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Conceptual Design and Flight Simulation of Space Station Missions beyond Low Earth Orbit

Dissertation

A thesis accepted by the Department of Aerospace Engineering and Geodesy
of the Universität Stuttgart as partial fulfilment of the requirements
for the degree of Doctor of Engineering Sciences (Dr.-Ing.)

by

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Date of defence: 10 January 2006

Institut für Raumfahrtsysteme
Universität Stuttgart

2006

Bibliografische Information der Deutschen Nationalbibliothek
Die Deutsche Nationalbibliothek verzeichnet diese Publikation in der Deutschen Nationalbibliografie; detaillierte bibliografische Daten sind im Internet über **http://dnb.d-nb.de** abrufbar.

ISBN 3-89963-441-1

D93

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1. Auflage 2006

Druck und Bindung: printy, München (www.printy.de)

Abstract

Conceptual Design and Flight Simulation of Space Station Missions beyond Low Earth Orbit

Humans will live and work in space for the exploration and development of the solar system. A wide range of space infrastructure elements will be required in low Earth orbit and beyond. Besides new transfer and re-entry vehicles as well as planetary surface installations, space stations in the Earth-Moon system can be a crucial element of forthcoming exploration missions.

This dissertation documents an investigation on conceptual design and flight simulation of such space station missions beyond low Earth orbit, namely in the Earth-Moon system. The goal is to develop and extend the methodology and software tools of the conceptual design process (Space Station Design Workshop, SSDW) in order to enable spaceflight systems engineering of space stations in the context of future mission scenarios and architectures.

The methodological approach for human spaceflight mission design is discussed with taking into account the special characteristics and requirements of interdisciplinary teamwork and software tool support. The results reveal that mission aspects such as the transfer problem are much more relevant than before. The emphasis lies then on the software engineering approach and major characteristics of the computer programmes developed for space systems modelling and dynamic simulation.

A design example demonstrates the application of the methodology and tools on a conceptual design problem targeting at a space station mission at the lunar Lagrange point one (LL1), upon which near-term lunar surface exploration missions can build on. Challenged by the constraint of using existing and tailored European/Russian technology and infrastructure elements, the results manifest the feasibility of such a space station that offers various utilization possibilities. The results documented include the station configuration and modules, the transfer vehicles for crew and cargo transport, the station's life support system and a logistics concept. The concept outlines enhancements of the current transportation and station infrastructure and shows that the International Space Station (ISS) as a transportation node can beneficially support lunar scenarios.

Keywords: space systems engineering, modelling, lunar exploration architecture, transfer vehicle, Lagrange point, human spaceflight

Zusammenfassung

Vorentwurf und Flugsimulation von Raumstationsmissionen außerhalb des erdnahen Weltraums

Menschen werden zur Erforschung und Erschließung des Sonnensystems in den Weltraum reisen, dort leben und arbeiten. Eine Vielzahl von Weltrauminfrastrukturelementen werden dazu notwendig sein, sowohl im erdnahen Raum als auch außerhalb. Neben neuen Transfer- und Rückkehrfahrzeugen und planetaren Oberflächeninstallationen können Raumstationen eine tragende Rolle bei den kommenden Explorations-Missionen übernehmen.

Dieser Bericht dokumentiert eine Arbeit zu Vorentwurf und Flugsimulation von Raumstationen für Missionen außerhalb des erdnahen Weltraums, speziell im Erde-Mond-System. Das dabei verfolgte Ziel war die Weiterentwicklung und Verbesserung der Methodik und der Softwarewerkzeuge für den Vorentwurfsprozess erdnaher Raumstationen (Space Station Design Workshop, SSDW) damit der Missions- und Systementwurf im Kontext zukünftiger Missionsszenarien und -architekturen gelingt.

Zunächst wird der methodische Ansatz für den Entwurf bemannter Raumfahrtmissionen diskutiert, wobei die spezifischen Charakteristika und die Organisation der interdisziplinären Entwurfsarbeit und ihre Unterstützung durch Softwarewerkzeuge besondere Berücksichtigung finden. Das Ergebnis zeigt, dass im Vorentwurf nun auch Missionsaspekten, wie der Transferaufgabe, verstärkte Relevanz zukommt. Daran anschließend werden auf die entwickelten Computerprogramme zur Systemmodellierung und dynamischen Simulation eingegangen und die eingesetzten Modelle und Methoden zusammengefasst.

Schließlich demonstriert ein Entwurfsbeispiel die Anwendung des Ansatzes und der Werkzeuge auf ein Entwurfsproblem für eine Raumstationsmission der näheren Zukunft. Hierbei wurde eine Missionsarchitektur mit einer Raumstation im lunaren Lagrange-Punkt eins (LL1) entwickelt, die besonders vorteilhaft für Explorations-Missionen zur Mondoberfläche genutzt werden kann. Die gewählte Randbedingung, dass existierende europäische bzw. russische Technologien und Infrastrukturelemente Verwendung finden sollen, zeigte, dass eine solche Raumstation machbar ist und vielfältige Nutzungsmöglichkeiten bietet. Der dokumentierte Entwurf beinhaltet die Stationskonfiguration und -module, die Transferfahrzeuge für Besatzungen und Fracht, das Lebenserhaltungssystem der Station und ein Logistikkonzept. Dadurch werden einige Weiterentwicklungsmöglichkeiten der heutigen Infrastrukturen skizziert und aufgezeigt, wie die Internationale Raumstation lunare Szenarien sinnvoll als Transportknoten unterstützen kann.

Stichworte: Systementwurf, Modellierung, Mondexplorationsarchitektur, Transferfahrzeuge, Lagrange-Punkt, Astronautik

Danksagung

Nach Abschluss meiner Doktorarbeit möchte ich an dieser Stelle allen Personen danken, die mir auf diesem Weg fachlich und persönlich zur Seite gestanden haben.

Allen voran danke ich meinem Doktorvater Prof. Ernst Messerschmid für sein stetes Vertrauen und die vielfältigen Anregungen. Trotz seiner Verpflichtungen als Direktor des europäischen Astronautenzentrums gab er mir fortwährend und immer zu richtigen Zeitpunkten die Unterstützung, die grundlegend für die erfolgreiche Durchführung der Arbeit war. Herrn Prof. Ingolf Grieger danke ich herzlich für die Übernahme des Mitberichts.

Ganz besonders danke ich dem ESA ATV-Team in Les Mureaux, namentlich Herrn Robert Lainé und Herrn Jean-François Clervoy für ihre freundliche und großzügige Unterstützung. Herzlicher Dank geht auch an Herrn Prof. Hans-Ulrich Keller vom Planetarium Stuttgart für seine Anregungen zur Verifikation des Sonnensystem-Modells. Ebenfalls möchte ich Herrn Dr. David Hoffman vom JSC in Houston für den fachlichen Austausch und die Hinweise bezüglich Librationspunkte-Orbits danken.

Meinen Kollegen am Institut für Raumfahrtsysteme gebührt mein besonderer Dank, vor allem meinem früheren SSDW-Team Kollegen Herrn Dr. Jan Osburg für die Unterstützung beim Einstieg in die der Universität eigenen Welt und als Mitstreiter bei der Ausrichtung der gemeinsamen Space Station Design Workshops, sowie Frau Ilse Müllner-Korell, die sich erfolgreich um mein kulinarisches Wohlergehen am Institut kümmerte. Frau Jennifer Baer-Engel bin ich für die engagierte Sprachkorrektur des englischen Manuskriptes sehr dankbar.

Besonders danke ich auch „meinen“ Studenten, speziell Herrn Daniel Bindel, Herrn Thomas Drodzek, Herrn Rémi Demenet und Frau Britta Ganzer. Durch ihr Mitwirken im Rahmen von Studien-/Diplomarbeiten und als studentische Hilfskräfte haben Sie Anteil am Erfolg dieser Arbeit.

Mein herzlichster Dank gilt meiner Frau Eva Yazdi, die nicht nur meine zuweilen späten Arbeitsstunden entschuldigte, sondern mir in den richtigen Augenblicken vergegenwärtigte, dass neben der fachlichen Berufung weiteres Lebenswertes existiert.

Stuttgart, April 2005 Kian Yazdi

Table of Contents

Abbreviations and Acronyms

#	Number
2L	Two-Launch
2M	Two-Man
3D	Three-Dimensional
3L	Three-Launch
3M	Three-Man
A.D.	Anno Domini
A5	Ariane 5
AC	Assembly Complete
ACM	Attitude Control and free-flight Manoeuvring
ADM	Airlock and Docking Module
AES	Air Evaporation System
AFT	Aft
AHM	Autonomous Habitation Module
AIAA	American Institute of Aeronautics and Astronautics
ANSI	American National Standards Institute
AOCS	Attitude and Orbit Control System
AOS	Acquisition of Signal
ARD	Atmospheric Re-entry Demonstrator
ASL	Aligned to Sun Line
ATV	Automated Transfer Vehicle
AU	Astronomical Unit
B.C.	Before Christ
BCEC	Barycentric Ecliptical coordinate system
BCS	Body-fixed Coordinate System
BM	Body Mounted
BOL	Begin Of Life
CHX	Condensing Heat Exchanger
Cryo-LTV	Cryogenic Lunar Transfer Vehicle
Cryo-CTV	Cryogenic Crew Transfer Vehicle
CM	Command Module (Apollo)
CoG	Centre of Gravity
COL	Columbus Orbital Laboratory
CoM	Centre of Mass
CoP	Centre of Pressure
CRV	Crew Rescue/Return Vehicle
CSM	Cargo and Storage Module
CTV	Crew Transportation Vehicle
DAVIS	Data Visualiser
DC	Docking Compartment
DM	Descent Module (Soyuz)
DMLSS	Delivered Mass to Lunar Space Station
DMS-R	Data Management System - Russian
DSE	Deep Space Escape
E	Equator
EADS	European Aeronautics Defence and Space Company
ECA/B	Etage Cryotechnique A/B (French: Cryogenic Stage A/B)
ECLSS	Environmental Control and Life Support System
EDC	Electro-chemical Depolarised CO2 Concentration
EGM	Earth Gravity Model
ELISSA	Environment for Life-Support Systems Simulation and Analysis

ELLIPSE	European Lunar Libration Point Scenario
EO	Earth Oriented
EP	Exposed Platform
EPB	Equipped Propulsion Bay (ATV)
EPS	Electrical Power System
EPS	Etage Propergols Stockables, *also:* Ensemble Propulsif Stockables (French: Solid Propellant Stage)
ERA	European Robotic Arm
ES/V	Etage Supérieur/Versatile (French: Advanced/Versatile Stage)
ESA	European Space Agency
ESC-A/B	Etage Supérieur Cryotechnique A/B (French: Advanced Cryogenic Stage A/B)
ETI	Earth Transfer Injection
ETO	Earth Transfer Orbit
EVA	Extra Vehicular Activity
FGB	Functionalnyi Gruzovoi Blok (Russian: Functional Cargo Block)
FWD	Forward
GEO	Geostationary Earth Orbit
GG	Gravity Gradient
GHA	Greenwich Hour Angle
GM	Gravitational constant times Mass of Earth
GMM	Goddard Mars Model
GMST	Greenwich Mean Sidereal Time
GMT	Greenwich Mean Time
GPS	Global Positioning System
GRAM	Global Reference Atmospheric Model
GTO	GEO Transfer Orbit
GTS	Global Time Services
Hab	Habitat
HCEC	Heliocentric Ecliptical coordinate system
HLLV	Heavy Lift Launch Vehicle
HST	Hubble Space Telescope
ICC	Integrated Cargo Carrier (ATV)
IERS	International Earth Rotation Service
IMLEO	Initial Mass in Low Earth Orbit
IN	Inertial Oriented
IOC	Initial Orbital Capability
IOP	In the Orbit Plane
IRIS++	Interaktives Raumfahrtspezifisches Interpretations-System basierend auf C++ (German: Interactive spaceflight-specific interpretation system based on C++)
IRS	Institut für Raumfahrtsysteme (German: Institute of Space Systems)
ISPR	International Standard Payload Rack
ISS	International Space Station
IVA	Intra Vehicular Activity
J2000	Standard epoch referring to 2451545.0 JD or 12:00 January 1, 2000 A.D.
JD	Julian Date
JDE	Julian Ephemeris Date
JEM	Japanese Experimental Module (ISS)
JPL	Jet Propulsion Laboratory
Lab	Laboratory
LEO	Low Earth Orbit
LES	Launch Escape System
LEV	Lunar Exploration Vehicles
LH	Local Horizontal plane
LH	Liquid Hydrogen
LLO	Low Lunar Orbit

