sensors resulting in the relative distance and pose between both spacecrafts.

For absolute navigation GPS can be used in LEO whereas ranging methods with radio antennas are available for GEO. For passive targets an active radar antenna, e.g., the antenna of the "Forschungsgesellschaft für Angewandte Naturwissenschaften" (FGAN) in Bonn, Germany, can be used.

For relative navigation Radio Frequency Sensors (RF-Sensors), LIDAR and camera type sensors are feasible for any kind of orbit and targets. Relative GPS (RGPS) is available in LEO and when the target is prepared for RGPS. It should be noted that a camera-based sensor is a passive sensor which relies on proper illumination conditions.

Figure 7.20 shows a comparison of the accuracy of typical rendezvous sensors. The diagonal denominates an accuracy of 1 % of the relative distance which was

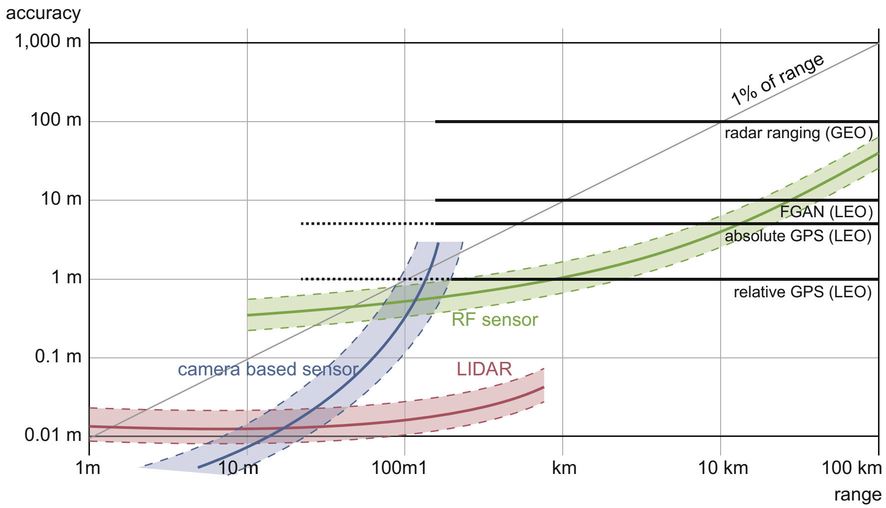

Fig. 7.20 Typical operational ranges and measurement accuracies of rendezvous sensors. The diagonal denominates an accuracy of 1 % of the relative distance

discussed as "1 % rule" above. This is the accuracy usually required for approach navigation (cf. first example given in Sect. 7.2.4.1).

During far range approach accuracies between 100 m (ranging in GEO) and 5 m (GPS in LEO) can be reached by absolute navigation. On the other hand a camera-based sensor is the most accurate sensor in the vicinity of the client (cf. Fig. 7.20 below 10 m). Additionally, camera-based navigation is optimally suited for a pose estimation of the client. The field in between GPS and camera-based sensors can be covered by RF sensors and LIDAR due to the higher range of these sensors.

A method to combine the approach strategy with sensor-based navigation and to extend the range of camera-based sensors is described in the next section.

Angles Only Navigation

Angles Only Navigation is a method to extend the range of camera-based navigation. The problem with camera-based navigation is that the determination of the distance to the target is based on triangulation. The baseline for this triangulation is either the distance between two stereo cameras on the servicer or the size of the target when object resolution is used to determine the distance. Hence, the length of this baseline is limited to a few meters limiting the applicable range for this method to a few hundred meter distance.

Angles Only Navigation uses a calibrated maneuver to replace the baseline for triangulation by a segment of the fly around shown in Fig. 7.21. The only problem with angles only navigation is that it needs a trajectory which is at least to some percentage perpendicular to the line of sight to the target.

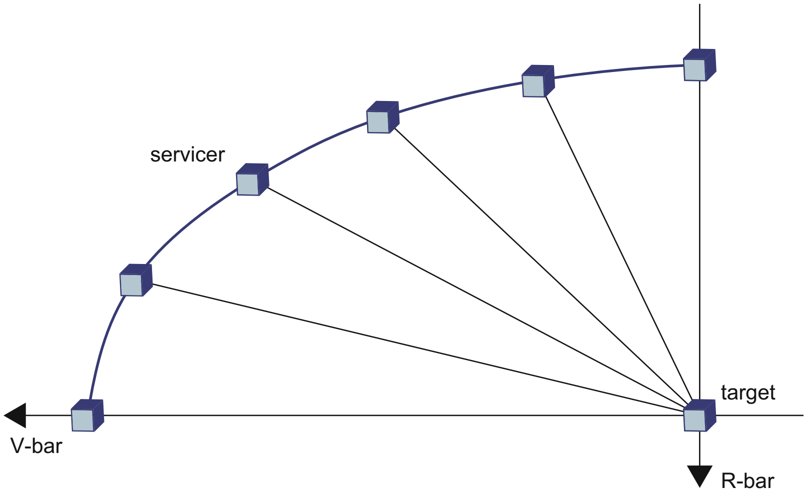

Fig. 7.21 Angles-only measurements during fly around maneuver

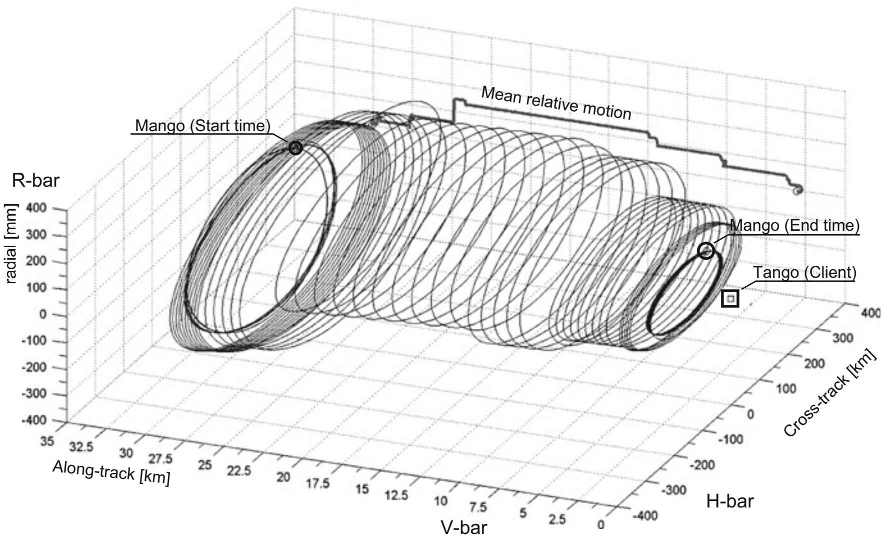

Fig. 7.22 ARGON experiment on PRISMA demonstrating an approach using angles only navigation (credit: DLR)

Figure 7.22 shows an approach strategy optimal suited for camera-based sensors which was demonstrated during the ARGON experiment on PRISMA. This approach combines passive safety with the possibility to use Angles Only Navigation. Passive safety is introduced since the servicer will continue to circle around the client in case a stop maneuver fails. Angles Only Navigation is supported since the trajectory is almost perpendicular to the line of sight to the client (Sellmaier et al. 2010; Spurmann 2011)

7.2.5 Satellite Capture

The capture of a client—either docking or berthing—is the most critical part of the entire OOS mission. In the following we discuss the aspects which are relevant for the capture process which is especially the communication concept and the interaction of the manipulator and the servicer platform.

7.2.5.1 Communication Concept

It is a question of mission philosophy whether to put most of the development effort into autonomy in space or into an enhance space to ground link. The two US missions DART and Orbital Express had the focus on autonomous rendezvous and docking. The failure and collision of DART highlighted the complexity and difficulty with autonomous RvD. However, the success of Orbital Express subsequently demonstrated that autonomous docking can indeed be realized.

The German DEOS mission has adopted an alternative strategy, and its mission will be based on using an enhanced communication link including teleoperation during capture. The requirements for such a communication link between ground and space are:

1. Duration: For preparation and execution of a capture maneuver, there should be enough time to monitor and control the satellites. The timescale depends on the relative velocities involved in the final approach (e.g., 10–30 min in LEO and a few hours in GEO).
2. Stability: The link should be stable and protected against interference and shading by the client satellite.
3. Teleoperation: Whenever a manipulator is operated as "ground in the loop" there should be video streaming and/or force feedback, low delay time (less than 100–500 ms), and low jitter.

Since the first philosophy of enhanced autonomy concerns the space segment only, we focus here on the requirements to improve the communication concept.

Contact Duration

Most of the OOS missions described above are located in a typical low earth orbit with a maximum of 8–10 min contact time per path with one antenna which is barely enough time to monitor and control a capture maneuver itself. Additionally, missions require monitoring during preparation and for follow-up especially if this

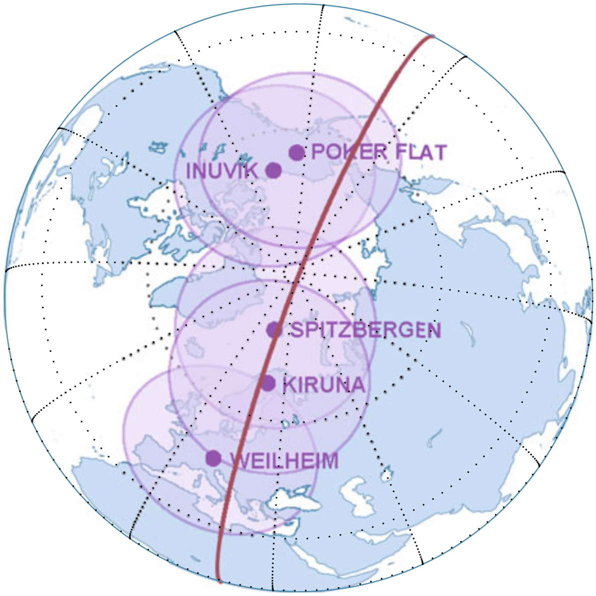

Fig. 7.23 Chain of ground stations to extend contact times in LEO

maneuver is complicated by, e.g., a tumbling client. Hence, the duration of the communication link is a major problem for OOS mission in the low earth orbit.

One solution is to combine a chain of ground stations as demonstrated by Fig. 7.23 which provides a quasi-continuous telemetry (downlink) data stream for more than 20 min. However, telecommand (uplink) will be interrupted since a handover from one ground station to another requires ~1 min. Since the earth is rotating the depicted chain of ground stations can be used every 12 h only.

A more elegant solution to extend the possible contact time is to use an inter-satellite link to a geo data relay satellite between servicer and a dedicated ground station. The additional signal propagation time adds up to about 250 ms (up and down) which is tolerable. However, the availability of geo data relay satellites and/or costs may be a limiting factor to this strategy.

Interference and Shading

The situation regarding link duration is much better for a geostationary OOS mission. However, the final approach to a geostationary satellite is usually combined with a forced motion straight line approach in direction towards the earth with the client between servicer and earth (cf. Fig. 7.19—approach from S4 to client) since this is the most convenient docking axis. This means that the client satellite will be in a direct line between the servicer antenna and the ground station. In this case it is likely that the structure of the client will shade the direct signal between servicer and ground station. Additionally, the transmitter of the client might interfere with the signals from the servicer. Solutions to the problem of shading are either to use additional antennas on the servicer or to use a network of ground station on earth. The problem of interference has to be investigated by a thorough RF compatibility test.

Teleoperations: Delay Time and Jitter

The requirement to minimize delay time and jitter is driven by the robotic operations in the final rendezvous and docking/capture phase. The Robotic Control System (RCS) of an operator conducted capture process requires a delay time of less than 500 ms (round trip) during the robotic phase, i.e., the capture of the tumbling client.

The problem is that the standard communication architecture introduces a delay time of typically 2–5 s, mainly due to electronic components on ground. Additionally, automatic switching of redundant lines may cause unpredictable jitter.

A possible solution to these problems is to connect the RCS directly with the CORTEX (CTX) of the teleoperation antenna with a dedicated nonredundant high rate TM/TC link (Fig. 7.24 dashed lines). The direct connection introduces a very small delay time of 2.5 ms. This solution was used for the operation of ROKVISS (Landzettel et al. 2006b), a robotic precursor experiment on the ISS.

7.2.5.2 Interaction of Manipulator With Servicer Platform

Whenever an OOS mission involves a manipulator, its interaction with the Attitude and Orbit Control System (AOCS) of the servicer platform has to be regarded. Since the overall momentum is conserved, motions of the manipulator will affect the momentum of the platform. DLR investigated this effect during the ETS VII contribution GETEX (see Landzettel et al. 2006a and references within). The conclusion of this investigation was that in most cases it is better to deactivate the servicers AOCS and pre-calculate and include the reaction on the free floating platform into the manipulator trajectory (Fig. 7.25).

7.2.5.3 Additional Aspects

The process of capturing a client spacecraft introduces additional aspects to operations.

First, a collision avoidance strategy has to be developed. The collision avoidance procedure should be independent from a successful communication link. For example, there should be always a valid collision avoidance procedure loaded on board. In certain cases this would mean a backward maneuver; in other cases it would result in a forward maneuver towards docking or capture.

Additionally, a strategy has to be worked out how to shift or split the control authority between the Robotic Control System (RCS) and the servicer's AOCS in order to coordinate the complete system.

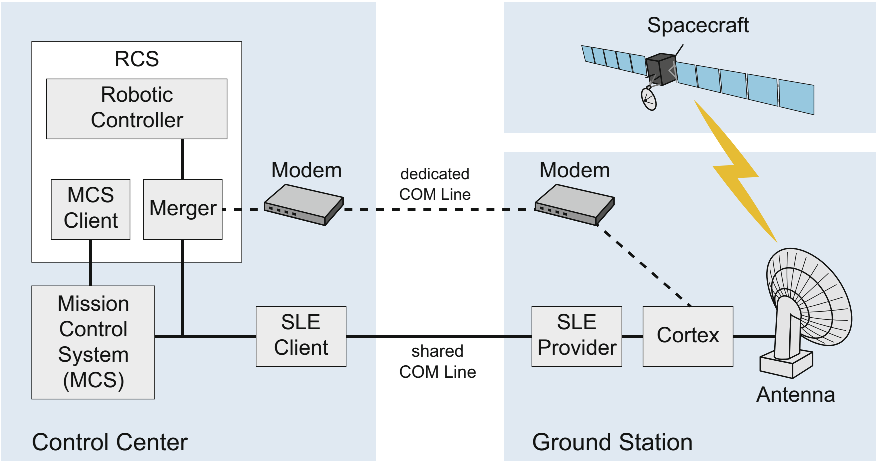

Fig. 7.24 Communication architecture to minimize delay time and jitter

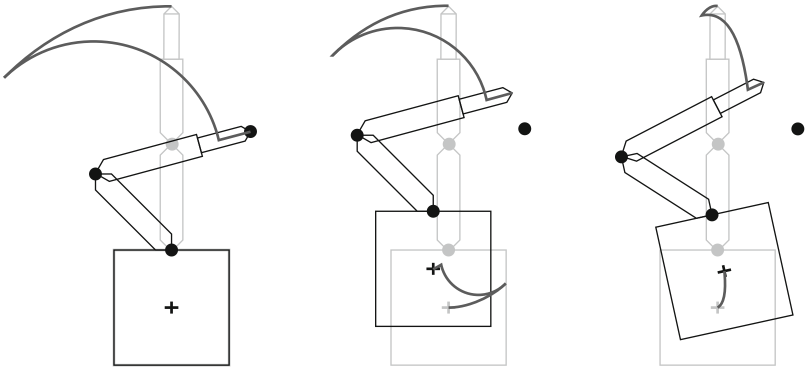

Fig. 7.25 The influence of the satellite attitude control mode on the path described by the robot manipulator. The same joint motion is carried out by a servicer with a fixed base (*left*), an attitude controlled servicer (*middle*), and a free floating servicer (*right*) (Reintsema et al. 2007)

Finally, in the moment the client is captured the clients AOCS have to be deactivated to avoid building up oscillations due to a positive feedback loop between the clients and the servicers AOCS.