LOCS	Local Orbital Coordinate System
LOK	Lunnyi Orbitalnyi Korabl (Russian: Lunar Orbital Spacecraft)
LOP	Lunar Orbital Plane
LOX	Liquid Oxygen
LSB	Lunar Surface Base
LSM	Life Support Module
LSS	Lunar Space Station
LTI	Lunar Transfer Injection
LTO	Lunar Transfer Orbit
LTV	Logistics Transportation Vehicle
LV	Local Vertical (Nadir pointing)
MCC	Mid-Course Correction
MCS	Module-fixed Coordinate System
MJD	Modified Julian Date
MMH	Mono-Methyl Hydrazine
MPLM	Multi-Purpose Logistics Module
MS	Microsoft (chapter 3)
MS	Mission Statement (chapter 5)
MSIS	Mass Spectrometer Incoherent Scatter
N/A	Not available
NASA	National Aeronautics and Space Administration
NTO	Nitrogen Tetroxide
OD	Orbit Declination
OI	Orbit Insertion
OOD	Object-Oriented Design
OOP	Object-Oriented Programming
Ops	Operations
ORCS	Orbit Reference Coordinate System
OM	Orbital Module (Soyuz/Progress)
P/L	Payload
PBR	Photo-Bio-Reactor
PC	Personal Computer
PCEC	Planeto-centric Ecliptical
PCEQ	Planeto-centric Equatorial
PCOF	Planeto-centric Orbit-Fixed
PCPF	Planeto-centric Planet-Fixed
POP	Perpendicular to the Orbit Plane
PxC	Permanent x crew Capability (with “x” being the number of crew members)
RAAN	Right Ascension of the Ascending Node
RCS	Reaction Control System
RDS	Russian Docking System
SC	Spacecraft
SCCS	Spacecraft Commanded Coordinate System
SCSA	Spacecraft Subassembly (ATV)
SFWE	Static Feed Water Electrolysis
SI	Système International d’Unités (French: International system of units)
SM	Service Module
SPCS	Spacecraft Principle Coordinate System
SPP	Solar Power Platform (ISS)
SR	Sabatier Reactor
SRCS	Spacecraft Reference Coordinate System
SRP	Solar Radiation Pressure
SSDW	Space Station Design Workshop
STS	Space Transportation System (Space Shuttle)
Sup	Supply

SWIS	Solid Waste Incineration System
TAI	Tempe Atomique Internationale (French: International Atomic Time)
TCCS	Trace Contaminant Control System
TCS	Thermal Control System
TDB	Tempe dynamic barycentric (French: Barycentric Dynamic Time)
TDRS	Tracking and Data Relay Satellite
TEA	Torque Equilibrium Attitude
TRL	Technological Readiness Level
TT	Terrestrial Time
TWP	Transfer Window Period
TxC	Temporary x crew Capability (with "x" being the number of crew members)
UDM	Universal Docking Module
UDMH	Unsymmetrical Di-Methyl Hydrazine
US	United States
UT	Universal Time (i.e. UT1)
UTC	Coordinated Universal Time (legal time system)
VPCAR	Vapour Phase Catalytic Ammonia Removal System
VRML	Virtual Reality Modelling Language
w/o	without
XEUS	X-ray Evolving Universe Spectroscopy
XMM	X-ray spectroscopy Multi-Mirror

For representation of numerical values please note following convention: A dot "." is used as decimal symbol and no digit-grouping symbol appears in this document.

1 Introduction

Looking back 30 years, no human space exploration effort beyond low Earth orbit (LEO) has existed since *Apollo*. Now, after the dawn of the 21st century, it seems that NASA and other space agencies have arrived at the conclusion that the time has come to leave Earth orbit for new human spaceflight challenges. Recently, the US government has decided on NASA's long-term strategic space programme leading back to lunar activities and more to come. The European Space Agency (ESA) formulated its *Aurora* programme in 2001, which includes the preparation and execution of the long-term human exploration of the solar system bodies. In connection with this, an international human mission to Mars is envisioned with the possibility of using the Moon as a way station. This means the examination of cis-lunar and interplanetary spaceflight scenarios has officially appeared on the agenda again. In the upcoming years design and analysis work will therefore mainly be determined by various concepts for building up an efficient and cost-effective support and utilization infrastructure for these spaceflight plans. This work requires appropriate methodology and tools for the task of generating, analysing and evaluating space mission and space system concepts.

This thesis addresses such conceptual design methodology and tools for spaceflight missions. It extends the developed framework of the *Space Station Design Workshop* (SSDW) that is an educational event of about one week offering an interdisciplinary team of participants practising systems engineering in a multinational environment. The SSDW is organised and performed nearly annually since 1997 by the *Astronautics* and *Space Stations* department of the *Institute of Space Systems*. It applies the methodology and tools for concepts' design and assessment of LEO space stations in terms of feasibility, technological requirements, utilization, operations and cost. The presented work concentrates on the fundamental components of the space station design methodology and on enhancing and extending the process already in use for future scenarios. It discusses mission design aspects and the organisation of interdisciplinary teamwork with also taking into account the "human factors" during the design process. Special emphasis is put on establishing the technical means for modelling, simulation and analysis of the space segment of cis-lunar and interplanetary exploration missions. This involves the

- development of an interactive modelling environment to create and edit 3D spacecraft structural configuration models;
- enhancement of the existing space stations simulation software in terms of flexibility, functionality and accuracy by including generic spacecraft and detailed perturbation models as well as implementing advanced numerical methods;
- extension of this software to dynamical simulation capabilities to open up a new range of applications, ranging from non-LEO scenarios such as orbital missions around celestial bodies other than Earth to transfer missions in cis-lunar and interplanetary space;
- introduction of state-of-the-art visualisation capabilities for representing simulation data and spacecraft configurations for documentation and presentation.

With these extensions, the SSDW offers a presumably unique environment for conceptual space mission design and analysis in terms of a harmonised integration of a proven interdisciplinary methodology and generic software infrastructure.

Organisation of this report

Chapter 2 gives an overview of the conceptual design of human space missions with special emphasis on the process of systems engineering, conceptual design and software tools applied. Furthermore, it outlines teamwork-related findings to control and enhance design team performance. Chapter 3 concentrates on spacecraft modelling and presents the developed approaches and implemented software for modelling space station structural configurations and geometrical surface discretisation. The developed flight simulation programme will then be the subject of chapter 4. It addresses in detail the software engineering approach and the dynamics as well as perturbation models. Finally, the purpose of chapter 5 is to demonstrate the application of the tools established with this work on an example spaceflight scenario. Here a lunar space station mission is envisioned and conceptually designed and analysed in detail. The actual range of possible utilizations of such a station is manifold and is addressed in the beginning of this chapter. This is followed by a discussion of the associated mission design and analysis challenges and presentation of the conceptually designed lunar exploration mission architecture. Documented results include the station configuration and modules, transfer vehicles for assembly, crew and cargo transport, an enhanced life support system and an assembly and logistics concept.

The conclusion in chapter 6 provides a summary of the results achieved and offers an outlook on further research possibilities.

2 Designing Human Space Missions

Every space mission and system design process is embedded into a typical context. The following sections address this context, namely the conceptual design problem, the complexity when dealing with missions beyond LEO, and the extensions proposed in this research on the methodological approach. The designer's tools to face this class of problems and the implementation within the framework of the Space Station Design Workshop (SSDW) are discussed as well.

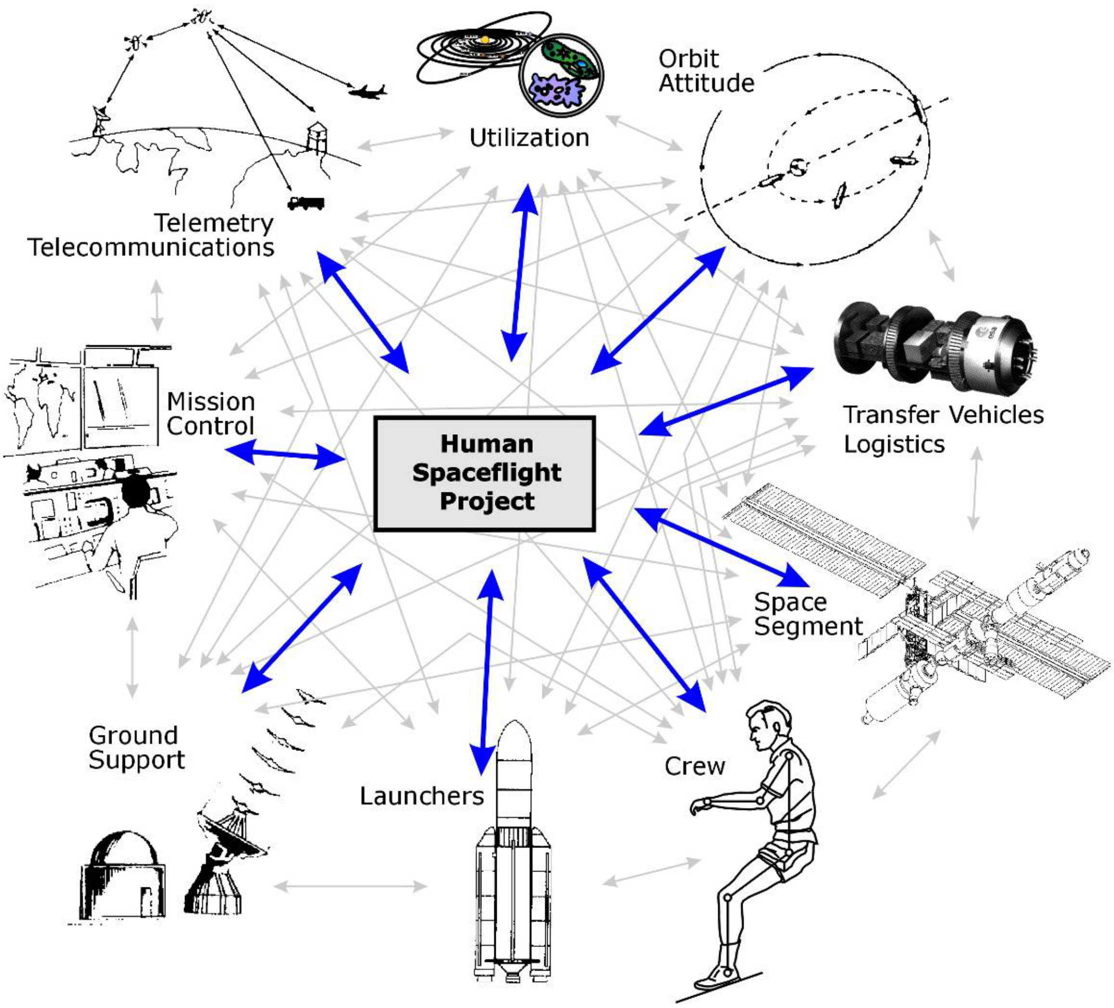


Figure 2.1: Mission and system element interactions

2.1 The Conceptual Design Problem

In the beginning of designing a space mission or system stands a mission statement describing the objectives of the customer. Politicians, economists or scientists have their specific expectations in mind to formulate these objectives. Therefore, from the engineering point of view, the given mission and system requirements are rather vague or "fuzzy" and have to be translated into primary and secondary objectives, defining technological requirements and technological as well as political and economical constraints. This understanding and verification of what the customer wants is crucial to satisfying the customer expectations and needs and therefore for project success. This early phase of a space project is referred to as the *conceptual design phase*.

As illustrated in Figure 2.1 all mission and system elements are strongly interdependent. Changes to one element impose direct or indirect changes to largely every other element. All local interferences could yield significant consequences to the whole system.

Therefore, within this early project phase of conceptual design of the overall mission and the systems, every one of its elements must be considered simultaneously down to a high subsystem requirement level. Conflicting requirements must be dispelled and fundamental mission and system parameters have to be concretised, optimised and fixed in a baseline concept following an iterative process. At this point in the project, major decisions have to be made in terms of project complexity, applied technologies and specific and overall cost elements.

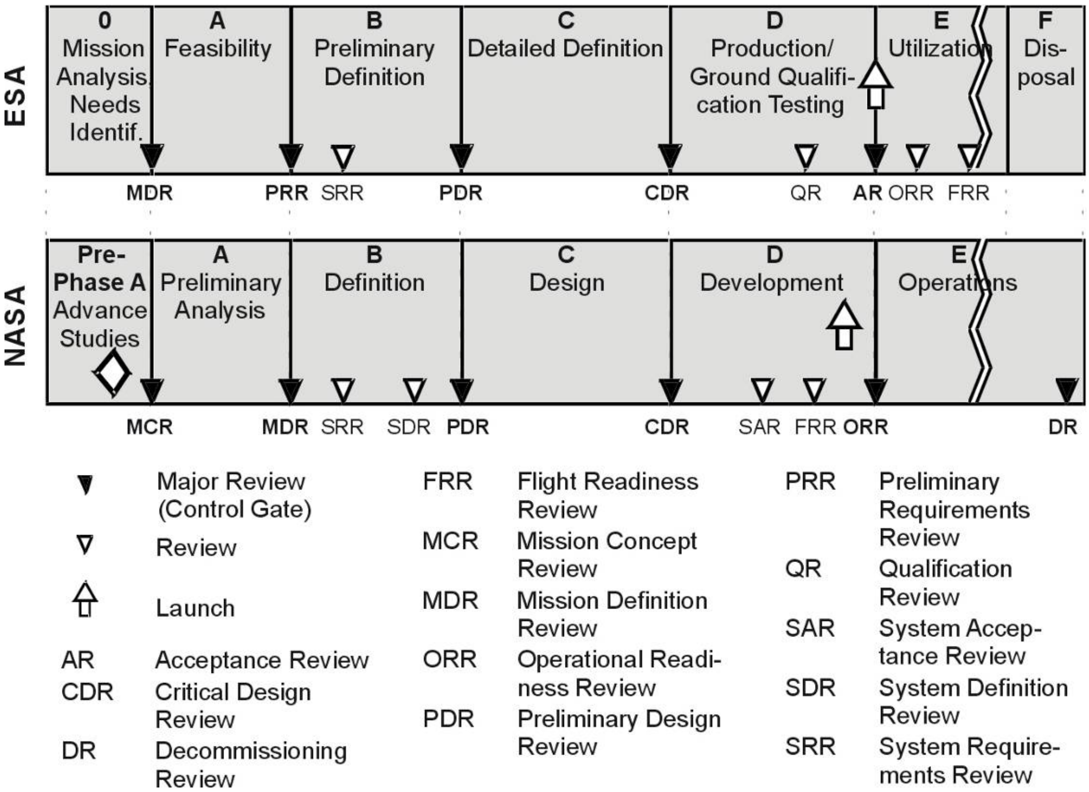


Figure 2.2: Space project life cycle ([Bertrand1998],[ECSS-M-30A])

In Figure 2.2 the successive phases within a typical life cycle of a space project are illustrated. Any project begins with phase 0 or rather pre-phase A, which is commonly referred to conceptual design phase of the project. Its methodology and tools can spread into phase A and even B enabling consistent project evolution. Figure 2.3 shows the approximate cost assignment and expenditures within a project life cycle. Considering that up to 50% of the total cost assignment is completed by the end of phase A and up to 80% by the end of phase B, the key role of the conceptual design work becomes evident. Thus, investing in this phase and focusing on “doing-it-right” is of primary importance. Conflicting assumptions or invalid decisions made during this period would only have to be corrected at later stages. This could lead to changes that would possibly influence every element, most likely producing extremely high costs and delays.

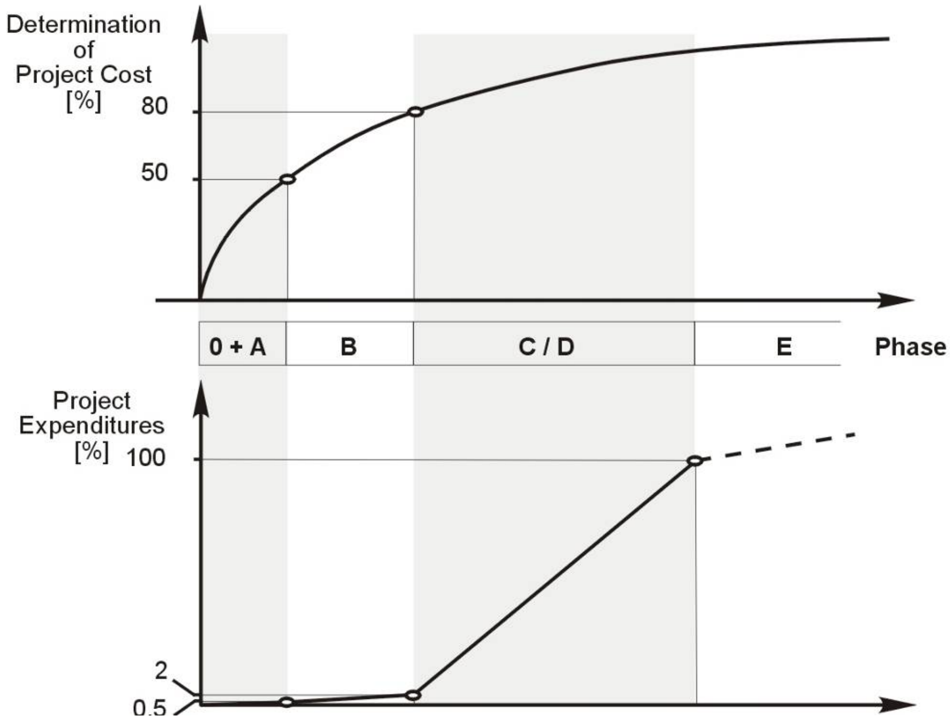


Figure 2.3: Cost determination and project expenditures during project life cycle
[Messerschmid2000]

Task of conceptual mission and system design

The objective of conceptual mission and system design is tackling the task of producing feasible, reasonable and sustainable mission and system concepts for a given mission statement. At its best it will produce a consensus - but more likely a compromise - of all influencing factors taken into account. Its outcome is a broad definition of the mission and its components [Wertz1999] and will give the customer an impression of project feasibility, complexity, utility and financial scale in order to allow the "stop" or "go" decision on the project's further development. Furthermore, the evaluation must give the design engineers a solid basis from which to proceed and it should indicate where the major technological challenges and cost drivers will lie.

The main task of the conceptual design phase can be summarised as follows:

- Clarification of mission objectives, system requirements and constraints in cooperation with the customer (target identification and target matching)
- Concept exploration by studying various alternatives of mission and system architectures
- Generation of a baseline mission and system design, satisfying the objectives and meeting the requirements and constraints
- Documentation of the results and the rationales for decision making

Conceptual design approaches

To meet the requirements summarised above, a process is applied that has many commonalities with the processes associated with *Systems Engineering* (SE), *System Architecting* (SA) and *Concurrent Engineering* (CE). Thus, it is important to define these terms in the meaning of how they make up the design philosophy:

The *International Council of Systems Engineering* [INCOSE] offers the following definition of Systems Engineering:

> Systems Engineering is an interdisciplinary approach and means to enable the realization of successful systems. It focuses on defining customer needs and required functionality early in the development cycle, documenting requirements, then proceeding with design synthesis and system validation while considering the complete problem. It integrates all the disciplines and specialty groups into a team effort forming a structured development process that proceeds from concept to production to operation. It considers both the business and the technical needs [...] with the goal of providing a [..] product that meets the user needs.

In other words, the SE discipline alone focuses on *methods* to solve problems, not the solution to the *problems* [ISR2004]. The following definitions will more detail how these methods are understood within this study to actually solve space system design problems.

Taking GRIFFIN and FRENCH for granted, systems engineering can also be described as follows [Griffin1991]:

> Space systems engineering is the art and science of developing an operable system capable of meeting mission requirements within imposed constraints including (but not restricted to) mass, cost, and schedule.

This is quite a broad definition, referring to the complete space system design process as part of the space mission – not only in the conceptual design phase. It includes not only the "science", i.e. the technical and business issues involved within the task to be performed. These issues range from creating a system architecture, coming-up with creative solutions to the use of mathematics, constructing physical models and simulating systems and subsystems to evaluate their properties, their performance, mission utility and cost. It also introduces the "art" of systems engineering pointing to the "soft-skills" necessary to successfully identify requirements and constraints, resolve conflicting objectives, manage the team work and interaction and combine the team's range of individual expertise's and experiences to the optimum. As science, art is - given reasonable talent - a business of practice, a thing that can be learned and that can be educated. And this is what the SSDW is meant to do (see also section 2.2.2).

System Architecting can be seen as a counterbalance to traditional systems engineering that uses a systematic application of science and mathematics (e.g. analysis and optimisation), whereas the architect reduces the complex system problem to a manageable level to define basic solution concepts on which the analytical engineering methods can bear [Bertrand1998]. On the other hand, and this is the understanding of SE within this report, system architecting can be seen as the initial step of SE, where a rather small interdisciplinary team (the *system architects*) performs the initial conceptual design [Bertrand1998]) and an accompanying process of SE that maintains consistency and assures system integrity throughout the whole design and definition process. Furthermore, the methodological approach incorporates creativity and communication tools taken from the discipline of *terrestrial architecture* to enhance the conceptual design process (see section 2.3).

Concurrent Engineering is a rather new methodology of the systems engineering process, by name indicating its difference to the classical, sequential process. While the traditional approach is defined by strictly linearly succeeding steps by separate individuals or small groups (A defines requirements, B defines the product, C defines the process, etc.) the iteration effort is high and the total process is stretched. Concurrent engineering's approach intervenes at this point in how the design and development process is organised. It creates an environment, namely a localised *facility*, where the involved parties join to form an interdisciplinary team. Thus communication and the number of iterations will increase and can now follow a non-linear pattern. Design and development steps are meant to overlap and, given an appropriated communication lead and team member experience, this leads to a significantly higher level of efficiency and final product maturity. Furthermore, these conditions foster *system awareness* of all involved individuals and enable them to become *system engineers*. Cornerstones for the implementation of CE are the following issues:

- Product, i.e. orienting all design efforts to meet the system's objectives and needs
- Team, i.e. the sum of personalities forming an interdisciplinary team of expertise
- Process, i.e. formalised methodology of managing communication and the overall work flow
- Tools, i.e. a hardware/software infrastructure supporting the team's communication, design and analysis work and documentation

Methodological requirements

The overall process of mission and system design and the involved interactions are illustrated in Figure 2.4. Due to the high number of interactions, this process often seems and tends to be disorganised and chaotic in reality [Larson2000]. This is also true for the conceptual design phase which mirrors this process but at a somewhat lower detail level. In order to accomplish this process successful the design and analysis approach must meet specific requirements that can be summarised as follows:

- Integrated mission and system design process
- Multidisciplinary approach
- Clear methodology with simple steps
- Sufficient iterations
- Appropriate tools supporting concept creation and assessment

Process definition for conceptual mission and system design

Using the system engineering philosophy described above, selected steps within the conceptual mission design and analysis process shall be concretised in this section. These tasks are listed in Table 2.1 and details are included on how the tasks are carried out. First listed is the design work preparation and monitoring, which is - as experience shows - fundamental for a flawless start and progress of the design work and quality design results. The topics listed here, especially the verification of information consistency, have to be recalled again from time-to-time during all successive steps in order to maintain an efficient and successful design flow.

Although listed as succeeding steps, the design work does not follow this scheme linearly but is a highly iterative process. To understand and to face the design problem efficiently, sometimes the designers can even "jump ahead" or iterate back at a certain point in the process [Osburg2002].