In the moment of capture a contact voltage will occur since it is rather unlikely that both spacecrafts will have the some electric potential. The system has to cope with this voltage.

After capture, a possible tumbling rate of the client has to be damped by the manipulator.

7.2.6 Verification and Test Facilities

The requirements of OOS missions for Guidance, Navigation, and Control (GNC) are quite different compared to standard spacecraft operations where for instance a communication satellite has to be positioned within a box of 70 km edge length. The critical phase of OOS missions, the Rendezvous and Docking (RvD) of two spacecraft, is a very complex maneuver which requires (relative) position accuracy of a few mm. In consideration of these circumstances, an RvD maneuver shouldn't be performed in space for the first time. All RvD maneuvers have to be analyzed, simulated, and verified on ground in detail beforehand. Numerical simulations deliver only limited results. Therefore tests or test facilities have to be defined where the entire RvD process including the flight hardware of GNC components and systems can be simulated and tested under utmost realistic conditions of the space environment.

The requirements on testing the robotic OOS missions can be summarized in following three categories.

Approach A test facility should be appropriate to verify sensors and systems within the entire range of vision-based relative navigation, i.e., from several kms down to contact. For camera-based sensors this can be realized in a combination of scaled models and a sufficient range of the test facility. Additionally, the facility shall provide utmost realistic environmental conditions, i.e., the simulation of the sun illumination effect under all angles of incidence and the simulation of the reduced gravity force in orbit.

Capture In order to verify the final "robotic phase" of the RvD maneuver, namely the capture of the client satellite, contact dynamics have to be included. This implies a sensor to measure the contact forces and torques and a dynamic model of both satellites (client and servicer) to simulate the reaction on the contact during the capture process, e.g., the implementation of the Clohessy Wiltshire equations (7.4). Furthermore, the test bed has to guarantee an accuracy which is a dimension better compared to the required spacecraft position accuracy which is in the range of millimeter. Hence, the test bed needs to guaranty accuracy in the range of sub-millimeter.

Integration The facility shall be able to support an integrated system test including RvD system hardware-in-the-loop. It should further be connected to the control center infrastructure including the mission control system (MCS) and the robotic control system (RCS) as well as a realistic ground data infrastructure with respect to delay time and jitter. Finally, the facility shall be used for operator training and mission support.

Figure 7.26 shows a test facility which was designed according to the above requirements. The European Proximity Operations Simulator (EPOS) facility is a

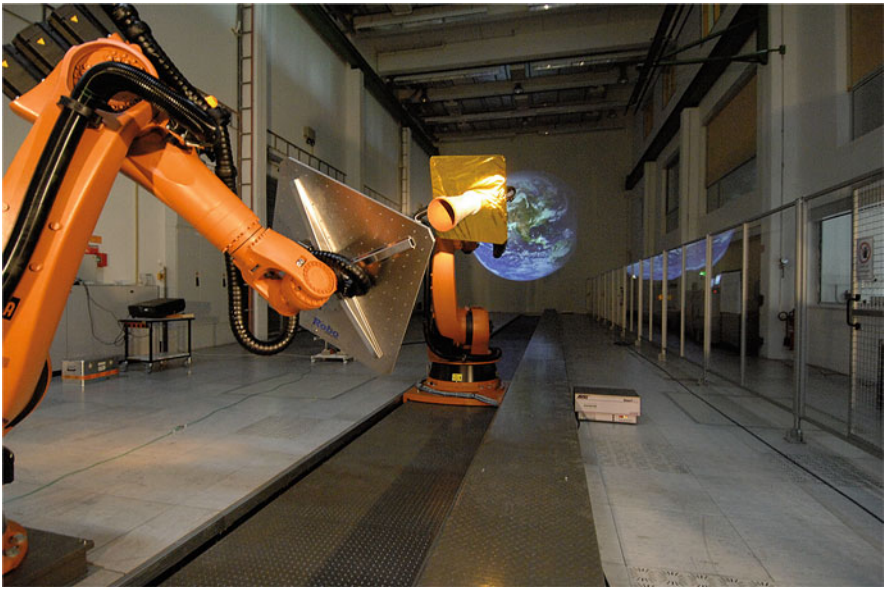

Fig. 7.26 European proximity operations simulator EPOS (*source*: DLR)

hardware-in-the-loop simulator based on two industrial robots, one of them mounted on a 25 m rail system.

7.2.7 Summary and Outlook

While OOS is quite common for human spaceflight missions, it is still a new field for robotic missions. Several demonstration missions have been performed so far and the corresponding technologies have been developed.

The upcoming DEOS mission (launch planned for 2018) will potentially extend these developments also to an application on non-supportive and tumbling clients.

For the future success of servicing missions, it will be necessary to develop interface standards for the client satellites, both for rendezvous and docking as well as for the exchange of so-called Orbital Replacement Units (ORU). Present technology development projects investigate the design of mechanical, electrical, data, and heat flux interfaces for modular components.

Also with regard to possible space debris mitigation programs, all future satellites should be equipped with a standard docking or berthing interface. Additionally, it will be necessary to build a system which is able to de-orbit a series of client satellite with one service spacecraft.

7.3 Interplanetary Operations

Paolo Ferri

7.3.1 *Types of Interplanetary Missions*

The common definition of interplanetary mission refers to spacecraft whose trajectory leaves the environment and enters a heliocentric orbit. Excluded are missions to Sun–Earth Lagrange points L1 and L2 (see Logsdon and Tom 1997), as these remain at relatively close distances to Earth (about 1.5 million km) and do not vary significantly their distance to Sun compared to the one of Earth.

The heliocentric trajectory is selected such that it brings the spacecraft to the selected target, either through a pure ballistic flight or with the aid of correction manoeuvres. These manoeuvres are performed via the onboard propulsion system, or make use of the gravitational slingshot effect of a planet.

The objective of interplanetary missions is the scientific exploration of the Solar System. To this aim, spacecraft are flown towards the Sun, the major planets, or other minor objects like asteroids and comets. The type of trajectory and approach to the target object largely drive the complexity of the mission. We can classify interplanetary missions, according to this criterion and in order of complexity as: fly-by missions, orbiting missions, and landing missions.

7.3.1.1 Fly-by Missions

Fly-by missions were the first historical interplanetary missions. These simply attempted to fly-by the target object, which means to cross the trajectory of the target object at the smallest possible distance, to allow observations and scientific measurements to be taken with the onboard instruments during the fly-by. After the first Lunar fly-bys (Lunik 1, USSR, and Pioneer 4, USA, 1959) various spacecraft were sent onto fly-by trajectories to the nearby planets, Mars and Venus. Missions to the outer planets remained confined to the fly-by type until the NASA Galileo spacecraft entered in orbit around Jupiter in the 1990s, and NASA's Cassini around Saturn in 2005. Also Mercury was only visited by a fly-by spacecraft (Mariner 10, two flybys in 1974) until Messenger entered in orbit around it in 2011. Table 7.2 shows all the solar system objects flown by spacecraft to date, including the date, flyby distance, and spacecraft that performed the first flyby. The

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Table 7.2 Solar system objects flown by spacecraft (status end 2012)

Type	Target	Mission	Date	Distance (km)
Moon	Moon	Lunik 1	02-Jan-59	6,000
Planet	Venus	Mariner 2	14-Dec-62	34,773
Planet	Mars	Mariner 3	14-Jul-65	9,844
Planet	Jupiter	Pioneer 10	03-Dec-73	130,000
Planet	Mercury	Mariner 10	29-Mar-74	705
Planet	Saturn	Pioneer 11	01-Sep-79	21,000
Planet	Uranus	Voyager 2	24-Jan-86	81,422
Planet	Neptune	Voyager 2	25-Aug-89	4,824
Asteroid	951 Gaspra	Galileo	29-Oct-91	16,200
Asteroid	243 Isa	Galileo	28-Aug-93	10,500
Asteroid	253 Mathilde	NEAR	27-Oct-97	1,200
Asteroid	9969 Braille	Deep Space 1	29-Jul-99	26
Asteroid	5535 Anne Frank	Stardust	02-Nov-02	3,300
Asteroid	2867 Steins	Rosetta	05-Sept-08	800
Asteroid	21 Lutetia	Rosetta	10-Jul-10	3,160
Asteroid	4179 Toutatis	Chang'e 2	15-Dec-12	3.2
Comet	21P/Giacobini-Zinner	ICE	11-Sep-85	7,800
Comet	1P/Halley	Giotto	14-Mar-86	600
Comet	26P/Grigg-Skjellerup	Giotto	02-Jul-90	200
Comet	9P/Tempel1	Stardust	14-Feb-11	181
Comet	19P/Borrelly	Deep Space 1	22-Sep-01	2,200
Comet	81P/Wild2	Stardust	02-Jan-04	240
Comet	9P/Tempel1	Deep Impact	04-Jul-05	500
Comet	103P/Hartley2	EPOXI	04-Nov-10	700

flybys of the Jupiter and Saturn moons performed by Galileo and Cassini have been omitted for simplicity.

Fly-by missions do not require complex manoeuvres at arrival at the target and the requirements on navigation accuracy can be adapted to the available capabilities by adjusting the fly-by distance. The design of the spacecraft is therefore relatively simple and its mass does not have to include heavy propulsion systems and the necessary fuel. Fly-by missions can further be designed such that they visit multiple targets. Perhaps the most successful typical examples of such a mission are the two Voyager spacecraft, whose trajectory was designed such that they could visit several planets of the external solar system. Drawback of such missions is the short time spent in the proximity of the target, which is typically of the duration of a few hours, due to the large relative speed at the close encounter.

7.3.1.2 Orbiting Missions

Orbiting missions are those that bring the spacecraft to the target and then execute trajectory correction manoeuvres to allow it to be captured by the gravity field of the object and enter a closed orbit around it. Only a few Solar System objects have

Table 7.3 Solar system objects orbited by spacecraft (status end 2012)

Type	Target	Mission	Nation	Year
Planet	Earth	Sputnik 1	USSR	1957
Moon	Moon	Lunik 10	USSR	1966
Planet	Mars	Mariner 9	USA	1971
Planet	Venus	Venera 9	USSR	1975
Planet	Jupiter	Galileo	USA	1995
Asteroid	(433) Eros	Near-Shoemaker	USA	2000
Planet	Saturn	Cassini	USA	2004
Asteroid	(25143) Itokawa	Hayabusa 1	Japan	2005
Planet	Mercury	Messenger	USA	2011
Asteroid	(4) Vesta	Dawn	USA	2011

been orbited by spacecraft: apart from Earth and the Moon, the planets Mars and Venus were (and still are) orbited by several interplanetary probes. Mercury, Jupiter, and Saturn have experienced to date only a single orbiter each. Uranus and Neptune have not been orbited yet. Of the minor bodies, only three asteroids have been orbited (Eros, Itokawa, Vesta). All the objects orbited by spacecraft to date are listed in Table 7.3 below.

It is obvious that orbiting missions require a higher navigation accuracy, to reach the required precision in the trajectory determination and orbit insertion manoeuvres execution. The spacecraft is also heavier at launch, as it has to carry the necessary propellant for the execution of the orbit insertion manoeuvres, and it has to be equipped with a dedicated propulsion system. The orbit insertion operations are time critical, and require heavy ground support to ensure correct and timely execution and rapid intervention in case of problems. The advantage of such missions is that the spacecraft and its scientific instruments remain in the proximity of the target object for the entire mission lifetime, which is typically of several years, allowing in-depth observations, mapping of the surface, etc.

7.3.1.3 Landing Missions

The next step in complexity of interplanetary missions is represented by the ones that attempt a landing on the surface of the target body. In the history of spaceflight and Solar System exploration successful landing missions have reached the Moon, Mars, Venus, and Saturn's moon Titan. Also a special mention should be made of NASA's Near-Shoemaker spacecraft, which at the end of its successful orbiting mission around asteroid Eros, was commanded to an unplanned, graceful "landing" onto the surface of the asteroid, and managed to survive for about 16 days after the touchdown. Table 7.4 lists all the solar system objects on which a successful landing has taken place, together with the first landing mission and the year of landing.

Landing missions are more complex as they normally require a separate landing module which, depending on the characteristics of the target body, in particular its gravity and atmosphere, may require complex and heavy systems like heat shields,

Table 7.4 Solar system objects on which spacecraft successfully landed (status end 2012)

Type	Target	Mission	Nation	Year
Planet	Earth	Sputnik 5	USSR	1960
Moon	Moon	Luna 9	USSR	1966
Planet	Venus	Venera 7	USSR	1970
Planet	Mars	Mars 2	USSR	1971
Asteroid	(433) Eros	NEAR-Shoemaker	USA	2001
Moon	Titan	Huygens	Europe	2005

Only the corresponding first landing mission is included

parachutes, retro-rockets, landing gear, airbags, etc. In addition a landing mission increases the complexity of communications to Earth, generally implying also the use of data relay orbiters around the target body. The complexity of a landing mission is therefore much larger than the one of an orbiting mission. Advantages of having a landing platform capable to carry scientific instrumentation onto the surface of the target body are, on the other hand, enormous and obvious: a lander (in case it is mobile on the surface it is called a rover) allows in situ observations and experiments that would not be possible via remote sensing from the orbit.

A special case of landing missions, which add further complexity to the mission and spacecraft design, are sample return missions. These require the capability, after having landed on the target object, to take off again, leave the object, return to Earth, and deliver the samples to the surface. Apart for the Apollo manned Moon landing missions, which returned to Earth Moon samples collected by the astronauts, sample return from a solar system object was only successfully achieved by the JAXA Hayabusa 1 mission, which in 2005 touched asteroid Itokawa and collected small particles lifted by the contact, returning them to Earth in a small reentry capsule in 2010. For the sake of completeness, two NASA missions, Genesis in 2004 and Stardust in 2006, managed to return some samples of solar wind particles and comet's coma dust back to Earth. These, however, were flyby missions and cannot be classified as landing sample return achievements. Landing missions and their complex operational implications are described in detail in Sect. 7.4.

7.3.2 The Challenges of Interplanetary Flight

The main problems of interplanetary flight are all related to energy: the enormous dynamical energy required to achieve its target orbit, the generation and delivery of electrical and thermal energy to power and keep warm the onboard systems, and the energy of the radio signals required to keep communications to Earth. All three aspects pose major challenges both to the spacecraft design and to the operations of interplanetary probes. These are very briefly summarized in the sections below.

7.3.2.1 Trajectory Dynamics

Interplanetary travel means escaping from the earth orbit into a heliocentric orbit. This requires a minimum escape velocity of 11.2 km/s that can be provided by launchers which are available today. Once the spacecraft is injected in the escape heliocentric orbit, it can start its cruise towards the target, which may last several months or years. Small trajectory corrections may be required over the cruise, or in some cases larger "deep space manoeuvres," i.e., deterministic modifications of the interplanetary trajectory performed at an optimal (from the energy point of view) time to direct the spacecraft to the target. Once arrived at the target, if the objective is to enter in a closed orbit around it the spacecraft has to perform an injection manoeuvre, which is typically a large burn (typically a few km/s) of chemical fuel over a relatively short time (typically of the order of 1 h). Examples of direct trajectories to the target are all the historical fly-by missions (e.g., Giotto with comet Halley in 1986, or all the early missions to the planets). ESA has used this approach for its two missions to our neighbor planets, Mars Express and Venus Express.

However, the orbital energy that can be provided by the most powerful existing launchers may be sufficient to leave the earth gravitational influence, but is often not sufficient to reach the mission target. This is the case for instance for journeys to planets beyond the main asteroids belt (Jupiter and beyond), where the orbital energy increase is enormous, or for journeys to Mercury, where in fact the spacecraft coming from Earth has to release large amounts of orbital energy. The method adopted in this case is the use of gravity assist manoeuvres. This means flying close to a planet and utilizing the gravity field of the planet to transfer some of the heliocentric orbital energy of the planet to the spacecraft. Changes to the orbital velocity of the order of a few km/s can be achieved with a single "swing-by." This is equivalent to carrying a few tons of chemical fuel, which is of course impractical for deep space probes. Table 7.5 shows the accelerations achieved by Rosetta with each of its four planet swing-bys, compared to the total acceleration (Delta-V) achievable with its own propulsion system.