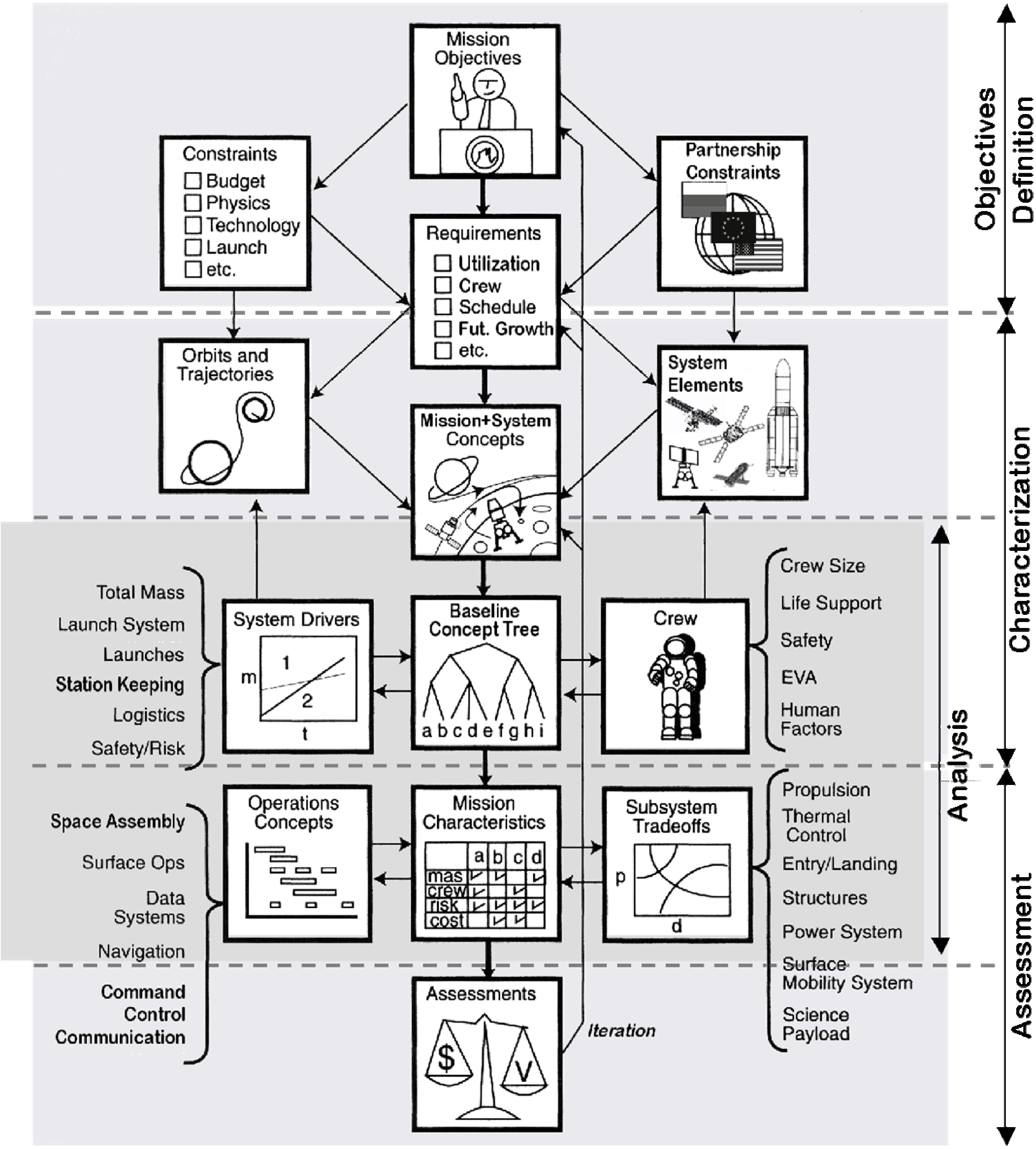


Figure 2.4: Principle interactions within Mission and System Design (adapted from [Larson2000])

While step A in Table 2.1 focuses on understanding what the design goal is, step B begins with the actual design work. This task of space mission and system characterization incorporates in particular the following eight sub-tasks, or rather sub-steps [Bertrand1998]:

1. Brainstorming for alternative approaches for mission accomplishment
2. Selecting major mission components (station, transfer vehicles, crew, launchers, supporting satellites and platforms, etc.)

3. Selecting the location and orientation (e.g. station's orbit and flight mode)
4. Selecting the station keeping approach (e.g. station's orbit and attitude control)
5. Selecting space segment modules (e.g. trusses, panels, pressurised modules such as labs, habs and nodes, airlocks, external platforms, etc.)
6. Developing system configuration architectures (geometrical design and module topology) and assembly strategy
7. Making decisions on the station's critical subsystems (power, AOC, life support, recycling and potential synergies)
8. Developing a logistics approach for re-supply and crew transfer

As with the whole process, these sub-steps are also performed in a highly iterative manner. The preliminary decisions made here for each alternative concept are reviewed, analysed and evaluated through step C. While in step B creativity should dominate initially without criticizing premature ideas and therefore unnecessarily fencing the range of imagination, in step C an analytic viewpoint is required to determine pros and cons of the individual approaches and then finally selecting, or rather defining, a baseline scenario. Here budgetary calculations as well as first modelling and numerical simulation activities are introduced in order to get impressions of mission utility as well as mission and system complexity. The analysis via simulations is detailed for the baseline concept within step D with fine-tuning of the concept iteratively. Then the technical requirements and their allocation to the system elements are formally defined.

Typically steps 0, A, B and E primarily involve the design team as a whole and steps C and D will be performed by splitting into sub-teams with individual responsibilities corresponding to the mission and system elements to be addressed.

Table 2.1: Process of conceptual mission and system design (adapted from [Bertrand1998], [Wertz1999])

Task	Steps
0. Design work preparation and monitoring	▸ Assemble design team and allocate sub-team responsibilities ▸ Verify information consistency throughout the team ▸ Perform background information retrieval and analysis
A. Objectives definition	▸ Review mission statement ▸ Identify primary and secondary objectives ▸ Identify requirements and constraints
B. Mission and system characterization	▸ Develop alternative concepts and architectures ▸ Characterise the elements ▸ Identify design drivers and critical technical requirements
C. Concepts assessment	▸ Prepare system and subsystem budgets ▸ Evaluate mission utility and cost ▸ Select baseline scenario
D. Requirements verification	▸ Review concept and refine mission and system elements ▸ Define technical requirements ▸ Allocate requirements on system and subsystem level
E. Results documentation	▸ Conserve baseline concept and rationales for selection ▸ Conserve alternative concepts for later reference

Mission Design Aspects of Space Stations beyond LEO

Besides fulfilling of the utilization needs, in general, mission statements of interplanetary human spaceflight mission design will demand several additional requirements to be fulfilled:

- Easy access via available transportation systems (efficient utilization of launchers and their launch sites)
- Safe mission operations (i.e. large departure and arriving windows, robust free-return and/or abort options) [Brown1998]
- A human factors supporting concept: especially the reduction of the radiation level (i.e. accumulated radiation dose) and other hazards (micrometeoroids and orbital debris)
- Frequent departure and arrival windows (once per revolution, daily, weekly, monthly, etc)
- Minimum delta-v for transfers, station keeping and orbit maintenance
- Minimum transfer time

Furthermore, constraints arise due to superior issues typically involved in such projects:

- International cooperation: access from launch sites of all potentially involved project partners
- Cost reduction: utilization of the already existing systems in orbit, especially the International Space Station (ISS)
- Long-term programme objectives: e.g. Lunar Space Station: reachable lunar regions (surface latitudes range) or further transfer window properties to Lunar Lagrange Points, Earth-Sun Lagrange Points and other celestial bodies, especially Mars

2.2 Conceptual Design in Practice

The following sections of chapter 2 discuss details on how the conceptual design work is proposed and implemented to fulfil the requirements generated by research and education practice.

2.2.1 The Challenge of Implementation

Creating a systems engineering approach is one aspect. Its implementation into the "real world" is a different one. To demonstrate, test, validate and improve this approach with its methodology and software tools, the *Space Station Design Workshop* (SSDW) is used serving as a test-bed and serving synergistically as an educational platform for systems engineering (see next section).

Nearly every year since 1996 these workshops have been carried out with varying design objectives, the number of participants and the workshop length. This has allowed for the continuous enhancement of this approach. While the organisers observe project evolution and validate the methodology and tools, the participants are being assisted in performing the conceptual design of a space station within a few days.

This thesis' contribution is an extended design and analysis capability of the SSDW for addressing interdisciplinary mission and system design and enabling mission statements dealing with interplanetary missions and cis-lunar space stations. For this purpose the SSDW serves as a realistic playing field for changes in the conceptual design approach and the developed tools. Experience has shown that for successful design team work and reasonable conceptual designs it is not only the methodological approach which has to be defined properly by formulating the tasks and the process and developing appropriate supporting software tools. Just as human factors must be taken into account when designing inhabited space systems, these human factors are also relevant to the systems engineers during the actual design phase. These psychological issues refer to team creation, initiating and organizing the teamwork.

The following section will give a brief overview of the SSDW and will then address team-related issues of the design work.

2.2.2 Space Station Design Workshop (SSDW)

A systems engineering approach aiming for the efficient conceptual design of a space station has been under continual development at the Institute of Space Systems since 1996. The approach is defined by an interdisciplinary methodology and appropriate, custom-designed software tools supporting this methodology. Both are used to carry out interdisciplinary *Space Station Design Workshops* (SSDW) for graduate students of various disciplines from European and other countries. The participant's disciplines range from Aerospace Engineering, Architecture and Industrial Design to Economics and Law. Thus, the participants are given an opportunity and assistance in gaining first-hand experience of the challenges of the conceptual design process in a multinational, interactive, team-centred environment; attributes that are so characteristic for the "space-arena".

One example of an international course was the SSDW 2002 at ESA's technology centre ESTEC in Noordwijk, The Netherlands, with 30 participants forming two competing teams [SSDW2002]. Several short versions of these workshops have been taking place since 1997 at the International Space University (ISU) in Strasbourg, France in the frame of the Masters of Space Studies (MSS). The latest was in 2003 with 50 participants forming three teams.

The objective of these workshops is to generate viable conceptual designs of space stations within a few days, starting from scratch, only with a fuzzy mission statement at hand. The design results include top-level system budget data, configuration drawings, simulations, and scale models. The work is supported by a SSDW-folder with relevant information, design rules, and a well-scripted design process (formulated as "recipes"), allowing even inexperienced workshop participants to tackle the design task. The design task specifically focuses on the following issues:

- Mission analysis, station configuration and human integration
- Launch, assembly and utilization issues
- Attitude and orbit stability and performance assessment
- Life support system analysis
- Power and thermal subsystems sizing
- Assessment of synergistic links between subsystems
- Determination of re-supply requirements
- Determination of microgravity quality as one major utilization objective of LEO

The workshop is made up of two parts. The task of the design part is formulated as a *Mission Statement* including general objectives of the projected space station. A virtual customer presents the Mission Statement to the participants, who play the role of a virtual industry in terms of design teams. As in real projects there are different phases and milestones within the week. First, there is a kick-off meeting for task assignment. The first phase is the requirements engineering followed up by the preliminary requirements review. On this occasion the identified project objectives, requirements and constraints are presented and verified via a question and answer session with the customer and experts. Then, the actual systems engineering process begins, with the initial system design and subsystem budgeting as prelude. This phase ends with the system concept review to present first alternatives and the baseline concept of the design. The final phase focuses on detailed systems and subsystem engineering with simulations and analysis. Preparation of the final system design presentation and the presentation to the customer concludes the design part of the workshop.

Table 2.2: Design sub-teams and task distribution

Sub-team	Tasks
1. Cost, configuration and assembly	▸ Cost analysis and control/cost management ▸ Overall system architecture ▸ Station assembly strategy and planning
2. Mission analysis & AOCS	▸ Orbit and attitude analysis and control strategy
3. EPS & TCS	▸ Power and thermal systems
4. ECLSS	▸ Life support and synergisms
5. HF	▸ Human factors, safety and other crew-related issues
6. Logistics	▸ Identifying/summarizing re-supply needs and logistical planning
7. Public relations & marketing	▸ Marketing strategy, political awareness, documentation and presentation
8. Future	▸ Utilization and station's growth potentials after main objectives fulfilled

The second part of the workshop comprises design evaluation of the teams' findings. The system design presentation and documentation of both former teams are given to all workshop participants. Note that the original design teams are now broken up and the participants form several Evaluation Committees, each referring to special evaluation criteria, mostly related to sub-systems designs. Indicating this break, a new official Mission Statement or rather task description is given. From the educational point of view this part of the workshop emphasises reviewing the design work and all the decisions made on the way to the final design by comparing and analysing the designs differences. In this connection the students have to apply or in some cases even develop evaluation criteria which allow a comparison of different designs or of a given design vis-à-vis the original Mission Statement. This assures that the students are reflecting on their approach and their solutions.

2.2.3 Design Team Considerations

The design teams often consist of people of mixed gender, different cultural backgrounds and various disciplines, mirroring the heterogeneous environment of space business. This multidisciplinary team layout reflects the particular sub-tasks to be solved during the project. Therefore there are several sub-teams and a designated team leader who takes responsibility for proper communication and information flow and consistency between the sub-teams, organization of regular team meetings and making sure that the project goals are achieved properly and in time. The team leader is selected by the design team and can be a member of one sub-team. Table 2.2 lists the task distribution of the sub-teams. Each sub-team should consist of at least two people.

In addition, successful design teamwork builds especially on initiating and organizing the design work and maintaining efficient workflow. The most important design work qualities are summarised together with utilities and their possible influences in Table 2.3. Such interventions should preferably come from the team itself but can also originate from the organisers, or rather the team supervisors.

From the perspective of participants, the workshop is goal-oriented to maximise output with a focus on a proper and feasible space station design concept, but it is also highly process-oriented. Team building, getting acquainted with the problem, identifying the team members' know-how, coordinating the process flow, and last but not least reviewing the design during evaluation are the processes to be coped with.

Table 2.3: Design work qualities and their influencing factors

Design Work Quality	How it can be influenced/Utilities
Dedication of team members	Is prerequisite. Through selection
Trigger motivation	Team itself (through leadership), Organisers (Task, Competition)
Talent to cope with different cultural backgrounds and mentalities	Team itself (through leadership) or Supervisor (Coach)
Talent to integrate all team members and their individual expertise	Team itself (through leadership) or Supervisor (Coach)
Activating and gathering available resources	Team itself (through leadership) or Supervisor (Coach)
Organising team work	Team itself (through leadership) or Supervisor (Assist)

Motivation and Communication

The organisers can motivate the design teams by defining a challenging but feasible design task and by providing a proper working environment with state-of-the-art technical equipment. Another positive effect comes through *competition*: On one hand the design work is highly teamwork-centred, on the other hand it is embedded in a competitive framework where two or three teams work on the same mission statement. Finding a "champion team" at the end of the SSDW simulates the industry competition for orders on the world market and has been shown to have an extremely positive effect on motivation and dedication. It is irrelevant whether the competition and some of the success criteria are artificial. Experience has shown that being curious about what solutions the other teams come up with and the will to be better is a powerful and positive driving force. Certainly this can be adapted and utilised in internal non-educational workshops in industry and the space agencies.

For the majority of the other design work qualities and the role they play depends on the team members themselves and especially on how they *communicate* with each other. Team working qualities of particular relevance are:

- The team hierarchy and leadership, e.g. leader serving the group
- The ability to direct communication flow, i.e. discussing the right thing at the right time, e.g. system-level issues versus details
- Tolerance and the ability to resolve conflicts
- The right mix of team conversation forms

The last point includes team-wide sessions (brainstorming, reviews, decision making), sub-team work phases (subsystem and discipline-specific design work) and time periods for concentrated and undisturbed individual design work. Here, the supervisors can only indirectly influence the work qualities, i.e. through selection of the team members, coaching and assisting the participants during their effort. Doing this one finds that team members with different personal characteristics foster design work quality.

Design Room Set-up

Analogously to the sub-teams layout the team's design room set-up reflects the particular tasks with the primary goal of supporting teamwork and communication. Figure 2.5 shows the proposed basic layout of the rooms as it was used for the SSDW 2002.

In the centre there is a meeting table for joint team meetings. Internal system concept reviews, information exchange and "round-table" discussion are held here. Surrounding this area are the sub-team desks with PC workstations, all networked with a central team fileserver and the Internet. All PCs have the same basic office and Internet software installed. In addition, the PCs have sub-team specific software installed, especially COMET for the configuration modelling, IRIS++ for mission analysis, simulation and AOCS design, ELISSA for ECLSS design and synergy analysis, and others (see section 2.3).

In addition to the sub-teams' PCs there is a team presentation PC connected to a beamer that can be used for sub-team presentations or for displaying overall system information during design work. Other equipment may include different presentation and discussion hardware, e.g. flipcharts, whiteboards etc. Last but not least, the room should be equipped with a drinking water supply and it should be possible to eat fruits and other snacks in the direct vicinity.

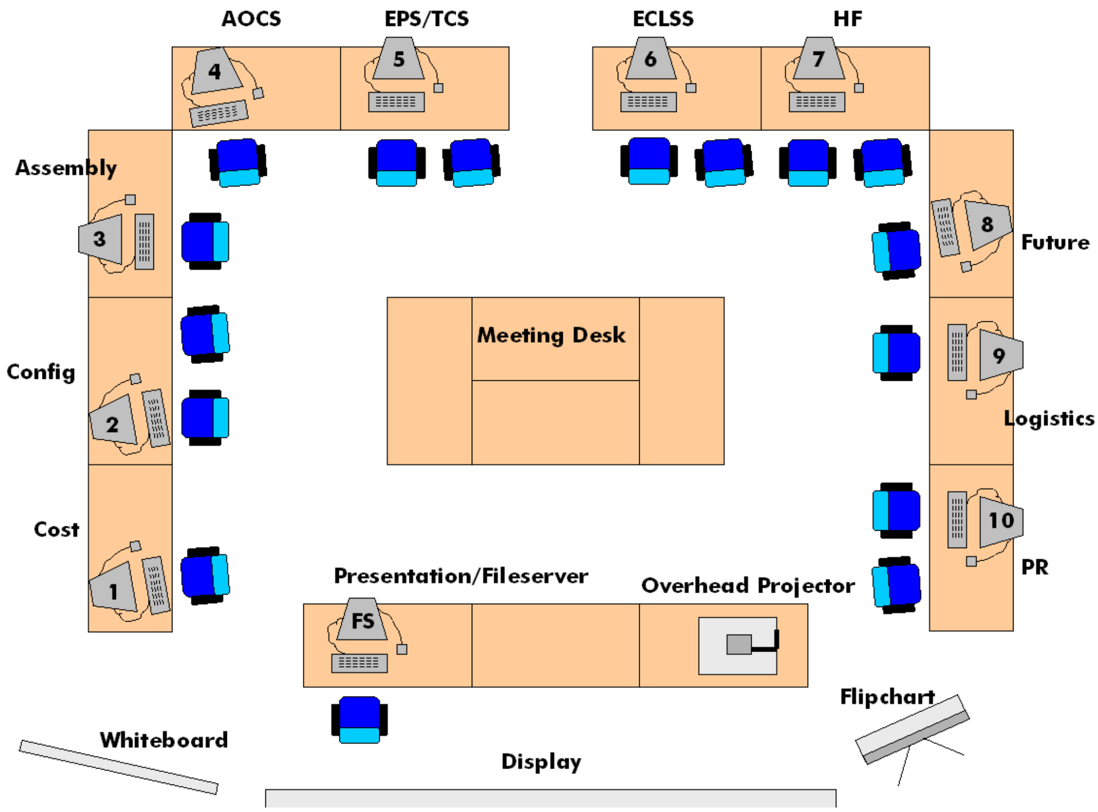


Figure 2.5: Team design room layout

2.3 Conceptual Design Tools

To make the conceptual design of a complex technical system feasible, design and analysis supporting tools are crucial. Table 2.4 gives an overview of different types of tools principally used within conceptual design. They range from formal tools for management support, documentation and communication tools, partly originating from terrestrial architecture with a human-centred point-of-view to characterise the system architecture [Osburg2002], procedural tools leading through the design and analysis process (e.g. "recipes", [SSDW2002]) and analytical tools used in mission analysis and design (e.g. [Wertz1999], [Brown1998], [Bandecchi2000], [SDO]) to numerical software tools used for extensive calculations for mission and system design, characterisation and verification. Often these tools are combined with hybrid implementations in software, manual form or other.

The software developments included within this thesis focused on numerical tools, namely software tools for modelling, simulation and visualisation and analytic tools, namely parametric mission and system design spreadsheets. Other tools are not addressed any further.

Table 2.4: Types of conceptual design tools and their field of design (S: system, M: mission design)

Tool Type	Tool	Field
Formal tools	• Management tools (e.g. work breakdown structures, project planning software) • Creativity and communication tools (brainstorming, sketching, abstracting) • Documentation tools	S/M
Procedural tools	• Recipes (step-by-step procedures, incl. approximation formulas, if applicable) • Interference matrix (incl. system/subsystem links) • Parameter tables (databases)	S/M S S
Analytical tools	• Mission design calculation schemes (e.g. patched-conic calculation, BREAKWELL diagrams) • Budgeting and parametric system engineering tools (e.g. spreadsheet-based)	M S
Numerical tools	• Modelling software (configuration, subsystem functionalities) • Dynamic simulation programs (orbit, attitude, perturbations, subsystems) • Trajectory generation and optimisation • Data analysis and visualisation software	S/M S/M M S/M

SSDW Software Tools

Software programmes are indispensable for a rapid and efficient conceptual design process, especially if several iteration loops – and thus repetitive execution of numerical analysis tasks – are required. Besides the modelling of space systems, geometry (topology and topography) and performing flight simulations, visualisation of resulting data and illustrations of resulting design concepts are important tasks. Because the SSDW is so design-intensive and because of the short timeframe, the tools to be used during the workshop must meet certain prerequisites:

- Easy to learn and intuitive user interface
- User friendly and reliable operation
- Fast calculation and appropriated data output for analysis and visualisation
- Configurable to adapt to various design problems
- Compatibility with a common operating system
- Modular software design to insure easy further development and software maintenance

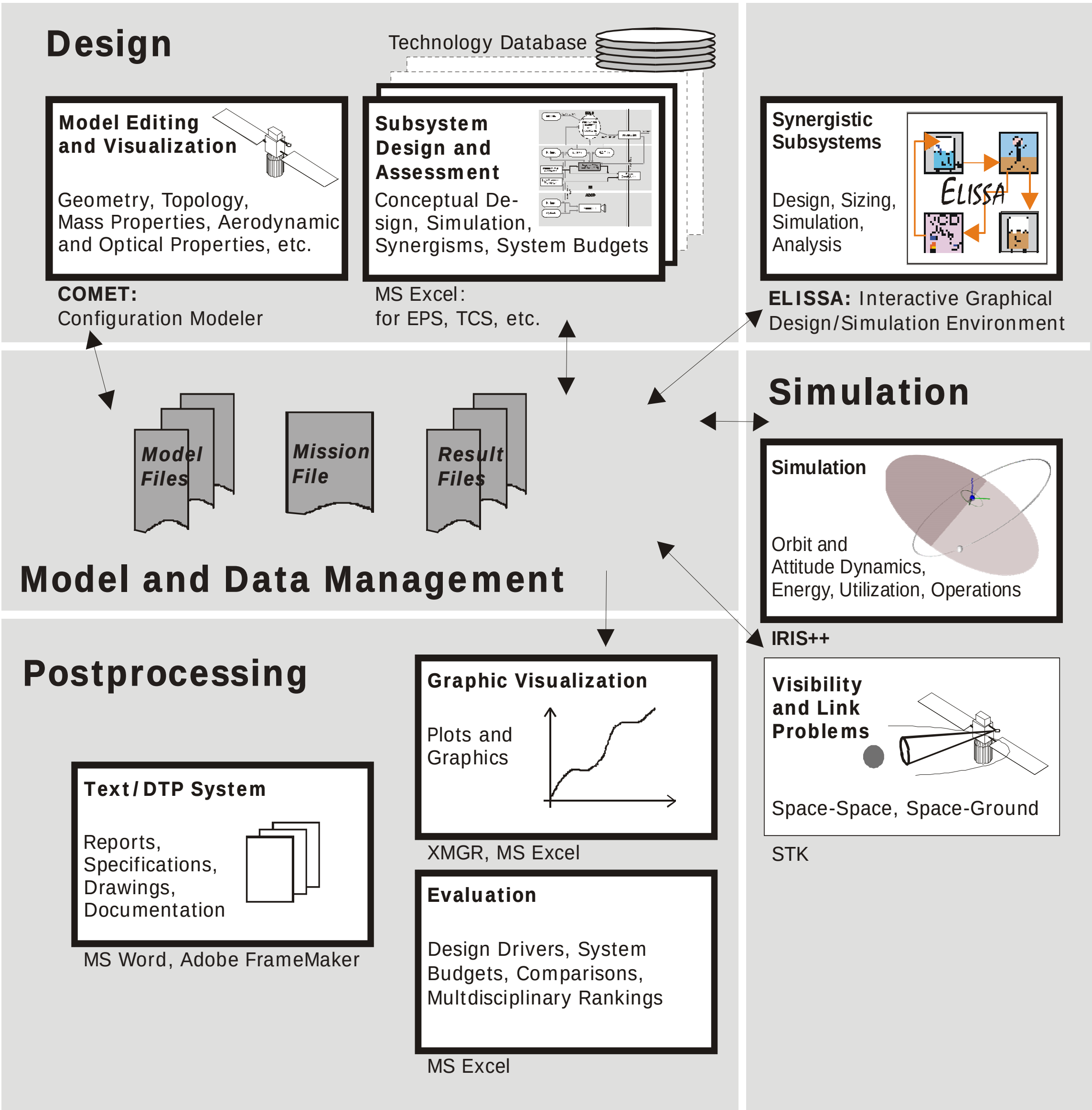


Figure 2.6: SSDW's software infrastructure

The package of software tools currently in use consists of custom-developed software dedicated specifically to the conceptual design of space stations as well as commercially available general-purpose software. The operating system used is *MS Windows*. Figure 2.6 presents an overview. Four areas can be identified: *Design*, *Simulation*, *Postprocessing* and the *Model and Data Management* area as the common interface.