With the right combination of swing-bys, a spacecraft can be directed to any target in the solar system. Examples are all the missions to the external solar system, from Voyager to Galileo and Cassini. The NASA Messenger spacecraft and, in the future, the ESA BepiColombo mission also utilize several planet swing-bys in order to enter in orbit around Mercury, using its gravity to decelerate the spacecraft.

The operational implications of the complex trajectories and the critical activities such as planet swing-bys are explained in Sect. 7.3.4.1.

7.3.2.2 Energy for the Onboard Systems

All spacecraft require electrical energy to power the avionics systems and the auxiliary units like sensors and actuators and their control electronics, or to be

Table 7.5 Rosetta achieved acceleration with planet swing-bys

	ΔV (km/s)
Earth Swing-by 1	5.9
Mars Swing-by	2.3
Earth Swing-by 2	5.2
Earth Swing-by 3	6.3
1.7 Tons chemical fuel	2.2

Table 7.6 Mean solar flux at the major planets

	Min–max sun distance (AU)	Mean solar flux (W/m^2)
Mercury	0.31–0.47	9,066
Venus	0.72–0.73	2,601
Earth	1.0	1,358
Mars	1.38–1.67	586
Jupiter	4.95–5.45	50
Saturn	9.0–10.0	15
Uranus	18.2–20.3	4
Neptune	30.0–30.3	2
Pluto	29.5–50.0	1

converted in thermal energy to ensure temperature control of the various spacecraft units. Whilst for spacecraft around the earth the most common and practical source of electrical energy is a solar generator, for interplanetary spacecraft the different distance to the Sun can make this solution either difficult or impossible to utilize.

When the distance to the Sun increases, i.e., for missions to Mars and beyond, the electrical power that can be produced by a solar generator decreases with the square of the distance. Table 7.6 shows typical figures for solar electrical power at the average distances from Sun of some of the major planets.

Current solar generators technology is capable of providing sufficient energy to provide the required power to an interplanetary probe at distances up to the orbit of Jupiter (around 5–6 Astronomical Units from the Sun). Rosetta is the current solar powered spacecraft that has flown at the largest distance from the Sun (about 5.3 AU reached in October 2012). The NASA Juno probe, currently on its way to the Jupiter system, will reach similar distances in the coming years, as well as the planned ESA Juice probe, to be launched in the next decade to explore the Jupiter icy moons Callisto, Europa, and Ganymede.

Beyond the Jupiter orbit the only method currently existing and utilized to power interplanetary probes is the use of Radioisotope Thermoelectric Generators (RTGs). An RTG utilizes a radioisotope (e.g., Plutonium-238, normally utilized in the form of Plutonium Oxide, PuO_2) to produce thermal energy and then convert it into electrical energy via thermocouples. The advantage of this type of generator is of course the independence from Sun distance, but also from the attitude of the spacecraft with respect to Sun. This simplifies the attitude and orbit control of the probe, and it allows a higher robustness to failures. The disadvantage is the presence of a radioactive source on board, which has system design, safety, and political implications which are difficult to manage and overcome.

Also at closer distance to Sun the solar generator may become a problem. This is due to the fact that it must be—by definition—exposed to Sun illumination, and therefore to its radiation and thermal flux. At distances of the order of the orbit of Mercury, the technological challenges related to the selection of the materials and the operation of the solar array under extreme solar fluxes become extreme. This is one of the main challenges in the design and operation of the future ESA missions to Mercury (BepiColombo) and to Sun (Solar Orbiter).

7.3.2.3 Communications with Earth

Another major aspect of interplanetary flight, also related to energy, is the communications with Earth. While for spacecraft orbiting the earth even the use of a low gain, omnidirectional antenna is sufficient to maintain a bi-directional link with ground for telemetry and telecommanding, an interplanetary probe requires medium and high gain antennae on board (and much larger antennae on ground) to establish the necessary communications. The problem again is the limited amount of energy available on board for transmission of the radio signal to ground, and the enormous distances to be covered to have the signal reach the Earth stations. In order to limit the dispersion of energy the radio beam has to be concentrated to the maximum extent, and this is achieved using highly directional antennae. This of course poses a fundamental challenge to the spacecraft design and the operations, which is the accurate and continuous pointing to Earth of the high gain antenna (also in case of contingencies). Complex mechanisms and/or pointing algorithms have to be adopted to ensure the continuity and reliability of this vital link to ground. A perfectly functioning and Sun pointing interplanetary probe is totally useless if the high gain antenna is not pointed to Earth within a small fraction of a degree. Figure 7.27 shows the ESA Rosetta probe and its various onboard antennae, which include Low Gain Antennae (LGA) in S-band, Medium Gain Antennae (MGA) in S and X band, and a large, 2,2 m High Gain Antenna (HGA) supporting both S and X radio frequency bands.

Due to the energy limitations the achievable bitrates are also limited (typically to ranges between a few bits per second to a few hundreds of kbits per second). The achievable bitrate is also proportional to the frequency of the carrier radio signal. For interplanetary probes the most used frequencies are in the S-band (2 GHz) and X-band (7–8 GHz). The recent use of higher frequency bands (e.g., Ka-band, 18–30 GHz) has allowed higher bitrates, also in the range of Mbits per second. However, the use of higher frequency bands carries additional operational complications, e.g., the higher sensitivity to water vapor in the Earth atmosphere and therefore to the local weather conditions.

On ground, the stations utilized for deep space communications employ large antennae, with a diameter of typically around 25–65 m, like for large radio-telescopes for astronomy. ESA for instance uses a network of three antennae of

Fig. 7.27 Rosetta onboard antennae

Fig. 7.28 The ESA Deep Space Antenna (35 m diameter) in New Norcia (Western Australia)

35 m diameter, located in Australia, Spain, and Argentina (see Fig. 7.28). NASA has also three complexes, in Australia, Spain, and California. In each complex there are several 26 m and 34 m, and one 70 m antenna.

7.3.3 Mission Control Approach

Operations of an interplanetary mission are not only very different compared to operations of Earth-bound spacecraft but also strongly dependent on the type of mission. On the other hand there are factors that characterize the operations concept and approach that are common to all types of interplanetary missions.

7.3.3.1 Specifics of Interplanetary Flight Operations

Long Signal Propagation Delay

Interplanetary spacecraft travel at very large distances from Earth, and the travel time of the radio signal from and to the spacecraft is not negligible like around Earth. Travelling at the speed of light the radio signal takes 8 min to cover the distance between Earth and Sun (called Astronomical Unit, AU, equivalent to approximately 150 Mkm). For instance, spacecraft orbiting Mars are subject to signal propagation delays that vary between 4 and 20 min one way (depending on the relative position of Earth and Mars in their heliocentric orbits). A spacecraft at Jupiter experiences a propagation delay of the order of 40–50 min. At Saturn, this increases to about 1.5 h. Such large delays in the radio signal propagation practically prevent any real-time interaction between the ground and the spacecraft, and in particular any activity that involves human real-time decisions based on the spacecraft response to telecommands.

Long Cruise to Target

Interplanetary mission profiles are usually characterized by long cruise periods in the interplanetary space, during which the spacecraft activity is reduced to a minimum, interrupted by short periods of intense activities, e.g., around a planet swing-by or an asteroid fly-by. An interplanetary probe is visible from a single ground station over typically 12 consecutive hours (for a spacecraft on the ecliptic this varies seasonally between typically 9 and 14 h). This is due to the fact that at large distances, the motion of the spacecraft in the sky as seen from Earth is very slow (i.e., the spacecraft can be compared to a "fixed star" over the duration of one station pass[1]), thus the dominating factor constraining visibility is the rotation of the Earth. However, it is common practice, in order to save cost, to minimize the

[1] In fact, depending of the distance to Earth, the apparent motion of the spacecraft in the sky is not completely negligible, as it is typically of the order of a fraction of a degree over the duration of a ground station pass (typically 12 h). This has to be taken into account when calculating the antenna pointing angles: in particular the movement of the spacecraft over a period equivalent to twice the signal propagation delay is important, as it affects the determination of the optimal pointing.

frequency of the ground contacts during a quiet cruise, typically down to one contact per week. This implies of course that the spacecraft is able, under nominal circumstances but also in case of unexpected failures or external events, to survive without ground contact for a period of the order of a week. Also the level of activities has to be low enough to allow all communications with the spacecraft (uplink of telecommands, downlink of stored information) to be completed in a single weekly contact period.

However, the quiet cruise periods, which can last several months to years, are usually interrupted by mission critical events like planets swing-bys, which require intense preparation and operations execution effort over a short period of a few weeks. This variable mission profile affects the composition and profile of the mission control team, which has to be as small as possible during the quiet cruise, but able to cope with sudden workload peaks during critical cruise events. The long duration of the mission, and in particular of the cruise before reaching the target and starting the actual “productive” phase of the mission, present severe managerial challenges. First of all enough flexibility in the size of the team has to be ensured during cruise, to cope with planned and unplanned critical events but at the same time minimize the cost of a large operations team. This can be achieved relatively easily if resource sharing with other similar missions in different execution phases is possible within the same center. Another challenge is keeping the motivation and the knowledge base in the team throughout the low activities cruise, to preserve it for the future science operations at the target. Very important to tackle this problem is the organization of a plan of cross training and proficiency training activities. Also the existence of high fidelity tools for training (e.g., software simulators or spacecraft engineering models) is essential to support this type of activities. Motivation can be stimulated by postponing some development (e.g., ground software tools or onboard software maintenance activities) from the traditional prelaunch phase to the cruise phase. Finally a significant problem is presented by the natural turnover in the flight control team composition. This can be mitigated by offering career opportunities within the same area (perhaps by moving temporarily to other missions in a different role, to avoid sudden and uncontrolled losses of expertise when people decide to change job on their initiative.

Long Periods Outside Ground Contact

An interplanetary probe has to be able to autonomously sustain long periods—of the order of several weeks—without ground contacts. During cruise this is often due to the decision to reduce operations cost, thereby imposing infrequent ground contacts, i.e., typically once per week to once every few months. In these periods the spacecraft has to be configured into a low activity mode and rely on its onboard autonomy to react to any anomalous situation. However, also during normal science operations an interplanetary probe will enter periods of non-visibility, caused by the conjunctions between Earth and Sun in the direction of the spacecraft. As shown in Fig. 7.29, when the spacecraft, as seen from Earth, comes to an angular distance of a

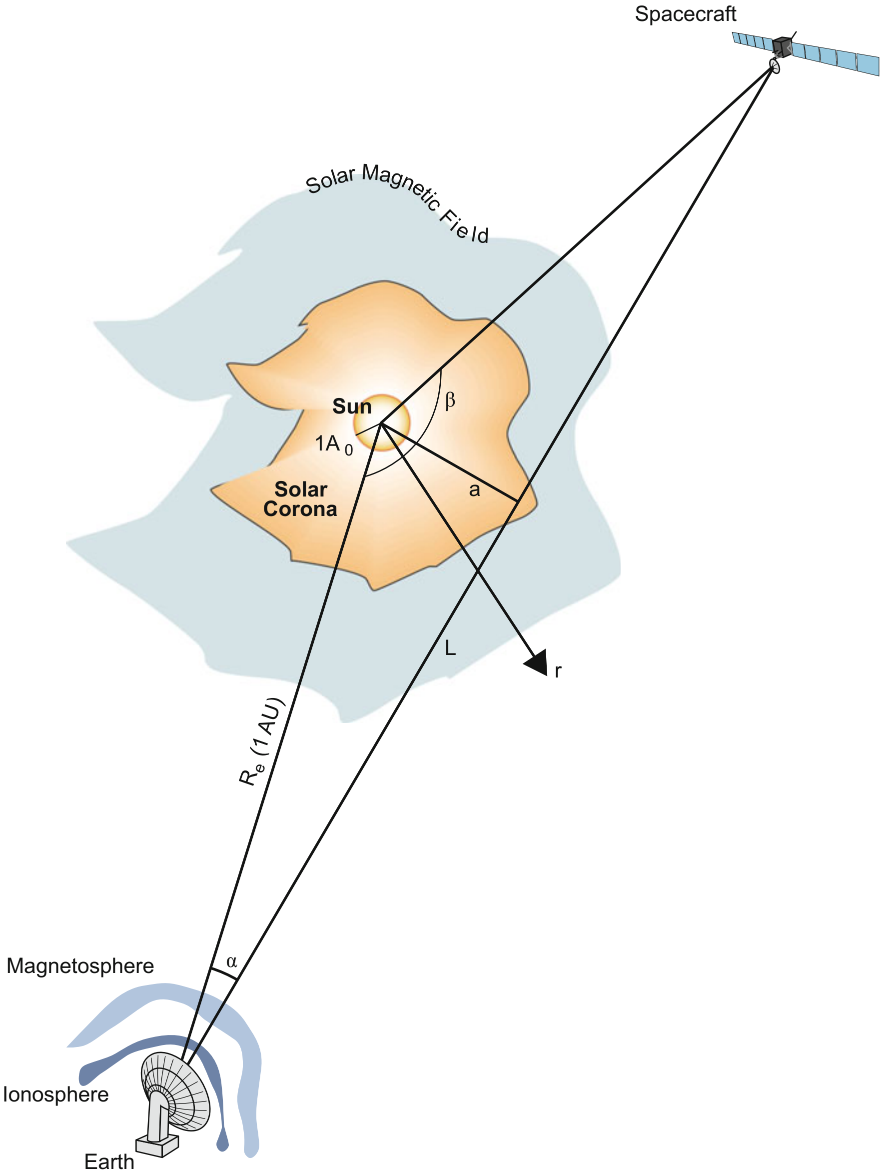

Fig. 7.29 Solar Conjuncion (superior)

few degrees from Sun, the effect of the solar corona starts affecting the quality of the radio signal, and therefore impose limitations to the telemetry and telecommanding activities.

There are in fact different types of conjunctions, depending on whether the spacecraft is between Earth and Sun (inferior conjunction) or beyond Sun (superior conjunction). Impacts on the radio link also have to be taken into account during solar oppositions, i.e., when Earth is between Sun and spacecraft. In general the radio signal starts being affected below 10° Sun–Earth–Spacecraft angle (indicated with alpha in Fig. 7.29). Below 5° the navigation accuracy is severely affected and below 3° the Telemetry/Telecommand (TM/TC) link may be totally disrupted. Interplanetary missions have therefore to be planned such that activities over these periods which, depending on the trajectory, can last several weeks, take the link interruptions and degradations into account.

High Spacecraft Vulnerability

An interplanetary spacecraft is by nature much more vulnerable to problems, whether caused by failures or operational mistakes, than Earth-bound spacecraft. All spacecraft, at least the solar-powered ones, require continuous and reliable pointing of their solar arrays to Sun. This is the only vital pointing aspect the spacecraft has to ensure. Interplanetary probes require, in addition, accurate pointing of their high gain antenna to Earth. Any problem causing loss of Earth pointing will prevent any communications from and to ground, thereby leading to loss of mission. Other vulnerability factors are trajectory correction manoeuvres, which are almost always time critical and do not allow a second opportunity. Even a temporary inability of the spacecraft (e.g., due to safe mode triggering) or the ground (e.g., loss of a ground station) in a critical moment such that a trajectory correction manoeuvre is missed may directly lead to the total loss of mission. The same is true for critical mission events like planet orbit insertion or entry descent and landing. Everything on the spacecraft and on ground has to be ready and perfectly functioning at a specific time: no delay in preparation can be tolerated, and the occurrence of a single, minor problem that impacts the timing of critical activities can lead to mission loss.

Complex, Variable Navigation and Attitude Control Operations

Interplanetary missions require extremely complex and accurate navigation activities. Special orbit determination techniques are required to increase the accuracy, in addition to the standard radio frequency ranging and Doppler tracking: this includes delta-differential one-way ranging (Delta-DOR, see Sect. 7.3.2.2 below). For navigation in proximity of small bodies, like asteroids or comets, whose trajectory cannot be determined with sufficient accuracy via optical or radar observations from ground, relative optical navigation techniques have to be used: onboard cameras take pictures of the target body, which are then processed on ground and used in the orbit determination process.

Attitude control of an interplanetary spacecraft is also more complex than a typical Earth-bound spacecraft. This is due to the large variety of attitude modes that have to be used during the mission, to cope with geometrical constraints during cruise, to execute remote sensing observations of the target body during the science phase, and to perform special pointing to celestial objects, or automatic tracking of small objects during fly-bys. Ground control has to cope with several different attitude control modes, manage the spacecraft momentum for future complex pointing profiles (e.g., to predict and control the level of momentum loading of the reaction wheels for attitude control), and design strategies over long periods to avoid unnecessary and propellant consuming mode changes.

Limited Knowledge of the Target

Even after 50 years of exploration of our Solar System with robotic spacecraft, the knowledge of the planets and all the small objects that populate the space around Sun is still very limited. The first interplanetary spacecraft (e.g., the NASA Mariner series) launched towards the planets had to cope with the very inaccurate knowledge of the position of the planet itself (of the order of hundreds of km!). Nowadays this problem only exists with asteroids and comets, but planets maintain their challenges when it comes to landing on their surface, for which the knowledge of the atmosphere density, variations, and the surface roughness is still extremely inaccurate and limited. Flying around small bodies also presents high risks due to the unknown environment in their close vicinity, which could include boulders orbiting the asteroid, or dust and gas jets around a comet. Spacecraft design and operations approaches have to take into account this high uncertainty on the characteristics of the target. In most of the cases the operations strategy has to be reinvented or at least adapted to the actual environment during the mission, when the spacecraft reaches the target and starts with its instruments the collection of vital information about it. An extreme example is the case of the ESA mission Rosetta, which will for the first time in history orbit a comet nucleus in 2014. Before the spacecraft will be able to select a landing site and design the operations strategy for delivering its lander module, Philae, onto the surface of the nucleus, it will have to spend several weeks in the proximity of the unknown target, mapping the surface and modeling the gravity potential of the comet. A full “engineering model” of the comet and its environment, including dust density and speed, will have to be built in these early phases of the comet proximity operations.

7.3.3.2 Ground Contact Activities

As mentioned above, the ground station contact for an interplanetary mission is constrained by the earth rotation rather than the spacecraft movement in the celestial sphere, which is negligible over the duration of a communication pass. This results in a theoretical visibility of 9–14 h, depending on the season (interplanetary spacecraft trajectories do not normally deviate too much from the ecliptic).

However, if the spacecraft is in orbit around a planet, the geometrical visibility may be interrupted by occultations (i.e., when the spacecraft flies behind the planet as seen from Earth) or by spacecraft constraints (e.g., transmission cannot be activated in eclipse due to power limitations, or the high gain antenna cannot be directed to Earth during scientific observations due to pointing constraints). Thus the daily ground station contact becomes a period of short windows, of the order of tens of minutes up to hours, during which contact with the spacecraft can actually be established. The examples of Venus Express, currently flying around Venus with a 24 h orbit period and Mars Express, flying around Mars in an orbit with about 6.5 h period, show how variable the type of contact can be. In the case of Venus Express the spacecraft orients itself to the planet in a specific part of the orbit to take scientific measurements, and then turns the high gain antenna to Earth during the period of the orbit in which the ground station is in visibility. This results in a very comfortable and repeating pattern in which the full duration of the ground station geometrical visibility can be effectively used without interruptions. Also Mars Express turns every orbit alternatively to the planet and to Earth for communications, but the orbit period of 6.5 h is not synchronized with the daily station visibility. Also planet occultations occur more frequently and power constraints force a careful operation of the transmitter, excluding eclipses. This results in a very irregular pattern of short ground contacts, typically of the order of 30–120 min, interrupted by non-contact periods of the same order of magnitude. The short duration of the contacts, combined with their irregular frequency, make the planning of operations (in particular commanding and download of recorded telemetry) extremely complex. Also, the rules for recovery from temporary outages or failures are not straightforward, making the replanning tasks complex and slow.

An extreme case will be typical for missions with even shorter orbit period (in the order of 2 h), such as ExoMars (which will be orbiting Mars in 2017 on a very low circular orbit of 350–450 km altitude) or BepiColombo (which will be orbiting the small planet Mercury in the next decade with an orbital period of about 2 h). For BepiColombo, for instance, the analysis shows that over the duration of a daily ground station visibility window up to 10 interruptions may occur, and the duration of continuous contact between two interruptions may be as short as about 20 min. Figure 7.30 shows an example of the BepiColombo visibility pattern during a ground station pass. The complexity of the telemetry and telecommanding activities induced by each link blockage is shown in the lower part of the figure: events on board and on ground have to be synchronized using two time references which differ from each other by the length of the signal propagation delay. As this delay is variable during the mission, tools are required to be able to plan and work with the two time references and ensure synchronization. For instance, if the start of the dump of telemetry data recorded on the onboard mass memory is programmed on board via time-tagged command, the time stamp has to take into account the signal propagation delay to reach ground at the beginning of the pass.

Other more complex examples are related, e.g., to the dump suspension when the first uplink sweep is started: shortly after the spacecraft signal is acquired at the beginning of a ground station pass, the station transmitter is activated and the radio

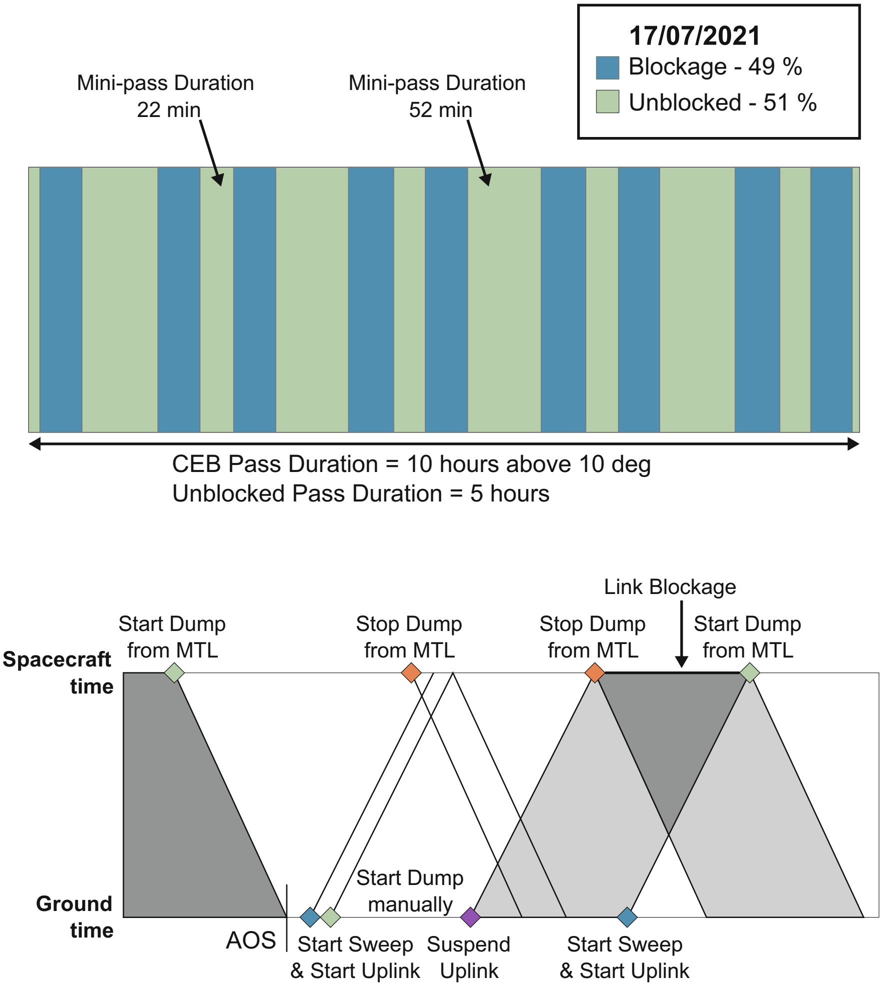

Fig. 7.30 BepiColombo ground contact profile for a single pass

frequency of the uplink signal is slowly changed up and down within a predefined range (this technique is called "uplink sweep"), to facilitate the automatic lock of the onboard receiver onto the received signal. But since the transponder on board is normally configured in coherent mode (see Sect. 6.1), once the onboard receiver locks on the uplink radio signal it will automatically change and move the downlink radio signal frequency. This causes a short interruption of the downlink reception at the ground station. Therefore an interruption of the onboard memory dump has to be programmed at this point to avoid a temporary loss of data.

The pass activities on the other hand are relatively simple, and mostly concentrated at the beginning, shortly after AOS (Acquisition of Signal, i.e., the moment in which the spacecraft rises over the horizon at the ground station and the first radio signal is received).

At AOS the first engineering telemetry is received and a quick check of the spacecraft status is performed. Once the antenna has reached 10 elevation, which is the safety limit for starting the uplink signal transmission, the ground station transmitter is activated and uplink sweep starts. It should be noted that the uplinked radio signal from ground will reach the spacecraft only after the one-way signal travel time, and the confirmation of onboard receiver lock will be received only after another signal travel time delay. This is normally too long compared to the duration of the pass. So the commanding activities start immediately—success oriented—after the end of the uplink sweep (i.e., a few minutes after the station transmitter activation), without waiting for positive onboard receiver lock confirmation. In case the sweep is unsuccessful, the ground will realize it only later and the whole operation has to be repeated. This leads to additional waste of time, but as it happens relatively rarely, the success-oriented approach is preferred over the safest but slowest one of systematically waiting for positive uplink lock confirmation. Commanding is normally driving the dump of event logs and other telemetry storage areas, which have recorded on board the telemetry accumulated during the noncontact period. Depending on the amount of science data stored and on the length of the signal travel time, the science telemetry dump command may be stored time-tagged on board, to increase the total dump time during the pass.

Once the initialization activities have been completed, the entire duration of the pass is normally used just to retrieve the stored telemetry. The dump is suspended automatically by onboard command for the occultations and at the end of the planned contact period.

For long contact periods normally operator supervision is required for the initialization activities (that is around the first hours after AOS start). For the rest of the pass the dump operations do not normally require human supervision. Of course if special commanding activities are planned, this may require presence of the operator or of relevant engineering expertise. In general, though, the pass activities for an interplanetary mission are simple and relatively easy to automate. Human intervention is only required in case of anomalies, and even in this case the intervention does not have to occur within a very short time: interplanetary spacecraft are designed to survive for long periods and the rapidity of ground intervention is only driven by the need to minimize the possible interruption of the data recovery or loss of data in general, not by the safety of the spacecraft, which is ensured by the onboard autonomy.

The frequency of the ground station passes during cruise is driven by the amount of activities required on the spacecraft, but also in particular by the amount of engineering telemetry (in case of passive cruise) to be recovered. The typical useful engineering telemetry generation rate of a spacecraft in cruise should not exceed a few hundreds of bits per second. However, even at this low rate the amount of telemetry accumulated over, say, a week out of contact is large. Combined with the low downlink bitrates available to an interplanetary spacecraft traveling at large distances (normally of the order of a few thousands of bits per second), it may require several hours to downlink the entire history of accumulated engineering telemetry.

For a mission like Rosetta the weekly pass frequency was mainly dictated by the wish to recover, in a 10 h contact, the entire history of engineering telemetry. If this is not possible, either because the available downlink rate is too small, or because the frequency of passes is lower, then the spacecraft may be configured to downlink only essential telemetry, and store the rest to allow the ground to selectively downlink whatever information is deemed necessary on each occasion. This approach is more complex and requires a quite flexible storage and downlink mechanism to be implemented on the spacecraft.

Some missions may decide to downlink a simple "beacon" tone to the ground. This is a Radio Frequency (RF) carrier with a limited set of specific frequencies (each one is called a "tone"), which can be picked up even with relatively small antennas. The onboard autonomy decides which tone to downlink, to indicate, e.g., that an anomaly occurred or that everything is OK. The ground periodically checks the beacon and, depending on the received tone, may decide to establish a full link and acquire the complete engineering telemetry. This method is being used, e.g., by NASA's New Horizons, which is due to fly-by Pluto in 2015. This is even more complex and requires a full trust in the onboard autonomy, combined with a large onboard storage and a flexible store/downlink mechanism. However, for missions flying for many years at very large distances, this may be the only way to save precious ground station time and therefore mission cost.

7.3.3.3 Trajectory Determination

Interplanetary orbit determination is a very complex task that requires high accuracy and often is linked to critical operations and therefore directly to mission success. The main aspect that characterizes interplanetary orbit determination is the extremely high accuracy required, which implies high precision in all the involved measurements. For instance, an error of 1 m in the knowledge of the position of the ground station taking the radiometric measurements directly results in an error of 750 km in the spacecraft position determination at Saturn distances. The same is true with the precision of the clocks used to measure the radio frequencies involved in the radiometric data collection. An additional complication is the accuracy of the position of the solar system target, planet or asteroid, which becomes critical when approaching it. For instance the position of an inner planet like Mars is known today with an accuracy of 0.5 km, whilst the distant giant planets like Jupiter or Saturn suffer inaccuracies of the order of 10 km. The trajectories of small bodies like asteroids and comets are known with much less accuracy, resulting in errors of the order of 100 km.

The main methods used for orbit determination are the same as for Earth-bound missions: ranging and Doppler measurements. Ranging is based on the transmission of a sequence of tones to the spacecraft, which receives and retransmits them in its downlink signal. By measuring the time of flight of the signal the radial distance of

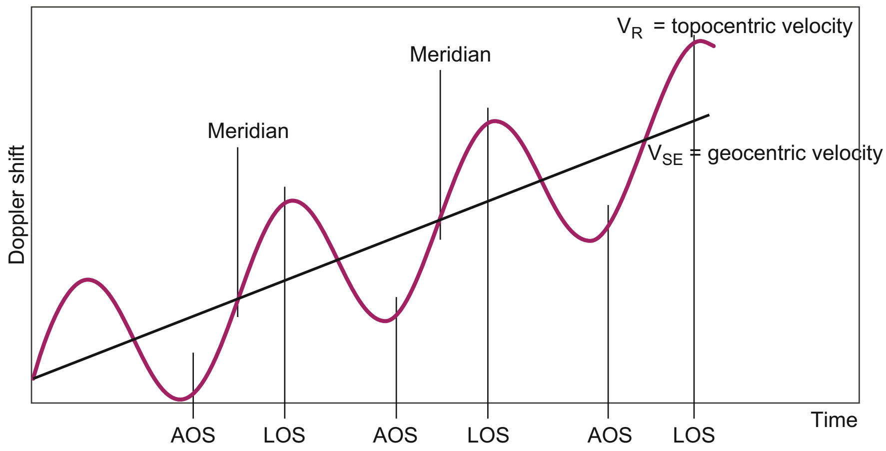

Fig. 7.31 Main components of the Doppler measurement

the spacecraft from the ground station can be accurately measured. Doppler is based on measuring the difference between the uplinked frequency and the downlink frequency, which is linked in a coherent transponder to the uplink frequency via a fixed (known) ratio. This allows determining the radial velocity of the spacecraft with respect to the ground station by analyzing the Doppler shift of the frequency.

The Doppler measurement can be utilized to achieve an accurate determination of the angular position of the spacecraft in the sky, as shown in Fig. 7.31 : Given that the main component of the measured Doppler shift is due to the ground station movement during the pass due to the earth rotation, the point in which this component is zero determines very accurately the point when the spacecraft crosses the local meridian (i.e., the local South direction).