The major computer programmes developed at IRS within these areas are:

- COMET: used for design of the space station configuration/modules and for visualisation.
- ELISSA: Modelling, design and simulation of life support systems and synergistically linked subsystems
- IRIS++: numerical system simulation programme, including spacecraft's orbit and attitude dynamics with various perturbations taken into account.
- Other supporting software (spreadsheets application, etc.)

Figure 2.7 illustrates how these tools are embedded regarding the design process.

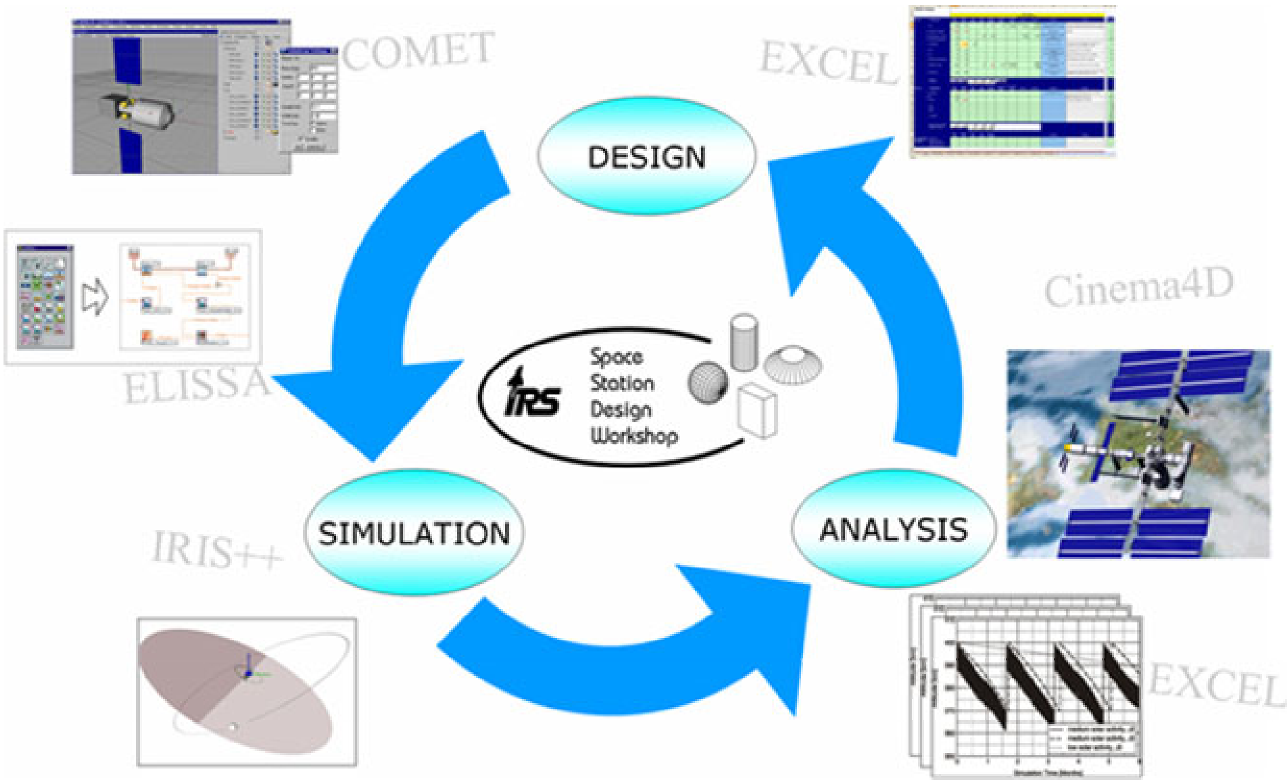


Figure 2.7: Main tools within the SSDW conceptual design process

Software Evolution

The tools used within this investigation are new developments or based on SSDW tools previously developed, which had to be adapted to the new requirements for future space station scenarios beyond LEO:

- COMET is a new software programme and was designed and developed from scratch during this study. A detailed overview is provided in chapter 3.
- IRIS++ software is based on a prototype [Yazdi1998] and a release version that was numerically restricted to near-circular LEO missions [Hinüber2002]. During this study it was completely refurbished, including major extensions of the environmental models [Drodzek2002] and perturbation calculation methods and enhancements of the simulation core and the numerical methods (e.g. [Demenet2002]). A detailed overview is presented in chapter 4.
- ELISSA was previously developed [Osburg1998]. For this study its components library was extended and now includes advanced life support system elements, namely a number of biological and other components for food production, waste treatment, water recycling and air regeneration [Ganzer2004]. Brief descriptions of these components are included in chapter 5.
- Supporting software, such as spreadsheet application based on *MS Excel* [Excel], were defined and standardised, enabling rapid simulation result visualisation. Among others this includes an analytical design tool for designing lunar transfer missions [Lutschinger2002] and a subsystem conceptual design assistant software [Karagah2002]. Cinema4D is used for 3D-visualisation of space station configurations.

Mission scenarios beyond LEO require more complex environmental effects to be taken into account during the conceptual design process. Thus, especially mission design and spacecraft flight simulation is of concern. Here, existing effects at possible future space station locations that are of relevance for system/subsystem design and analysis must be modelled. Simultaneously the range of applications increases drastically. Here the rationale for the chosen approach can be seen, namely developing highly generic software tools, meant to be “general purpose” to cover most human spaceflight scenarios for educational purposes and detailed enough in order to provide valuable information for research purposes.

3 Modelling and Visualisation of Space Stations

Modelling is the process of creating substitutes of a to-be-developed (space) system in order to predict and analyse its properties and behaviour when implemented and operated. Models are the only available tools on ground to verify a spacecraft design and the chosen approach to problem solution. For the specific area of the system (overall system, sub-systems, and functionalities) and the model depth of detail, one can differentiate between physical and virtual models.

While during satellite development classically three different physical pre-flight engineering models (EM) are built for structural, thermal and electrical tests and verification, new approaches try to reduce the number of physical models to save associated costs and time. Besides combining models (e.g. to Structural and Thermal Models, STM) and using test items as actual flight hardware, available computer performance offers extended possibilities today. Indeed CAD models, finite-element methods (FEM) and other numerical approaches allow detailed structural and thermal analysis and performance of all major tests and concept verifications virtually, thus replacing one or two physical models. Only acceptance tests must typically be performed with the physical Engineering and Flight Model (EFM). Furthermore, recent developments have made high-fidelity spacecraft system simulators available for the development and verification phase, including virtually every element of the actual spacecraft down to the single component level, such as switches, wires and connectors [Eickhoff2004]. Even data transfer protocols are modelled allowing onboard computer to be simulated and analysed. Although the development effort of such an infrastructure is enormous in the first place, the benefits regarding cost, schedule and design maturity for follow-up projects can be expected to be tremendous, if an appropriate generic approach for model design is used. This is basically true for models used for the conceptual design phase, although a much lower level of detail and higher abstraction level is required.

To achieve the objective of developing a *multi-spacecraft* and *multi-mission* simulation environment, for preliminary property and performance analysis as detailed as necessary for concept evaluation and selection, a possibility has to be found to create simple, but effective and generic computer models allowing quick-turn-around modelling, simulation and analysis.

This chapter discusses the subject of system modelling of spacecraft in order to enable system-level analyses with dynamic simulations. While this study concentrates on system-level modelling, sub-system specific modelling in this context has been covered in previous investigations [Osburg2002]. The following section, 3.1 gives an overview of the configuration modelling approach selected and implemented during this research and section 3.2 describes the new COMET software and the modelling process using this tool. Section 3.3 addresses the geometrical spacecraft surface model developed, which is necessary for accurate dynamic simulation of spacecraft models.

3.1 Configuration Modelling

Simple configuration modelling can be achieved by constructing only a simplified geometrical representation of all structural bodies, by sketching a configuration manually or with support of common 3D graphics or CAD software. If the configuration model shall also be used electronically for more detailed technical analyses via computer simulations, additional information has to be included, e.g. mass properties, basic functionality and mechanisms. As described in section 2.3, the software's utilization during the conceptual design workshop requires an appropriate, generic and easy-to-use tool. Because commercially available software packages do not offer such features, such a tool was developed in the course of this study.

The configuration modelling, i.e. simplified structural modelling, is based on simplification with geometrical primitives. Even a random complex geometrical scene can be assembled with only a few types of primitives, namely spheres, boxes, cylinders and cones [Bertrand1998], which can be described with just a few parameters determining their position, orientation and dimensions. Compared to polygon-based modelling, this leads to a simple and small scenery description enabling easy maintenance and re-use of scenery parts for other projects.

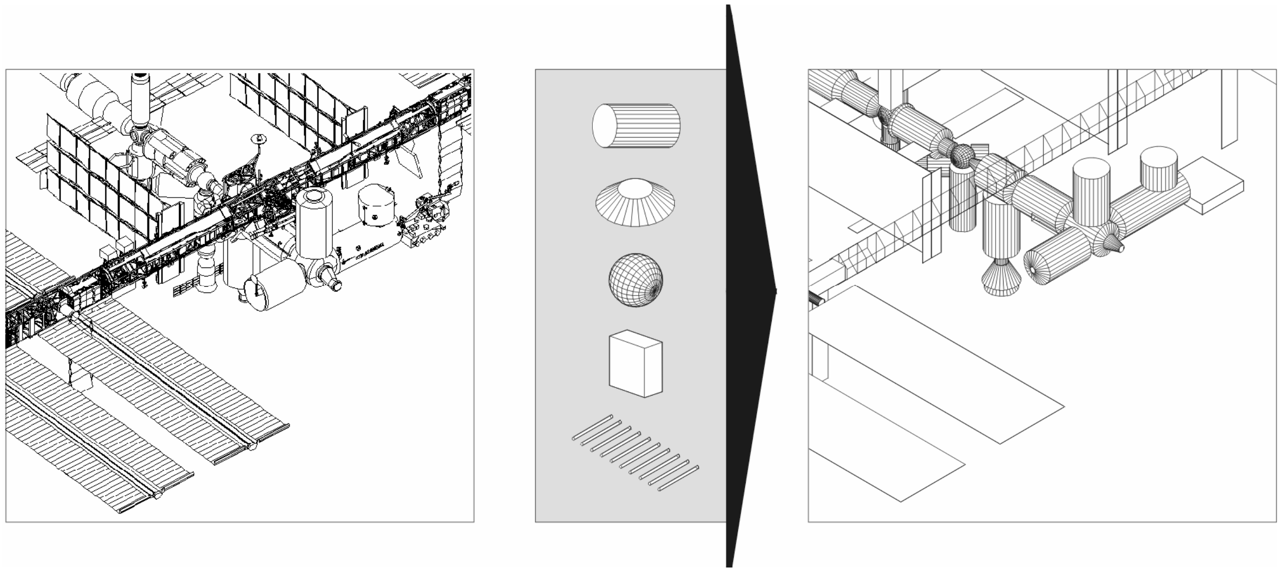


Figure 3.1: Simplified geometrical modelling using primitives [Bertrand1998]

As illustrated in Figure 3.1, space stations, the geometrically most complex spacecraft, can indeed be modelled this way, maintaining geometrical properties of the original structure. In addition to the geometrical and topological data of the spacecraft elements, mass property information must be included as well, which are the mass, the position of its centre and the inertia tensor of the represented physical body, be it solid or hollow. These properties influence e.g. the accelerations and momentums acting on the spacecraft in space. One important source of influences are surface forces, due namely to the aerodynamic and the solar radiation pressure. The prerequisite for properly taking these effects into account is a geometrical surface model (in addition to the parametric model above) and a shadowing algorithm. Dynamic simulations require flexibility and fast calculation, which has to be taken into account during model development.

Modelling Software Requirement

The requirements of the modelling tool are defined by its range of applications. These include:

- Modelling of modules and external configurations of space stations and other spacecraft
- Accommodation analysis of internal and external payloads
- Validation of spacecraft approach clearance
- Visualisation

Visualisation is relevant because illustrations are necessary for documentation and presentation of architectural designs. The modelling process results include a geometrical model, thus, it is efficient to also use this model for illustration purposes. Therefore, the modelling tool should include appropriate picture generation capabilities or an interface to appropriate software.

To meet these requirements the *Configuration Modelling and Editing Tool* (COMET) was developed.

3.2 Configuration Modelling and Editing Tool (COMET)

COMET was designed for modelling and conceptual design of space station modules and configurations. It is a proprietary add-on to the commercial 3D graphics software *Cinema4D* available from Maxon GmbH [Maxon]. COMET provides an intuitive user interface and a design environment with assistants for rapid graphic modelling and design and provides a convenient export filter to generate space station configuration files compatible with the spacecraft simulation software IRIS++ (see chapter 4). This enables quick-turnaround simulations with various space station configurations during the conceptual design process. Furthermore, due to the excellent visual capabilities of the underlying graphics software, the developed configurations can easily be rendered and animated for documentation and presentations.

3.2.1 Featured Tasks for Space Station Design

To summarise, the following basic modelling tasks of a modelling software tool are covered:

- Creation of primitive bodies and the specification of their geometrical properties (size and dimensions, i.e. length, radius, wall thickness, etc.)
- Specification of the mass properties of these objects (mass, centre of mass, inertia tensor)
- Specification of the other properties of structural elements (surface properties, mechanisms, etc.)
- Definition of the position and orientation of the primitives with respect to a space station fixed coordinate system (body-fixed coordinate system, BCS) or with respect to other primitives in order to create a configuration of primitive bodies
- Functional designation of specific structural elements (solar array, radiator, thrusters, tank)
- Saving the model data in a file format compatible with the simulation software

The design of a new space station configuration is typically based on re-using existing or using modifications of existing groups (or *modules*) of primitives and re-positioning these modules with respect to each other. Therefore, additional tasks were defined and implemented to enable a convenient and efficient design process:

- Grouping of primitives to modules and using these modules as one entity
- Storing of modules and other groups of objects in an object library for re-use

- Positioning and alignment of primitives and modules in respect to each other
- Specifying a name and a comment for each object, to specify its type, purpose, model version and data reference

With an appropriate object library, these applications can also be extended to an internal design of manned spacecraft. This has already been demonstrated [Irani2001].

3.2.2 Modelling and Software Engineering Approach

COMET is an object-oriented software programme. It uses primarily three types of object classes (i.e. spacecraft, module, primitive), which are ordered hierarchically: primitives make up modules and modules make up a space station. The system elements are therefore modelled in a hierarchical object tree.

All objects have a custom-defined data container including mass properties and other information relevant for dynamic simulation. These data can be specified and edited by the user during the design process for each individual object. During simulation data specified on primitive-level will be processed to calculate module-level data and this module-data will be processed to obtain the resulting data on spacecraft-level. Hereby data specified by the user in a higher hierarchical-level (e.g. on module-level) has a higher priority as in lower levels (e.g. on primitive-level) and will override it. This means user-defined module-data will be used instead of obtaining them through processing low-level data. This principle allows geometrical modelling of existing modules and using their *real mass properties* provided by the manufacturer.

To support calculation and assessment of the microgravity level/quality at specific locations within space station modules during simulation, the possibility to define *microgravity (µg) sample points* is provided. This enables payload/experiment accommodation assessment in laboratory modules.

Furthermore, basic mechanisms can be defined on primitive and module levels in order to realise one and two-axis rotation of bodies towards a target point during simulation (e.g. alpha and beta-tracking of solar array panels towards the Sun).

3.2.3 Components of COMET

The main programme window of COMET is depicted in Figure 3.2 and shows the Cinema4D user interface with COMET menu extensions. The window layout can be re-defined. In its standard layout the editor view is located in central position. The view controls are located at the upper-right corner of the view, allowing panning, rotation, zooming, and changing the view-mode (e.g. one 3D or three-view display). Located above is the menu and the toolbar with buttons for frequently used functions, including file access (open, close, load, save, export), object control (generation and “docking” of objects), and object editing mode (rotation, panning, axis control switches). On the right, there is the hierarchical object tree display of the *Object Manager* with a separate menu including file access to individual objects and object editing functions. Below the editor view there is the *Material Manager* view, where visual properties of objects can be defined. In the bottom-right corner is the *Coordinates Manager* supporting direct numerical input of object position and orientation.

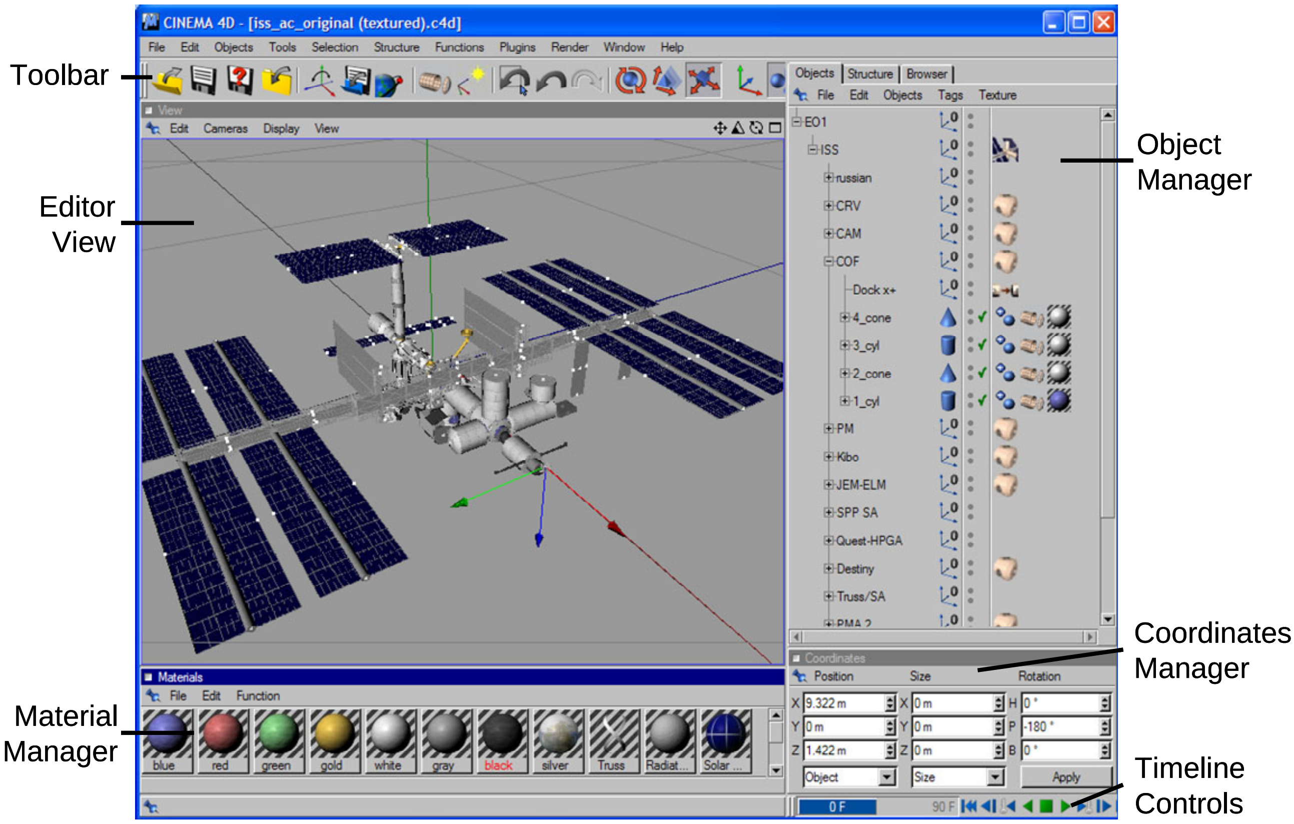


Figure 3.2: Main user interface of COMET software

In addition to the principal user interface provided by Cinema4D, the modelling software COMET extends the software with its following components:

1. Graphical User Interfaces (GUI) for managing space station objects
2. COMET object class structure
3. Object Generator for creating various space station elements
4. Docking Assistant and Dockpoint Positioning Module for convenient object positioning and alignment
5. Object library containing main elements of the ISS and various transfer vehicles
6. Output filter to export space station configurations as IRIS++ model files

Additional modules and support infrastructure are:

7. Data container for geometric, mass and functional properties
8. Event-triggered update functions for convenient user-object interaction
9. Tracking preview and verification routine
10. Import filter to load IRIS++ data files
11. Trajectory visualisation module

Figure 3.3 illustrates the interactions of main COMET software components embedded in the Cinema4D software environment. The following sections will provide a comprehensive overview of COMET components. For a detailed description the reader is referred to the COMET user manual [Yazdi2003b].

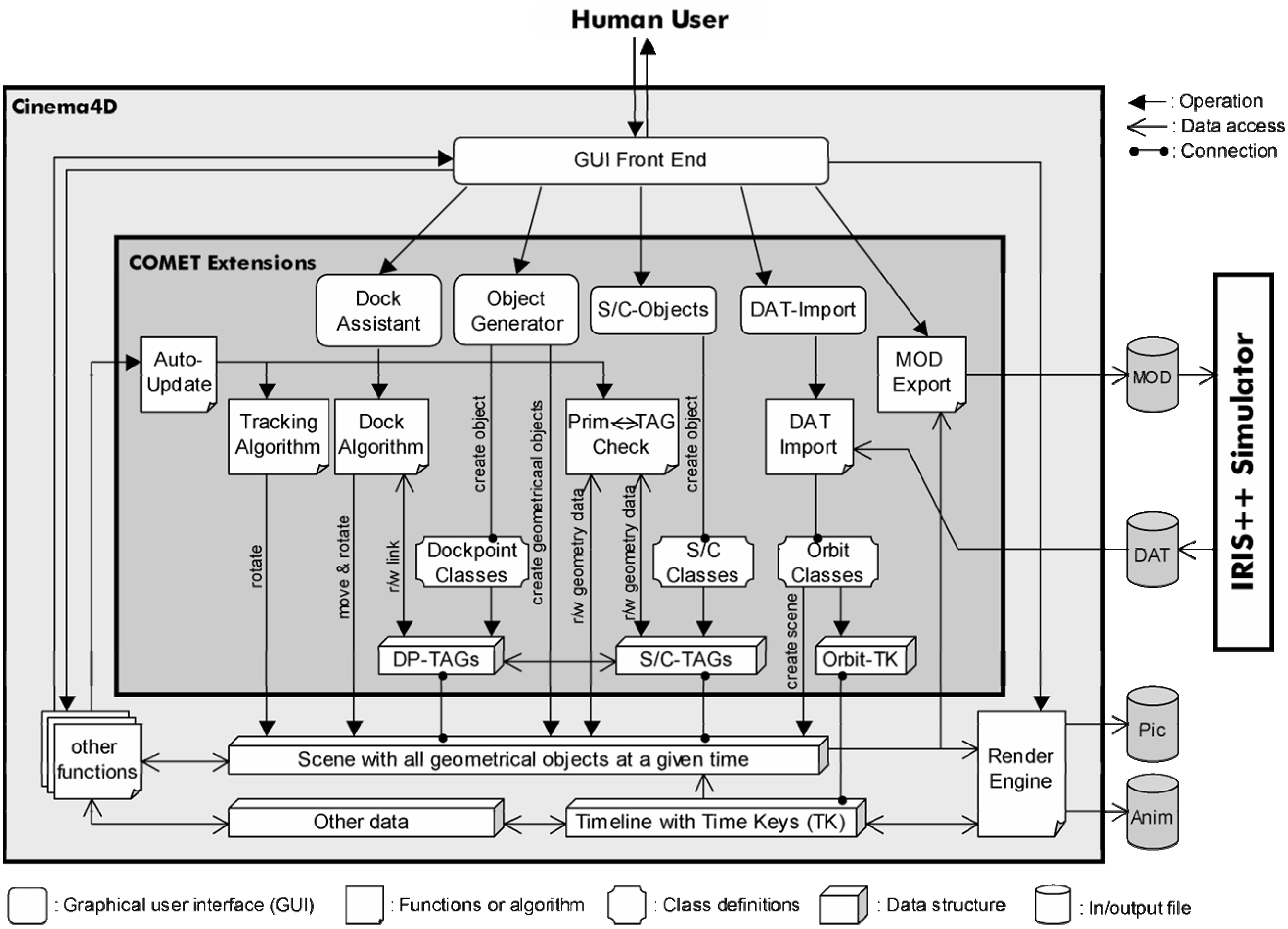
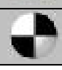


Figure 3.3: COMET software architecture

Object Generator and spacecraft object classes

COMET defines specific object classes including formal spacecraft objects listed in Table 3.1. The Object Generator creates these spacecraft objects, and allows the user to define their properties and locate them automatically in the object hierarchy.