Critical events like planet swing-bys or orbit insertions, which require extremely high spatial accuracy at a specific point in time, are nowadays supported by a new radiometric technique, called Delta-Differential One-Way Ranging (Delta-DOR). This method is based on interferometry measurements of specific spacecraft tones from two ground stations with simultaneous visibility of the spacecraft. The measurement is complemented by calibration of the local atmospheric delays by observing a quasar close to the direction of the spacecraft. This method, derived from radio astronomy Very Large Base Interferometry (VLBI) techniques, allows to achieve accuracy in spacecraft position determination one order of magnitude better than the one achievable with normal radiometric measurements. The principle is illustrated by Fig. 7.32.

The signal from the spacecraft is received by two ground stations. Due to the large distance between the stations on Earth, the same signal that is received at a specific instant in time has a different phase at each of the two stations (which is the same as saying that the same phase is received at slightly different times at the two stations). The difference in phase gives a precise measure of the different distance between each station and the spacecraft. By knowing the exact distance between the

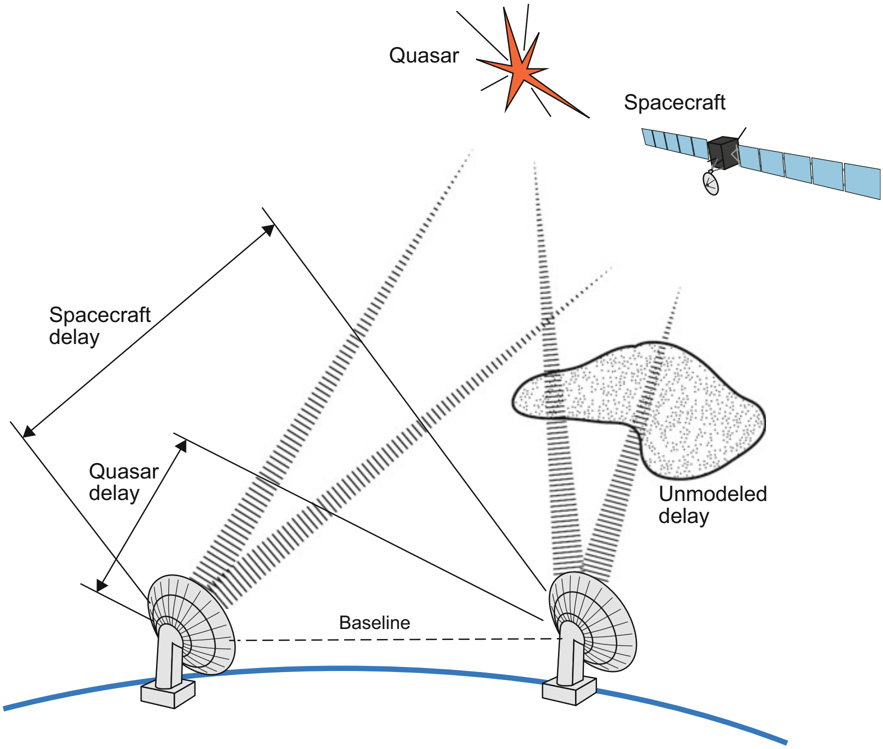

Fig. 7.32 Delta-DOR measurement principle

two stations and the angle between the stations and the spacecraft, a very precise measurement of the spacecraft position can be obtained. Also very important to emphasize, this technique allows to have an independent check of the resulting trajectory, eliminating potential systematic errors in the solutions.

To resolve the inaccuracy in the known position of a small target body, optical navigation techniques are used, through the acquisition of pictures from an onboard camera, which are elaborated on ground and included in the trajectory determination process. To illustrate this technique Fig. 7.33 shows the example of optical navigation applied to the Rosetta fly-by of asteroid Steins in September 2008.

The figure is based on the B-plane representation. The B-plane is defined as the plane perpendicular to the spacecraft trajectory (more precisely: the asymptote to the trajectory) that contains the target object (in this case the asteroid). The figure shows various ellipses with a cross in the middle of each. The cross represents the predicted point of crossing of the B-plane, i.e., when the spacecraft will be at the point of closest approach to the asteroid. The ellipse shows the bi-dimensional error (3 sigma) in the predictions. The error in the third dimension is represented by the inaccuracy in the time of closest approach. The largest error ellipse, shown in black,

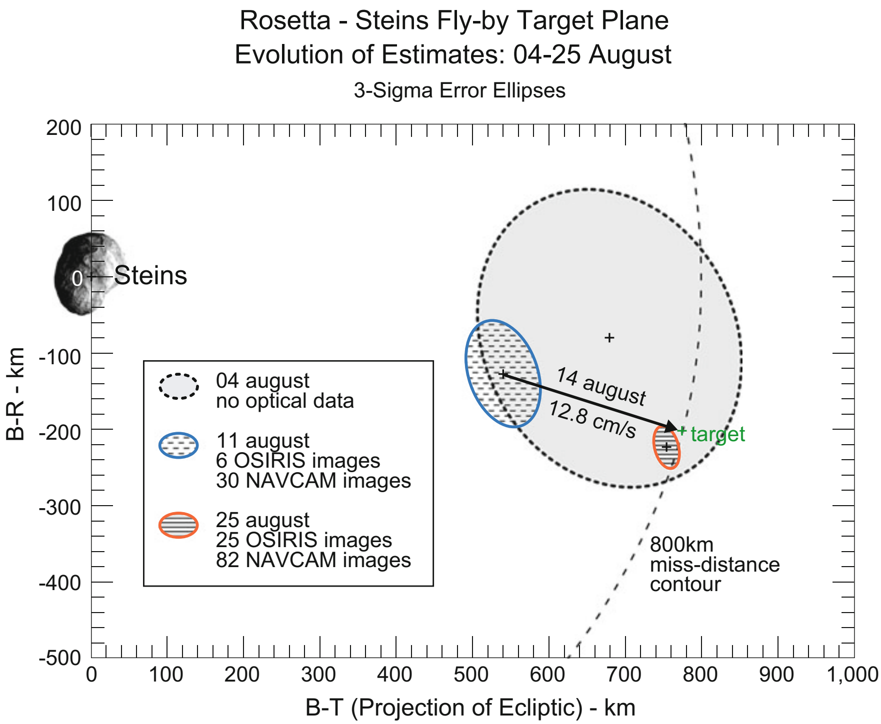

Fig. 7.33 B-Plane optical navigation for Rosetta asteroid fly-by

indicates the result of the orbit determination using only radiometric data. A sequence of pictures of the asteroid taken at 8.1 to 5.2 Mkm from the target shows the asteroid apparent movement relative to the background fixed stars. By analyzing this apparent movement the knowledge of the relative position between the spacecraft trajectory and the asteroid is enormously improved in the B-plane. Optical navigation allowed the reduction of the error ellipse (see the initial improvement of the error shown by the blue ellipse) and the planning and execution of more accurate trajectory correction manoeuvres, resulting in Rosetta missing the fly-by target point by less than 0.5 km.

Optical navigation has been tried as an autonomous process on board the NASA Deep Impact impactor spacecraft. The impactor was released by the mother spacecraft to impact on a comet nucleus about 24 h before the planned impact. As the relative speed of the impactor and the comet was about 10 m/s, the error in the impact trajectory became visible only in the last 2 h before impact, which made any intervention from ground practically impossible. Through its onboard camera the impactor determined its trajectory and performed three small correction manoeuvres, between T-90 min and T-13 min, finally crashing on the surface of the comet as planned.

Modeling of disturbance forces on the spacecraft is a very important part of the overall orbit determination process for an interplanetary spacecraft. Over the long cruise arcs disturbance forces like solar radiation pressure can have a significant effect on the trajectory, and its accurate modeling and prediction are particularly important at the end of the arc, when the encounter with a solar system body is planned.

Even more critical and complex is the modeling of the dynamic effects of a planet's atmosphere on the descent trajectory of a landing module. Small inaccuracies in this model can lead to enormous errors in the actual landing site. For instance error ellipses of the order of 100 km are normal for landing on Mars. The large variability and unpredictability of the density of Mars' atmosphere are one of the main problems related to designing a safe landing on the planet.

7.3.4 Special Operations

Interplanetary missions are characterized by a highly variable mission profile, with several complex and often critical operations which are unique of this type of missions. The sections below deal with the most important of these activities and the related operations approach.

7.3.4.1 Planet Swing-by

As already indicated in this chapter, planet swing-bys are the most commonly used method to acquire or release orbital energy, as they are practically free in terms of onboard fuel utilization. As such most of the interplanetary missions tend to make use of one or more swing-by (also called Gravity Assist manoeuvre, or GAM) to achieve their target. The disadvantage of such method is first of all the complexity of the interplanetary trajectory and the duration of the cruise, which increases significantly compared to a direct flight to the target. Also the operations around a planet swing-by are time critical and require high precision navigation and a robust Fault Detection, Isolation, and Recovery (FDIR) logic on board the spacecraft that prevents unplanned trajectory perturbations in proximity of the planet.

The activities related to a planet swing-by follow a relatively fixed pattern that begins a few months before the closest approach to the planet, with intense tracking campaigns to achieve the highest possible accuracy in orbit determination. Typically two ground station contacts per day from different ground stations, of the duration of a few hours each, are scheduled for the acquisition of radiometric measurements. In addition, DDOR tracks are scheduled with increasing frequency (starting from once per week up to several tracks per week). Typically 20–30 DDOR tracks are scheduled, depending on the criticality of the swing-by (e.g., the lower the swing-by altitude, the higher the criticality) in the last few months before closest approach.

Table 7.7 Generic activities timeline for a Solar Orbiter GAM

Time (relative)	Activity
GAM—1 month	Start of phase Delta-DORs: one per week Comm passes: two to four per week
GAM—30 days	Slot for Trajectory correction manoeuvre
GAM—16 days	Slot for Trajectory correction manoeuvre
GAM—2 weeks	Delta-DORs: two per week Comm passes: daily
GAM—1 week	Slot for Trajectory correction manoeuvre
GAM—1 week	Delta-DORs: four per week
GAM—5 days	Comm passes: 24-support
GAM—3 days	Slot for Trajectory correction manoeuvre
GAM—1 day	Slot for Trajectory correction manoeuvre
GAM—3 h	*Latest* slot for Trajectory correction manoeuvre
GAM	Closest approach
GAM + 1 week	Slot for Trajectory correction manoeuvre End of phase

Trajectory correction manoeuvre slots have to be planned as part of the planet swing-by phase. The first correction is normally performed at around 1 month before the swing-by. If the manoeuvre (typically of the order of tens of cm/s) is performed with high accuracy and the perturbation forces on the spacecraft accurately modeled, there are good chances that this correction will be sufficient to perform the swing-by within the precision required by the mission. Further trajectory correction slots are anyway planned typically 1 week and 1 day before the swing-by. As a safety measure, an emergency slot at T-6 h is scheduled, to perform a pericenter raising manoeuvre in case of late detection of major problems with the trajectory. Such manoeuvre would deteriorate the performance of the fly-by, but at least save the spacecraft from the risk of collision with the planet or reentering in the atmosphere.

As an example, the planned generic timeline of activities for each of the Solar Orbiter GAMs is shown in Table 7.7.

In order to avoid perturbations of the trajectory in the final weeks before closest approach, and in particular to minimize the chances of late disturbances to the trajectory that would force an emergency pericenter raising, the spacecraft must be designed such that its attitude control system does not create spurious forces. This "pure torques" attitude control is important when reaction wheel momentum is offloaded (required periodically to avoid saturation of the wheel momentum capacity) but in particular when the spacecraft enters into autonomous safe mode (with attitude normally controlled with thrusters): even a small spurious acceleration of the order of 10 cm/s/day can cause, over 1 week (the time that may be required to bring back the spacecraft into a normal attitude control mode), major deviations (of the order of tens of km) from the nominal trajectory at the time of closest approach to the planet.

In the period around closest approach normally the ground contact is lost due to either occultation by the planet itself or, in case of Earth swing-by, due to the high velocity of the spacecraft, which may be too fast for the tracking capabilities of the ground antenna. If the spacecraft is normally keeping contact via a steerable high gain antenna, the same limitations of angular speed may require to stop the movement and configure the link over fixed, low gain antennas over the few hours around closest approach. Also the fast dynamics of the few tens of minutes around closest approach may cause extremely high Doppler rates, which go out of the range supported by the onboard transponder.

There are other considerations that may force special configurations of the spacecraft for the few tens of minutes around swing-by. For instance the gravity gradient changes may impose high torques on the reaction wheels, forcing a change to a more robust thruster-based attitude control mode. Also the star trackers may be blinded by the planet, or other optical sensors may provide wrong inputs to the onboard software and must therefore be disabled.

As soon as the short period of closest approach is over, the spacecraft is reconfigured to its normal mode for interplanetary cruise. Intense tracking activities continue in order to determine the success of the manoeuvre. Typically a slot for a final correction manoeuvre is scheduled a few days after the swing-by.

The activities described in this section are valid for a generic swing-by of a major planet, separated in time from another swing-by or critical mission activity by several months. In case of missions that operate in the Jupiter or Saturn environment, and perform frequent swing-bys of the moons (for gravity assist or scientific reasons), the entire process has to be much more time-efficient, due to the fact that subsequent fly-bys are separated by very short periods of time (of the order of weeks). New strategies are currently under analysis for the future ESA mission to the Jupiter icy moons, JUICE, which will perform about 30 swing-bys and fly-bys of Europa, Callisto, and Ganymed in less than 3 years.

7.3.4.2 Asteroid Fly-by

Interplanetary probes often require to perform close fly-bys of small bodies as part of their scientific objectives. The fly-by of a small body like an asteroid or a comet is a completely different operation from the swing-by of a major planet described above. The main difficulty in this type of operations is caused by the limited knowledge of the target trajectory. Asteroids ephemerides are known with an accuracy of the order of 100 km. This means that the accuracy of the trajectory determination of the spacecraft itself is not sufficient to ensure a sufficiently precise asteroid fly-by distance and time.

To improve the precision of the fly-by distance in the B-plane optical navigation is used, as described in Sect. 7.3.3.3 above. Several trajectory correction slots are also planned in case of an asteroid swing-by. Normally the last slots are in this case always used, given the much higher requirements on the precision of the fly-by. Typically the last manoeuvre is executed less than 1 day before the closest approach.

Having solved with optical navigation the problem of steering the trajectory to the correct fly-by distance in the B-plane is not sufficient to achieve a successful fly-by operation: in order to point the instruments to the asteroid during the short time around closest approach (the most interesting period from a scientific point of view), the knowledge of the asteroid position along the line of flight has to be improved.

Orbit determination from ground may estimate the time of closest approach within a few seconds. This is not good enough to steer the spacecraft instruments during closest approach, when high relative velocities (of the order of 10 km/s) combined with relatively short fly-by distances (of the order of hundreds of km) impose a very high angular rotation of the spacecraft (or its instruments) to keep the target in the field of view. Two solutions have been adopted to ensure the proper observation of the target: either the spacecraft is preprogrammed to compose a mosaic of pictures to cover an area larger than the estimated error in orbit determination (thus ensuring that at least in a few of these pictures the target object will be captured), like in the case of the asteroid fly-bys made by NASA's Galileo spacecraft; alternatively the spacecraft uses its onboard cameras to autonomously detect the asteroid, and steers its attitude by trying to keep the asteroid in the field of view (by continuously updating its estimation of the time of closest approach). The latter technique has been used by Rosetta for its two asteroid fly-bys in 2008 and 2010, and by recent NASA spacecraft.

The operations related to asteroid fly-bys are concentrated in a few hours around closest approach. Being a one-shot activity everything has to work perfectly and therefore a lot of tests and validation activities are performed in the months preceding the event. The combination of all the scientific requirements coming from the various remote sensing instruments involved in the fly-by into a single, robust commanding timeline is another major challenge of such an operation.

7.3.4.3 Planet Orbit Insertion

The operations required for a planet orbit insertion are very similar to the ones necessary for a planet swing-by. The main difference is that, at arrival at the planet, normally a large orbit injection manoeuvre has to be performed with the onboard propulsion system. In order to minimize gravity losses,[2] this manoeuvre (typically of the order of a few km/s) has to be performed as fast as possible, so the spacecraft is equipped with a large engine (for instance, Mars Express and Venus Express have a 400 Newton thruster).

The orbit insertion manoeuvre normally occurs within a few tens of minutes before arrival at the planet, and lasts also several tens of minutes. Like all critical operations of interplanetary missions, it has to be executed autonomously, without any possibility for the ground to intervene in case of problems, due to the long signal propagation delays. Also, due to the pointing constraints of the spacecraft, it

[2] Gravity losses is a term to indicate the reduced efficiency of a manoeuvre if the manoeuvre is not performed fully in the optimal point in the orbit (driven by the "rocket equation"). An instantaneous manoeuvre would have zero gravity losses, i.e., its fuel efficiency would be perfect. Typical chemical propulsion systems require of course a finite time to perform the manoeuvre: the smaller their engine thrust, the longer the manoeuvre. The longer the duration of the manoeuvre, the larger are the "gravity losses," i.e., the fuel efficiency of the manoeuvre is smaller.

is normal that the high gain antenna cannot be pointed to Earth during the manoeuvre. If possible, a weak RF link is maintained through the low gain antennae. Even if this allows only RF carrier reception, this signal is extremely important to monitor the execution of the manoeuvre from ground, especially in case of anomalies. The signal Doppler shift can be used to compare it with the expected spacecraft acceleration profile, and any deviations from the expected manoeuvre performance can be rapidly detected, to an acceptable level of accuracy. Unfortunately also for planet orbit insertions it is quite common that part of the manoeuvre is performed with the spacecraft occulted by the planet itself. This reduces the visibility and makes the moment of reappearance behind the planet extremely tense for the ground controllers: the reappearance of the spacecraft signal at the predicted time gives a first rough indication that the manoeuvre was performed correctly and the mission control team can start breathing again.

Given that for many planetary missions the insertion into planet orbit is a mission critical activity (i.e., if it fails, the mission is lost), all decisions at mission level give priority to the execution of the manoeuvre w.r.t. even the spacecraft health itself. This means that the onboard failure detection and isolation autonomous functions may be partly disabled or relaxed during the execution of the manoeuvre, just to minimize the chance of small problems unduly interrupting or perturbing the manoeuvre execution. Also all non-essential units are deactivated, also to minimize chances of failure occurrence and propagation during this mission critical period.

After orbit insertion it is normally required to perform a set of orbit correction manoeuvres to move the spacecraft from the capture orbit to the final operational orbit. These manoeuvres are not directly time critical, as they can be performed with the periodicity of the achieved orbit. However, since the period of the initial orbit is normally large, it is important to perform them rapidly to avoid large delays in the start of the scientific operations phase at the planet.

Planet orbit insertion manoeuvres can be also less critical, like in the case of BepiColombo, which will inject into Mercury orbit performing a manoeuvre at the so-called weak boundary conditions. In these cases if the manoeuvre cannot be performed there is always another opportunity as the spacecraft remains in the proximity of the planet heliocentric orbit. This of course reverts the considerations on priority of spacecraft health w.r.t. manoeuvre execution.

7.3.4.4 Landing Operations

Landing operations are described in more detail in Sect. 7.4.

7.3.5 Conclusions

Mission control for interplanetary probes involves extremely complex activities, certainly amongst the most critical and complex in the field of spaceflight. The high accuracy and criticality of navigation, the time criticality of many mission operations, the high variability of the mission profile, and the fact that most operations have to be executed without the possibility for ground to intervene in real time are the main challenges to ground operators for this type of missions.

The ground tools required for interplanetary missions are also very specialized: the large ground stations, which include extremely sensitive instrumentation and radio-astronomy techniques; the flight dynamics algorithms and models, which have to achieve extremely accurate orbit determination and support complex pointing activities under very variable conditions; the mission control approach, which is mainly based on off-line interactions but has to include complex planning methods and extremely safe but flexible procedures systems.

The expertise required to perform end-to-end interplanetary mission control is consequently highly specialised and, given the relatively scarce amount of such missions, it is also rare to find and geographically concentrated in a few operation centres in the world.

After the pioneering decades (1960–1980) and the major successes of many world space agencies in the exploration of the solar system, the next steps will become more and more complex and require more costly missions. The key to the success of future interplanetary missions is international cooperation. The final target, a deeper understanding of our solar system, is the new frontier of human exploration.

7.4 Lander Operations

Stephan Ulamec and Paolo Ferri

7.4.1 Overview

A special case of interplanetary missions is the one including landers or entry probes. First because of the operational pre-requirements to allow the landing as such and second due to the particular constraints related to the operations of a spacecraft positioned on the surface of a planet, moon, or small body.

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There are several categories of missions, some of them to be discussed in more detail below. The design of landers differs considerably, depending on the actual target body. We distinguish between

(a) Landing on planets or moons with a significant atmosphere (Mars, Venus, Titan); with the sub-group of entry probes (where only the atmosphere is investigated during descent)
(b) Landing on airless bodies (Moon, also Mercury, or e.g., Ganymede)
(c) Landing on small bodies with very low gravity (asteroids, comets)

The descent to the surface can be performed with several strategies, notably by

(a) Landing the complete spacecraft, which is also used for the interplanetary cruise and remote observations (e.g., Near Earth Asteroid Rendezvous (NEAR) or Hayabusa)
(b) A dedicated lander (e.g., the Viking Landers or the Rosetta Lander, Philae)
(c) Probes that can position themselves after touchdown (e.g., Luna 9 or the MER Rovers)
(d) Penetrators

All of these have particular requirements from an operational point of view. Some concepts are described in further detail in Sect. 7.4.3.

Table 7.8 gives an overview of a selection of historic lander (and some atmospheric entry) missions.

Landing phases are critical and, as for all critical activities of interplanetary missions, have to be preprogrammed and executed fully autonomously. (We are not discussing here human missions where, as in case of Apollo, the landing is controlled by astronauts.) However, the contact with mission control on Earth during these phases is essential, to follow all phases of the automatic sequence and keep the possibility to investigate failures in case of problems. This is an important lesson learned from the catastrophic failure of NASA's Mars Polar Lander, which crashed in the 90ies onto the surface of the planet, but was designed to perform the landing operations "in the blind," so that the analysis of what went wrong was severely hampered by the lack of detailed information. Similar problems in failure analysis occurred after the loss of the Beagle 2 lander, which was released by the ESA Mars Express probe before entering into Mars orbit, 2003. The lander was scheduled to perform all entry, descent, and landing operations without contact with the orbiting spacecraft nor with ground stations on Earth and to establish the first radio link autonomously after successful landing. This never happened and a complete explanation of what problem could have occurred could not be established.

More recent landing activities have therefore featured all possible "observation points," from orbit (existing orbiters) and from ground (e.g., ground stations or even radio telescopes to detect at least the radio carrier in case the signal is too weak for a tracking station to decode telemetry from it). In the case of the latest Mars landing, NASA's MSL Curiosity, three Mars orbiters (Odyssey (NASA), Mars Reconnaissance Orbiter (MRO, NASA), Mars Express (ESA)), one ground station (New

Table 7.8 Selected Lander missions

	Country/ agency	Launch date	Comment
Moon			
Luna 9	UdSSR	Jan. 1966	First ever soft landing on an extraterrestrial body
Surveyor 1-7	USA	May 1966 to Jan. 1968	Series of five successful landings (Surveyor 1, 3, 5-7) on the Moon, in preparation of the manned Apollo missions
Luna 16	UdSSR	Sep. 1970	First robotic sample return mission from an extraterrestrial body. Followed by the successful missions Luna 20 and 24
Luna 17	UdSSR	Nov. 1970	Delivered Lunar rover Lunokhod 1 to the Moon. Followed by Luna 21 with Lunokhod 2. The rovers operated for several months and traversed 10,5 and 37 km on the Lunar surface
Apollo 11-17	USA	Jul. 1969 to Dec. 1972	Six successful manned missions to the Moon. Returned about 380 kg Lunar material to Earth and placed several scientific instruments on the Moon. Each Lunar Module (LM) was crewed with two astronauts. Apollo 15-17 included astronaut driven rovers
Mars			
Mars 3	UdSSR	May 1971	First soft landing on Mars. However, probe ceased to transmit data only 14 s after touchdown
Viking 1 & 2	USA	Aug./Sep. 1975	Two identical, highly successful Mars Stations. Operated on the Martian surface till 1980 and 1982, respectively
Mars Pathfinder	USA	Dec. 1996	Fixed landing station plus small Sojourner Rover. It demonstrated landing with airbag concept
Mars Exploration Rovers (MER)	USA	Jul./Jun. 2003	Two rovers (Spirit and Opportunity) landed, using airbag concept Spirit operated until 2010, Opportunity is still operational
Phoenix	USA	Aug. 2007	Landing in polar region (68°N)
Mars Science Laboratory (MSL), Curiosity	USA	Nov. 2011	Successful landing of rover by “sky-crane”

(continued)

Table 7.8 (continued)

	Country/ agency	Launch date	Comment
Venus			
Venera 4	UdSSR	Jun. 1967	First successful atmospheric probe at Venus. Transmitted data down to an altitude of about 25 km
Venera 7	UdSSR	Aug. 1970	First successful landing on Venus. Transmitted data till 23 min after touchdown
Venera 8-14	UdSSR	1972–1981	Successful series of landers transmitting data from the Venusian surface
VeGa 1 & 2	UdSSR	Dec. 1984/ Jun 1985	VeGa 1 and 2 delivered successfully Landers (and balloons) to Venus and then continued their flight to comet Halley
Pioneer Venus 2	USA	Aug. 1978	One large and three small entry probes to study Venusian atmosphere (One survived impact and continued to transmit data for about 67 min)
Jupiter			
Galileo Probe	USA	Oct. 1989	Entry probe to study the Jovian atmosphere
Titan			
Huygens	ESA	Oct. 1997	Part of the joint NASA/ESA Cassini/ Huygens mission to the Saturnian system. Huygens entered the atmosphere of Titan in 2005 and sent data during descent; kept transmitting data after touchdown
Asteroids			
NEAR-Shoemaker	NASA	Feb. 1996	Spacecraft not designed for landing. However, at the end of the orbiter mission NEAR could be safely placed at the surface of asteroid (433) Eros and continued to transmit data for about 16 days
Hayabusa (Minerva)	Japan	May 2003	Successful sample return mission from asteroid (25143) Itokawa Probe took samples by “touch & go” strategy. Attempt to deliver small Minerva lander failed
Comet			
Rosetta Lander, Philae	DLR and int. consortium, ESA	Mar. 2004	On its way to comet 67P/Churyumov–Gerasimenko. Landing foreseen in Nov. 2014

Norcia, ESA), and several radio telescopes were employed to follow the Lander's UHF and X-band signals emitted during the entire Entry, Descent, and Landing (EDL) phase. Even in the cases where the radio signal is not strong enough to decode digital telemetry, the reception of a RF carrier and the analysis of its Doppler shift allow the collection of a large amount of important information on the performance of the landing activities.

7.4.2 Landing Insertion

The insertion of a lander to actually descend to the surface of the target body or enter its atmosphere can be performed by either direct insertion from the incoming hyperbolic trajectory or via a set of maneuvers, in which the spacecraft first orbits the target body prior to insertion.

The latter is energetically less favorable, particularly for planets with an atmosphere; however, it relaxes operational constraints on timing, sensitivity to possible anomalies, or landing site selection/verification.

A release from orbit is the necessary choice in case the surface of the target body is not known well enough to preselect the landing site before launch or during the cruise. For instance, the NASA Viking Landers in 1976 arrived at Mars attached to the Orbiter. The assembly orbited the planet for about 1 month before the Landers were released and a de-orbit burn initiated the EDL (Entry-Descent-Landing) phase. The orbit phase was used to map the planet and finally select the landing sites, which indeed resulted in a change of both originally foreseen landing sites, since the analysis of the orbiter images proved that those sites were unsafe. Also the Rosetta-Lander Philae will be released from the ESA Rosetta Orbiter from an orbit (or hyperbolic flyby) around comet 67P/Churyumov-Gerasimenko after a comet nucleus mapping and characterization phase of several months, using the Rosetta orbiter instruments.

The option to perform landing via the incoming hyperbolic trajectory is more often chosen. Here, the spacecraft approaches the planet on a collision course. The lander module is released while still flying on this trajectory, and shortly after either the carrier spacecraft enters the atmosphere and is destroyed, or if the carrier spacecraft has to survive, it performs the orbit insertion maneuver. In the latter case—the most complex from an operational point of view—the relative timing of the lander module release and the orbit insertion of the remaining spacecraft is a crucial factor and a fundamental parameter in the overall mission and spacecraft design. In order to ensure the success of landing and orbit insertion operations also in case of contingencies, it is required to have at least a few days between the release of the lander and the start of the orbit insertion maneuver. This is due to the fact that in between, the carrier/orbiter spacecraft has to perform precise orbit determination, and also to take into account that the release operation may perturb the carrier/orbiter attitude and even cause safe mode triggering. The time for recovery from such contingency has to be accounted for planning of the orbit insertion maneuver.

Once released, the landing module (normally called Entry, Descent and Landing module, at least for planets with an atmosphere) is ballistically approaching the atmosphere of the planet. The start of the EDL phase is defined as the moment in which the module senses the atmospheric drag, called Entry Interface Point. From this moment onwards the spacecraft controls autonomously its attitude (keeping the heat shield in the direction of flight) and the sequence of EDL operations, normally based on accelerometers first and radar altimeters later. During the first phase the heat shield removes about 90 % of the dynamical energy of the module. Then the shield is released and a series of parachutes is deployed to gradually reduce the velocity further. For the final landing phase several methods have been used as described in the following chapter.

7.4.3 Various Landing Strategies

The exact landing strategy to be applied for a surface element is strongly depending on the actual target body as well as the particular mission requirements. EDL, obviously, is very different in case the planet has a significant atmosphere, where, e.g., parachutes can be applied—compared to airless bodies like the Moon. Small bodies like comets and asteroids are another dedicated category due to the low gravitational forces and the low velocities associated.

7.4.3.1 Entry and Landing Through an Atmosphere

Landing on bodies with a non-negligible atmospheric density like Mars, Venus, or Titan allows deceleration by atmospheric entry, using the drag of the atmospheric gases on the surface of the landing module. In this phase the spacecraft has to operate autonomously, since no interaction with ground is possible on the required timescales. Usually no autonomous back-up mode is designed for this phase, as recovery from major anomalies is not possible during this phase. The deceleration due to atmospheric drag is the most efficient method to reduce the kinetic energy of the descent module (for instance, on Mars this phase removes about 90 % of the kinetic energy). Protected by a thermal shield, which maximizes the aerodynamic drag and at the same time protects the lander from the aerodynamic pressure and the thermal load, the descent module decelerates to a speed that allows the opening of a parachute (or a set of parachutes), for further deceleration. Parachutes used today for spacecraft landing on Mars can sustain supersonic speeds (up to Mach 2.2).

The final phase of the landing scenario can use different methods, depending on the type of target body, of lander, and of operations to be conducted on the surface. For landing on Mars the first successful landers used retro-rockets that slowed down the landing module during the last hundreds of meters and allowed a controlled touchdown on landing legs. This was the case, e.g., of NASA's Viking landers in 1976, or the more recent Phoenix lander in 2008. The disadvantage of this landing technique is that the lander remains on the landing spot, which may also be contaminated or at least affected by the thrust of the landing rockets.

For the Pathfinder mission, which delivered the small rover, Sojourner, on the surface of Mars in July 1997, NASA successfully utilized a new landing technique, based on airbags that are deployed around the lander after the parachute phase and a short burn of retro-rockets to reduce the descent speed to a few m/s. The lander module bounces on the surface until it stops; then the airbags are deflated and the protecting shell is opened. After establishment of the initial contact and a rapid systems checkout, the rover is activated and can egress the landing platform and move on the surface of the planet. This technique was further successfully used for the landing of the two Mars Exploration Rovers Spirit and Opportunity in the first decade of this millennium. The airbag technique is not practical for a fixed platform like the one of Viking or Phoenix, since the deflated airbags remain on the terrain underneath the lander shell and prevent local access to the ground for investigations. On the other hand a landing on feet like Phoenix is not practical for rovers, since a complex egress system would have to be mounted on top of the landing gear to allow the egress of the rover after landing. Nevertheless such approach can be adopted in case the mission includes both a fixed landing platform and a rover module. This is currently the baseline for the joint ESA-Russia ExoMars mission that is schedule to land on Mars in 2019.

Finally, a new technique has been successfully employed by NASA in the recent landing on Mars of the Mars Science Laboratory called Curiosity. The technique, called "sky crane" involves the use of retro-rockets to control the descent and stop the lander module at an altitude of a few tens of meters above the surface. While the platform is hovering above the surface using the retro-rockets, the rover itself is lowered with a cable, like from a crane. Once it touches the surface, the cables are cut and the sky crane platform flies away and crashes at a distant place on the surface (Fig. 7.34). The rover Curiosity landed on its wheels and was ready for its first movements within a very short period (NASA http://marsrovers.jpl.nasa.gov/mission, 2014).

The thin atmosphere of Mars is per se a major challenge in the design of landing probes, since even small variations in its density can have dramatic effects on the efficiency of the two main tools used to dissipate orbital energy during the descent, the drag shield, and the parachutes. For this reason the altitude of the landing site and the seasonal and meteorological conditions of Mars at the time of landing play a major role in the mission and spacecraft design.

Landing on a body with a denser atmosphere is comparably simpler, as the atmosphere density variability plays a relatively smaller role. For instance, the atmosphere of Venus is so dense that no retro-rockets are required for soft landing. Parachutes (or even a drag shield) were sufficient to lead to impact velocities in the range of 7 m/s (as for Venera 9), to be damped by the landing system. The pressure of the atmosphere of Venus at ground is about 90 bar (mainly CO_2).

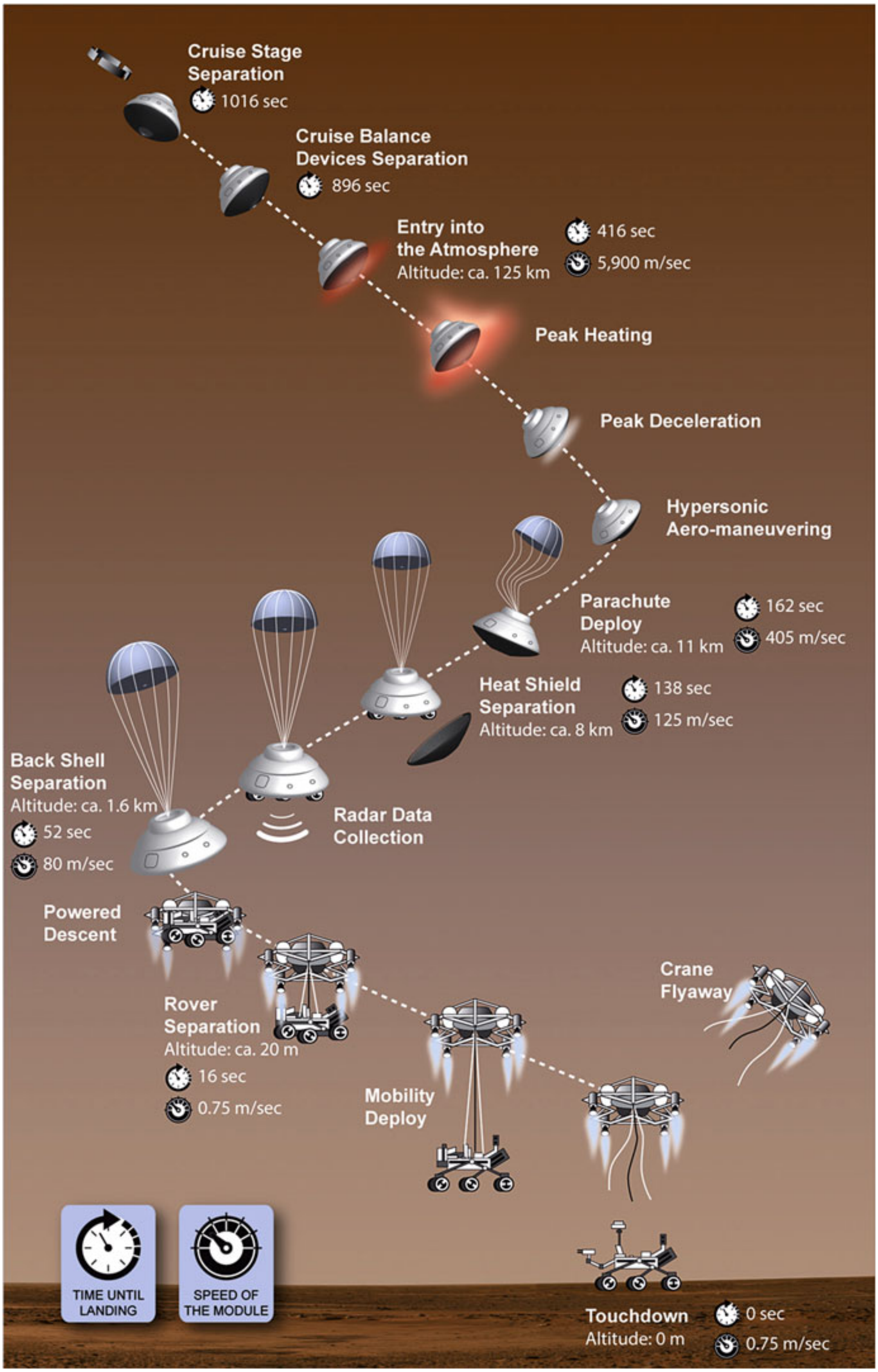

Fig. 7.34 Scheme of MSL-Curiosity landing

The first successful probe to enter an extraterrestrial atmosphere was Venera 4 in 1967. After the lander was separated from its transfer vehicle, it transmitted data of the Venusian atmosphere for about 96 min, until the batteries were empty at an altitude of about 25 km. However, in this case the scarce knowledge of Venus atmospheric density did have an impact on the mission: the density was estimated much lower than the one experienced by the descending spacecraft, resulting in an unexpectedly long descent and the fact that the surface could not be reached [the capsule either crushed or ran out of battery (Wesley and Marov 2011)]. All operations were performed fully autonomously.

Later, the design of Venera 4 was adjusted to the correct properties of the atmosphere and Venera 7 performed the first successful landing on Venus in December 1970. After a descent of 35 min the probe continued to send data from the surface for another 23 min. The Soviet Union continued the successful series of Venera spacecraft, keeping the same principal concept but applying modifications in the design and improvements of the payload (e.g., Venera 9 sent for the first time pictures from the surface in 1975).

The first Venera probes were transmitting data directly to Earth during descent and, in case of Venera 7 and 8, also after landing. Venera 9 to 14 included orbiters which were used to relay data from the landers.

Ground interaction was impossible for all Venera landers after entry has started (Fig. 7.35).

As in case of Venus, the density of Saturns largest moon, Titan, is high enough (1.6 bar at surface, mainly N_2) to allow the soft landing just with the help of a heat shield and a series of successive parachutes, as demonstrated successfully by the ESA Huygens probe in 2005. The main purpose of the probe was to perform atmospheric investigations, and therefore it was not designed to survive a landing

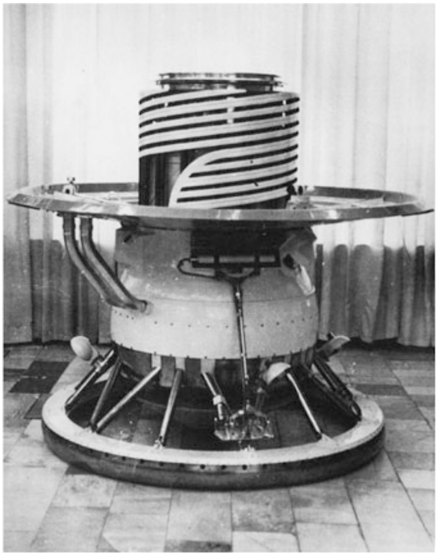

Fig. 7.35 Venera 9 Lander (image: NASA/Lavochkin)

on the surface. Nevertheless the probe touched down softly onto the surface of Titan and survived as long as its onboard batteries provided power, delivering pictures and measurements from the surface. All the telemetry data were relayed via the Cassini spacecraft, which had transported the Huygens probe to the Saturn system and had released it a few days earlier. The radio carrier signal of Huygens was also received directly on Earth by radio telescopes, which in real time measured its Doppler shift, allowing not only the detection of the successful landing but also complementing the atmospheric investigations done by the onboard instruments by measuring the velocity and acceleration profiles of the probe during the descent, thereby providing precious information on the winds field in Titan's atmosphere at different altitudes.

7.4.3.2 Landing on Airless Bodies: The Moon

Of course landing on an object with no atmosphere, like the Moon, is different, as neither a drag shield nor parachutes can be used, and the entire orbital energy has to be dissipated with retro-rockets. In comparison with Mars or Venus objects without atmosphere have a relatively low gravity, so the amount of energy to be dissipated is comparatively small.

The first successful soft landing on the Moon took place on February 3rd, 1966 by Luna 9. After direct approach to the Moon the landing sequence was initiated at a distance of about 8,000 km. A radar altimeter triggered the release of the cruise modules and initiated the breaking maneuver at an altitude of 75 km. Shortly before touchdown, a 5 m long boom detected contact to ground and caused the landing capsule to be ejected upward. The capsule eventually hit ground with a touchdown velocity of 15 m/s; the impact energy was damped with an airbag (Wesley and Marov 2011).

After a few bounces Luna 9 came to rest, the airbag was released and four petals which covered the top half of the spacecraft were opened, thus orienting the probe on the Lunar surface. Figure 7.36 shows an image of the Luna 9 lander with open

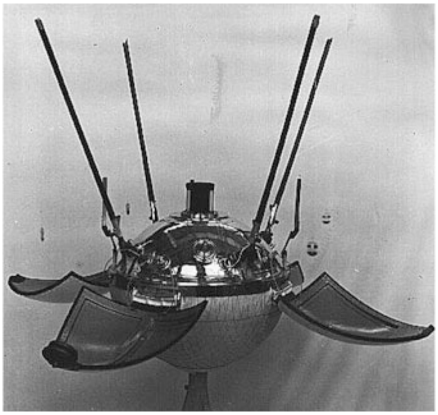

Fig. 7.36 Luna 9 with unfolded petals and deployed antennas (image: NASA/Lavochkin)

petals. The television camera was combined with a mirror system, which operated by rotating and tilting and provided panoramic images from the Lunar surface. Four panoramic views were transmitted during seven radio sessions, with a total transmission time of about 8:05 hours.

The probe could be commanded from Earth and survived about three days on the surface of the Moon (allowing images with different solar aspect angle) until the batteries were depleted. It is interesting to note that the signals used the internationally agreed system used by newspapers for transmitting pictures (FAX); the Jodrell Bank Observatory in England received the data and indeed, the images were first published by the Daily Express (and only 1 day later by Prawda). Luna 9 was followed by a very similar device with an enhanced payload, Luna 13.

Later Soviet Lunar Landers included automatic Sample Return (Luna 16, 20 and 24), even a 2.25 m depth drill-core (Luna 24) and the delivery of Rovers (Lunokhod 1 and 2, see Sect. 7.4.4.1). Those probes used a standard bus with retro-rockets for breaking before touchdown. The landing sequence ran automatic for all these missions.

In preparation of the manned Lunar programme by NASA, the unmanned Surveyor landers were sent to the Moon between 1966 and 1968. Five successful landings could be performed; the probes returned a plentitude of images and scientific data back to Earth. Apollo 12 landed very close to Surveyor 3, allowing a "visit" and the return of the probe's camera back to Earth (see Fig. 7.37).

The Surveyor spacecraft approached the Moon on a direct trajectory and slowed down to about 110 m/s by a solid fuel retro-rocket, which was jettisoned 11 km from the surface. The remaining descent was controlled by a Doppler radar and three vernier hydrazine engines. At an altitude of 3.4 m above ground, the thrusters were switched off and the landers descended in free fall leading to an impact velocity of about 3 m/s (Andrew et al. 2007).

In this chapter we will concentrate on the operations of robotic surface stations and, thus, do not explain the operations in the frame of the manned Apollo missions. However, it is to be noted that the Apollo astronauts installed so-called Apollo Lunar Surface Experiment Packages (ALSEP) near the landing sites, consisting of

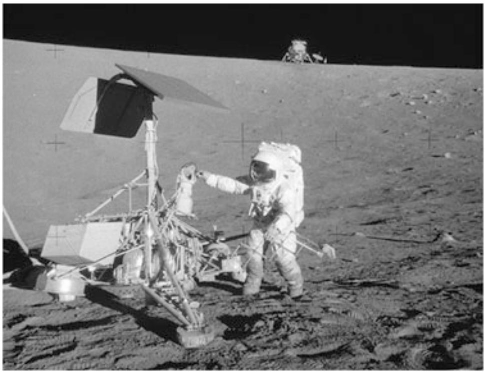

Fig. 7.37 Apollo 12 mission commander Pete Conrad at Surveyor 3; in the background the Lunar Module (image: NASA)

various individual units (including a Central Station containing a data management system, the communications unit and an antenna pointed towards the Earth by the astronauts, as well as a variety of instrument packages like, e.g., seismometers or particle detectors) all powered by a Radioisotope Thermoelectric Generator (RTG). Those ALSEPs were operational for several years until turned off for budgetary reasons in 1977.

7.4.3.3 Landing on Asteroids and Comets

Landing on small objects is a very different process. Due to the extremely low gravity, the landing is just a matter of imposing a small difference in the velocity of the lander compared to the one of the asteroid or comet. The difference is of the order of tens of cm/s, so the descent phase can last hours. The difficulty is the navigation in the proximity of the small body, and the required extreme accuracy of the involved maneuvers, as described in more detail in Sect. 7.3. Another difficulty is, once touched down, to avoid that the landing module bounces back into space. In the case of Philae, the small landing module that the ESA Rosetta probe will deliver onto the surface of a comet in 2014, a mechanism has been designed to shoot anchoring harpoons into the surface of the comet once the lander's touchdown is detected by sensors within the landing gear.

So far, landing on small bodies has been attempted only a few times (Andrew et al. 2007; Ulamec and Biele 2009): In the late 1980s the Soviet twin Phobos missions to Mars carried landers and hoppers which, unfortunately, due to early loss of the mother spacecraft, could not be delivered to the Martian moon. Japan's Hayabusa spacecraft tried to place a small lander, Minerva, onto asteroid Itokawa, but the maneuver failed and the lander missed the asteroid. However, NASA's Shoemaker-NEAR spacecraft, which orbited asteroid Eros and originally was not designed for landing, could be maneuverd to touchdown on the surface of the asteroid at the end of its operations phase. This maneuver was successful and the spacecraft managed to send from the surface a weak signal to Earth for about 16 days (Dunham et al. 2002).

Rosetta will be the next attempt to land on a small body. Its operation will be even more complicated by the fact that the surrounding environment of a comet nucleus is most likely much "dirtier" in terms of dust and gas compared to the one of an asteroid. This will make the navigation for the landing phase particularly challenging. The lander Philae is extremely complex, but a successful landing would make it the first real lander onto the surface of a small body (Biele and Ulamec 2008).

The largest difficulties described above, compared to landing on large moons or planets, are at least partially compensated, in case of small bodies, by a few advantages. First of all the landing systems do not require complex and heavy drag/heat shields, parachutes, retro-rockets. This is clearly a major advantage in terms of mass but also in terms of mission risk and complexity of the onboard autonomy. Another advantage is that, due to the slow relative velocity between the landing module and the target body, the landing trajectory and scenario can in general be planned such that a back-up opportunity can be guaranteed in case of last minute contingencies that force an abort of the landing operations.

7.4.3.4 Penetrators

A special case of surface elements are penetrators (e.g., Comet Rendezvous Asteroid Flyby, CRAF), hitting ground with high velocity and being decelerated in the surface material. Penetrators have been proposed for several missions, including applications on Mars (e.g., Mars 96 or Deep Space 2), the Moon (e.g., Lunar-A, MoonLite), or asteroids (e.g., CRAF). None of these missions has been realized successfully until now (Lorenz 2011). The advantages of such devices are the possibility to realize a relatively small and lightweight design, usually no breaking maneuver before touchdown is required and the fact that the subsurface can be reached by definition. However, a disadvantage is the high acceleration load during impact (typically in the 10^5 m/s^2 range) that makes it challenging to implement sensitive payload elements. Another problem arises due to the uncertainty in the knowledge of the surface properties, which causes difficulties to predict the actual penetration depth. From an operational point of view, penetrators suffer from the limited possibilities to apply high quality communications systems.

7.4.4 Surface Operations

Operations of a landed platform are challenging since in addition to the sometimes hostile environment, additional geometrical aspects have to be considered as compared to orbiters.

For instance the visibility is not only depending on the availability of terrestrial ground stations but also on the rotational state of the target body, the particular landing site, as well as the relative position of a relay orbiter. In addition day/night cycles have to be taken into account.

7.4.4.1 Operations of Stationary Surface Elements

The Viking landers were equipped to transmit information directly to Earth with an S-band communications system, or through the orbiter with a UHF relay system. The landers also received Earth commands through the S-band system.

One of two redundant receivers used a 76 cm parabolic high-gain antenna that required pointing to the earth. The second receiver used a fixed low-gain antenna to receive Earth commands. The UHF system was used almost exclusively during entry and the first days after landing.

The onboard computer had stored in its memory the sequence of the first 22 days after landing thus allowing autonomous operations without commanding from Earth. These sequences were updated as the communications link has been established.

Viking 1 eventually was lost in 1982 due to a faulty ground command (a software update intended to improve battery capacity erroneously overwrote the antenna positioning software) after 2,245 sols (Martian days) of operations. Viking 2 ceased operations in 1980 due to battery failure.

The NASA Phoenix lander descended to the polar region of Mars in May 2008. The landing site is at a latitude of 68° North. The probe was operational during Northern summer from August to November 2008, when, eventually, solar input became insufficient. In winter, the Martian polar ice cap was covering Phoenix. Any attempts to re-activate the lander in the following spring, in 2010, were unsuccessful.

Phoenix used UHF links, relayed through Mars orbiters during the entry, descent, and landing phase, and while operating on the surface, whereas during cruise, an X-band system was used. The communications system on the lander used the Proximity-1 CCSDS protocol as implemented for most Mars surface missions, relaying data via orbiters and was compatible with the relay capabilities of Mars Odyssey, Mars Reconnaissance Orbiter (MRO) (both NASA), and Mars Express (ESA). This international standard is extremely useful as it allows all missions implementing it to communicate between orbiters and landers, no matter which space agency they belong to. Overflights by NASA orbiters, which fly on a low circular orbit around Mars, are very short (of the order of minutes). During these contact periods the orbiter and lander establish the link automatically and after the lander has verified the link handshake procedure, it starts to send the data recorded by the lander during the previous noncoverage period. Mars Express was only providing back-up support in case of emergency. The highly elliptical orbit of the spacecraft allowed longer visibility periods, but the range of distances was larger and could not be covered entirely due to limitations in the power of the UHF transmitters and receivers. Also the spacecraft had to implement complex attitude profiles in order to point the UHF antenna in the optimal direction of the lander on the surface, in order to increase the efficiency of the radio signal.

During its 155 sol operations on Mars, the probe received regular commanding and performed a plentitude of scientific experiments, including the collection of soil samples with a grabber and analyses with a Thermal Evolved Gas Analyzer (TEGA) (NASA http://www.jpl.nasa.gov, 2014).

The Rosetta Lander, Philae, will be delivered to 67P/Churyumov–Gerasimenko applying a preloaded sequence that can only be worked out after arrival of Rosetta at the target comet and a dedicated comet mapping and characterization phase. The instruments aboard the orbiter (most prominently the camera, OSIRIS) will allow the production of a DTM (Digital Terrain Model) of the comet nucleus, determine the mass of the comet (possibly the higher harmonics of the gravitational field), and characterize the gas and dust environment in the coma. This information is essential in order to choose the delivery orbit of Rosetta from which, at a preprogrammed time with a predefined separation velocity, the lander will be ejected and descend to the surface. There is considerable flexibility in the system, however, once initiated,

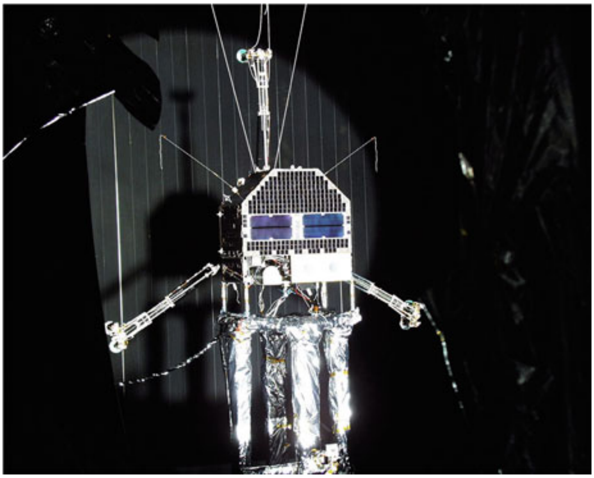

Fig. 7.38 Rosetta Lander during thermal vacuum tests in chamber at IABG

it has to run fully autonomously. The landing takes place at a heliocentric distance of 3 AU.[3] Due to the even larger geocentric distance, the RF signal travel time to and from Earth is more than 30 min one way. At touchdown, the Lander will fire anchoring harpoons and start a so-called First Science Sequence (FSS), based on the use of a primary battery and lasting for about 60 h. Although the sequence will be pre-loaded, in this phase there is the possibility for interaction with ground. Certain activities (like the actuation of a drill) require GO-NOGO decisions from ground; parameters can be changed and the timeline modified if required. After the FSS the so-called Long Term Science (LTS) phase starts, relying on a solar generator and a secondary battery and based on detailed commanding and preparation from ground. The sequences of commands that will be uploaded to the lander for automatic execution of the surface operations need careful planning to optimize battery charging, surface illumination periods, and orbiter visibility. All communications are relayed via the Rosetta main spacecraft via an S-band RF-system using low gain patch antennas.

Philae is operated by a Lander Control Center (LCC), located at DLR in Cologne/Germany, and a Science Operations and Navigation Center (SONC) which is located at CNES in Toulouse/France. All telemetry and telecommanding are linked via the Rosetta Mission Control Center at ESA/ESOC in Darmstadt/Germany, where also the responsibility for the overall Rosetta mission including the Lander delivery lies.

All Philae operations are planned and verified by using a Ground Reference Model representing the Flight Model, consisting of flight spare units and dedicated simulators (Fig. 7.38).

[3] AU stands for "astronomical unit" and equals the average distance Earth to Sun [~150 million km].

7.4.4.2 Rover Operations

One special case of operating surface elements is the one of moving devices as rovers. So far, rovers have been operated at the Moon (including those driven by astronauts in the frame of Apollo 15-17) and Mars. The small body hoppers Phobos PROP-F and Minerva are sometimes referred to as "rovers" as well; both could not be delivered successfully. MASCOT is a small surface package, provided by DLR and CNES to the Japanese Hayabusa 2 asteroid mission (to be launched in late 2014) and will have the capability of hopping using an internal torquer.

The first successful extraterrestrial rovers were Lunokhod 1 and 2 at the Moon, launched in 1970 and 1973 in the frame of the Luna 17 and 21 missions. These devices could survive the Lunar night (due to radiothermal heaters), resulting in an operational lifetime on the surface of the Moon of 11 and 4 months, respectively.

The rovers were controlled directly from Earth by teams of five persons operating in 2 h shifts. Each crew included the commander, a driver, a navigator, a flight-engineer, and an antenna-operator. There was no automated mode. The wheels of the rovers could be controlled individually; there was also an automatic fail-safe device preventing movement over extreme slopes or overheating of the motors. The control teams operated and steered the rovers by viewing through a pair of video cameras, transmitting images with a frequency of one frame per 20 s in case of Lunokhod 1 and adjustably up to one frame every 3.2 s in case of Lunokhod 2. The image rate together with the signal travel time (about 1.3 s, one-way) made operations challenging. Lunokhod 1 had only one driving speed (0.8 km/h) while Lunokhod 2 could be driven with two velocities (0.8 and 2 km/h) (Wesley and Marov 2011; Aleksandrov et al. 1971). The rovers drove through rough terrain and crossed valleys and craters. Lunokhod 2 was lost when driving accidentally into a shadowed crater and scraping its lid during recovery. This eventually resulted in dust on the radiator and solar generator and consequently caused the end of the mission (Figs. 7.39 and 7.40).

On Mars, so far, four rovers have been operated (not counting a Soviet walking robot in 1972, which never transmitted any data). The first such device, Sojourner, was part of the Mars Pathfinder mission, launched in December 1996. Sojourner

Fig. 7.39 Lunokhod control room (image: Aleksandrov et al. 1971)

had six wheels and a mass of 10.6 kg; it was equipped with three cameras in addition to an alpha/X-ray fluorescence (APX) spectrometer for rock analyses. Communications with the lander were via UHF link. The rover drove off the landing platform 2 days after touchdown and travelled an overall distance of about 100 m but was never away more than 12 m from the station. Maximum speed was 1 cm/s (Fig. 7.41).

The next big step in terms of mobility on Mars was the Mars Exploration Rover (MER) mission including two rovers, Spirit and Opportunity. As mentioned above, they were landed with an innovative airbag system (NASA http://msl-scicorner, 2014). After touchdown in Meridiani Planum and inside Endurance crater, respectively (at sites where sediments and proof for early liquid water was expected and indeed found), a protective tetrahedron opened, bringing the rover in an upright position. The rovers, at the landing platform, got activated and unfolded their solar generators and explored an area of many kilometers. Designed for a lifetime of about 60 days (assuming dust coverage on the solar cells), Spirit operated over 6 years while Opportunity is still operational [as this is written in July 2013].

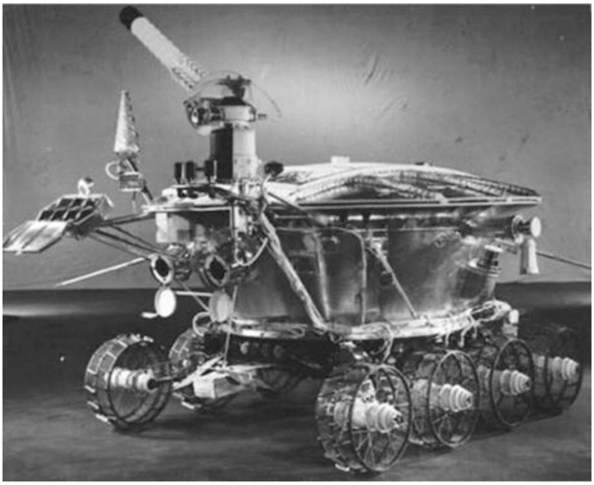

Fig. 7.40 Lunokhod 1 (image: NASA/Lavochkin)

Fig. 7.41 Sojourner on Mars at "Yogi rock" as imaged by central station (image: JPL/NASA)

After landing, Spirit did not send any science data; however, a signal could be received via Mars Global Surveyor. The problem turned out to be related to the flash memory and could be solved by reformatting it and updating the onboard software. Spirit was fully in science operations about 1 month after touchdown; the software update was also applied for Opportunity.

The egress phase including opening of the shell, rover stand-up, deployment of the pan-cam mast, and analyzing images to select the optimum egress pass before finally driving off the platform took the first 4 days.

The rovers could drive with a maximum speed of 5 cm/s; however, hardly more than about 1 cm/s (on flat ground) was reached due to hazard avoidance protocols.

Both MER rovers provided a wealth of scientific results; they also demonstrated the value of moving platforms allowing detailed exploration of an area rather than a single landing spot.

The by far largest rover operated on Mars, Curiosity, with a mass of about 900 kg, has landed successfully within Gale Crater, in August 2012. After being separated from the sky-crane (see above), the system was checked and the first images have been transmitted to Earth. Curiosity uses two communications systems: science data are primarily transferred with a UHF system (about 250 Mbit/d), while for commanding there is an X-band system available (Fig. 7.42). The rover is powered by an RTG providing 125 W electric power (beginning of life). The primary mission shall last at least one Martian year (i.e., 687 Earth-days).

In the first 3 months of Curiosity's operations on Mars, the operations teams were working according to "Mars time." Since a day on Mars (sol) is about 24:40 hours, work shifts are not easily correlated with comfortable working hours on Earth. However, since November 2012, the teams are working in shifts linked to

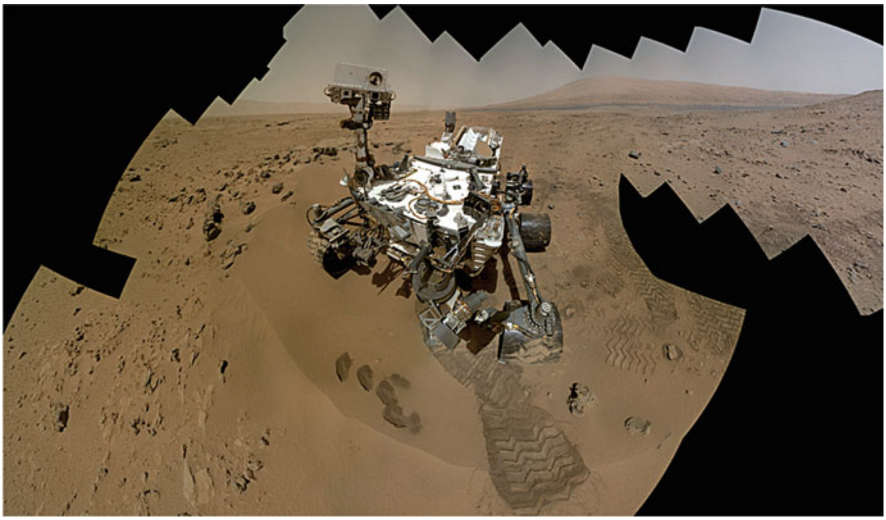

Fig. 7.42 "Self Portrait" of the Curiosity Rover at the surface of Mars in Gale Crater, taken with Mars Hand Lens Imager (image: NASA PIA16239)

Earth time, with the majority of the activities from 8:00 to 20:00 local (JPL) time. Science teams are now able to work from their institutes, while co-location was required in the first phase of the mission. For more details on science planning and operations, see, e.g., (NASA http://msl-scicorner, 2014).

The Curiosity Rover is the so far most complex device ever landed on Mars. It is a mobile laboratory and will enhance our understanding on Mars considerably. A consecutive Rover mission, reapplying the technologies developed for Curiosity, is now planned to be launched in 2020.

7.4.5 Conclusions

There is a general trend from short- to long-lived and from stationary to mobile surface elements. Accordingly, the operations gets more and more demanding but there is also an increased emphasis on autonomy. In the far future, devices based on highly autonomous exploration strategies with a high degree of artificial intelligence, including intercommunications of a number of probes or rovers may be realized. Theoretical studies have proposed such concepts. However, certain decision points within the mission timeline, requiring human involvement, as well as careful planning, will stay as essential elements to all future space missions.

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