Table 3.1: COMET's principle object classes

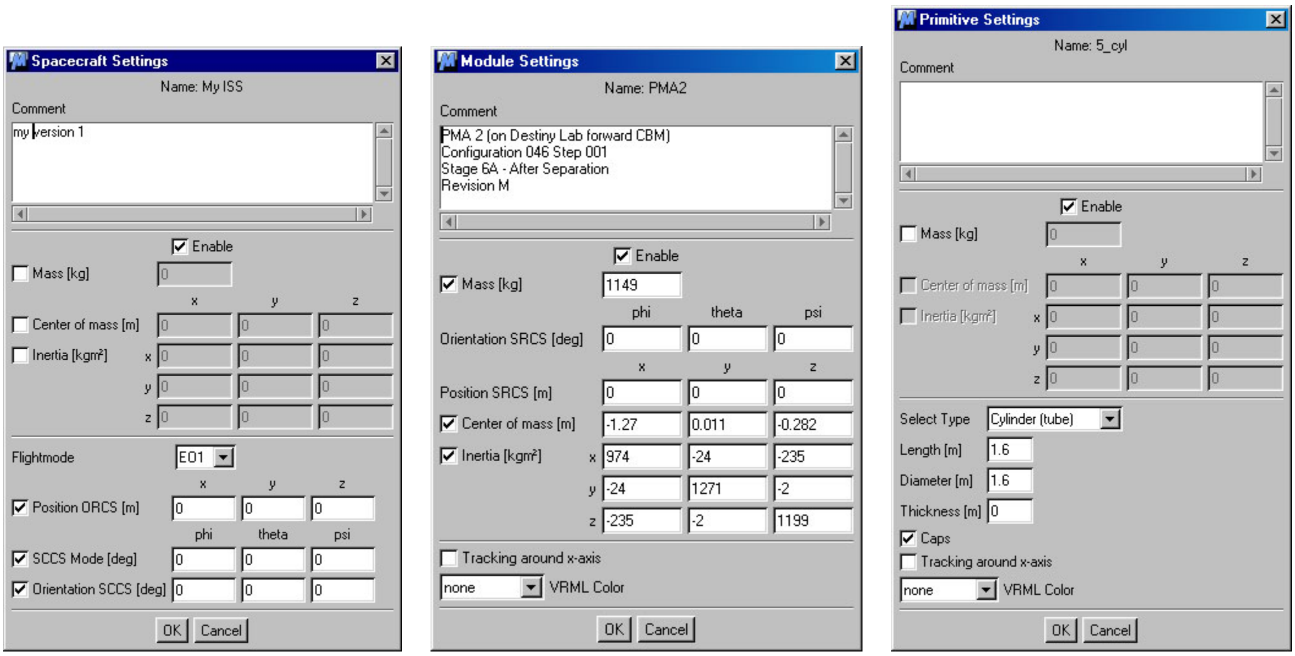
COMET object class	Icon
Spacecraft	
Module	
Primitive	
Dockpoint	
μg Sample Point	

Except for primitives, all COMET objects, including Dockpoints and μg-Sample-Points, are abstract objects consisting only of a coordinate system (position and orientation) plus an object-class-specific tag defining behaviour and properties with parameters stored in a data container. Its specific icon represents the object class. Through this icon the user has access to the information stored within these data containers via separate GUI windows (see Figure 3.4).

Table 3.2: COMET's primitive classes

Primitive classes	Icon	Geometrical parameters
Cylinder		diameter, length (x)
Cone/Truncated Cones		top (x+) and bottom (x-) diameter, length
Boxes		x-,y-,z-length
Spheres		diameter

In addition to this data container, primitives also have a parametric-geometrical shape description associated with them and which is provided by Cinema4D (Table 3.2). An additional geometrical parameter in COMET is the *wall-thickness*, when a hollow primitive is used instead of a solid. Figure 3.4 depicts the GUI parameter settings of the COMET main object classes controlled by their specific tag. The upper section is reserved for comments to the object written by the user (e.g. version number, source of the objects properties or recent modifications). The next section contains the mass property data, such as mass, the location of the centre of mass (CoM) with respect to the body coordinate system and the inertia tensor. This information is all optional, but at least the mass must be given for primitive objects, as the IRIS++ software will calculate missing module and spacecraft information out of primitives' data. This implies that entering mass property data for module and spacecraft objects will override data derived from lower levels, i.e. primitives and modules. In the third section at the bottom, object class specific entries are found. For spacecraft objects (Figure 3.4.a) we can define the flight mode (i.e. Earth-oriented or inertial) and the relative position and orientation in the orbital reference coordinate system (ORCS). Modules and primitives have a checkbox for activating automatic tracking and a data field for the object's output colour in the VRML file generated by IRIS++. In addition, primitives provide type-specific data fields for geometrical properties and the sub-type (solid, hollow, etc.).



a) b) c)

Figure 3.4: COMET object settings

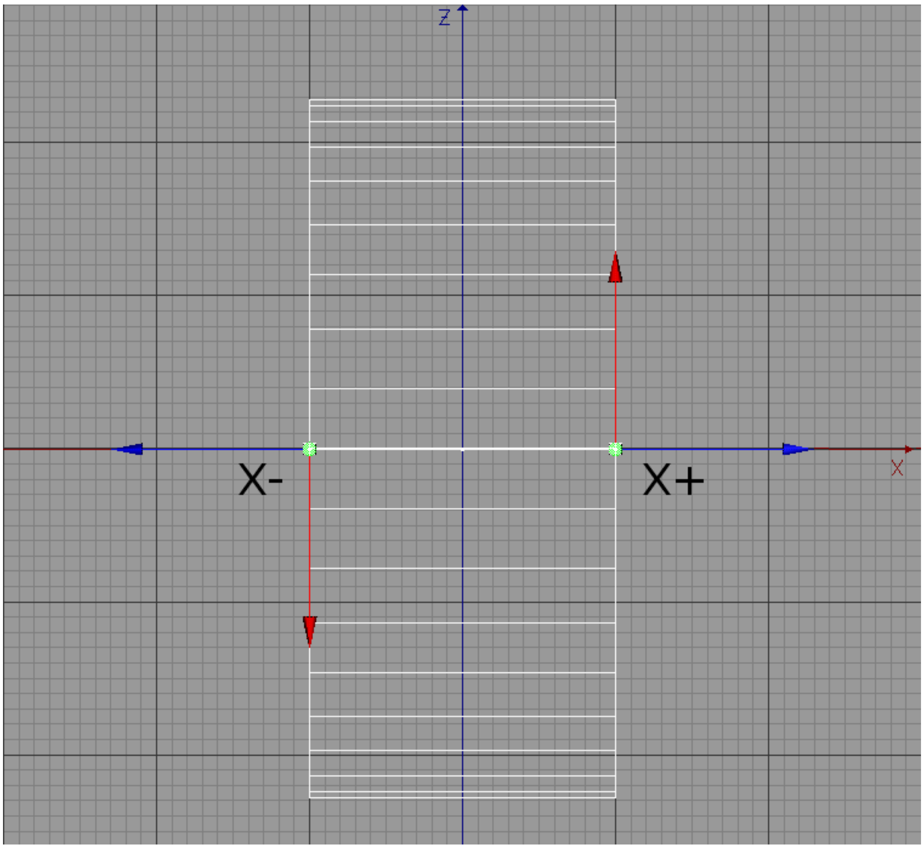


Figure 3.5: Cylinder primitive with dockpoints (x+ and x-)

The Dockpoint infrastructure

Creating modules out of primitives and building configurations by modules incorporates moving and orienting objects in respect to each other. To support this activity and enable quick and convenient workflow, the possibility of *docking* was introduced with COMET.

For this purpose the Object Generator automatically creates a number of so-called *Dockpoints* for every primitive. Cylinders, for example, are equipped with two default dockpoints at both abutting faces on their longitudinal x-axis (see Figure 3.5; red on screen), boxes have six default dockpoints at each quad, and so forth. Dockpoints are always on the surface of the primitive body, with the z-axis (blue) pointing outside, commonly 90 degrees to the local horizontal plane.

Furthermore, the user can add custom dockpoints via the Object Generator or create *instances* of existing dockpoints. These are copies but will automatically follow all modifications of their original, independent of where they are located in the object hierarchy.

To avoid the patience-exercising task of positioning and orienting user-defined dockpoints on the surface of their object properly, a *Dockpoint Positioning Module* is implemented. With this tool the user can interactively drag dockpoints across primitives' surfaces, while the algorithm maintains on-surface positioning and normal orientation at all times via live-update, even for non-planar surfaces.

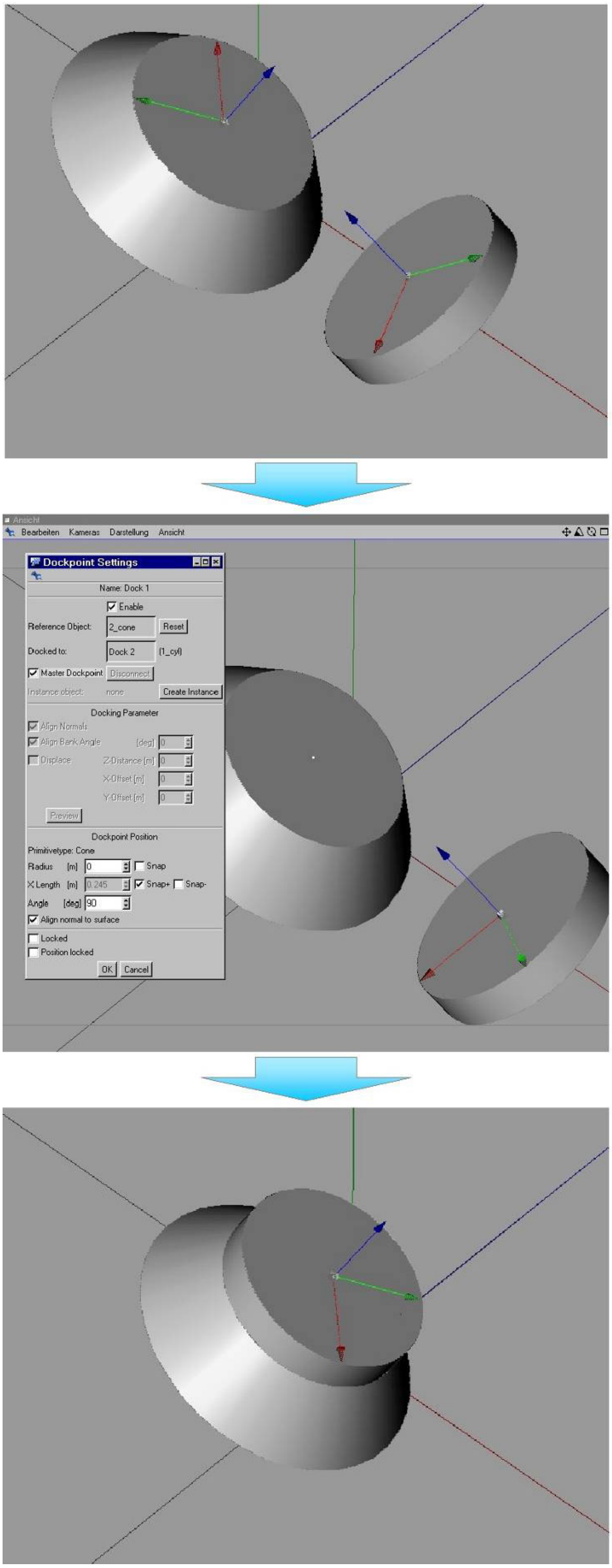


Figure 3.6: Docking of two primitives

Using the Docking Assistant

Using two dockpoints, one at each of the two objects to be connected, an algorithm was implemented to re-orient one of the bodies such that both dockpoints will join with opposing z-axes. Figure 3.6 shows how this process is accomplished with two simple steps: Selecting and opening the properties window of the dockpoint to be connected of one primitive (e.g. the small cylinder), then selecting the dockpoint of another primitive (e.g. the large cone) such that the first dockpoint is connected by drag-and-drop to the appropriate drop-field. After approving the sequence by clicking the OK-button the small cylinder will be connected.

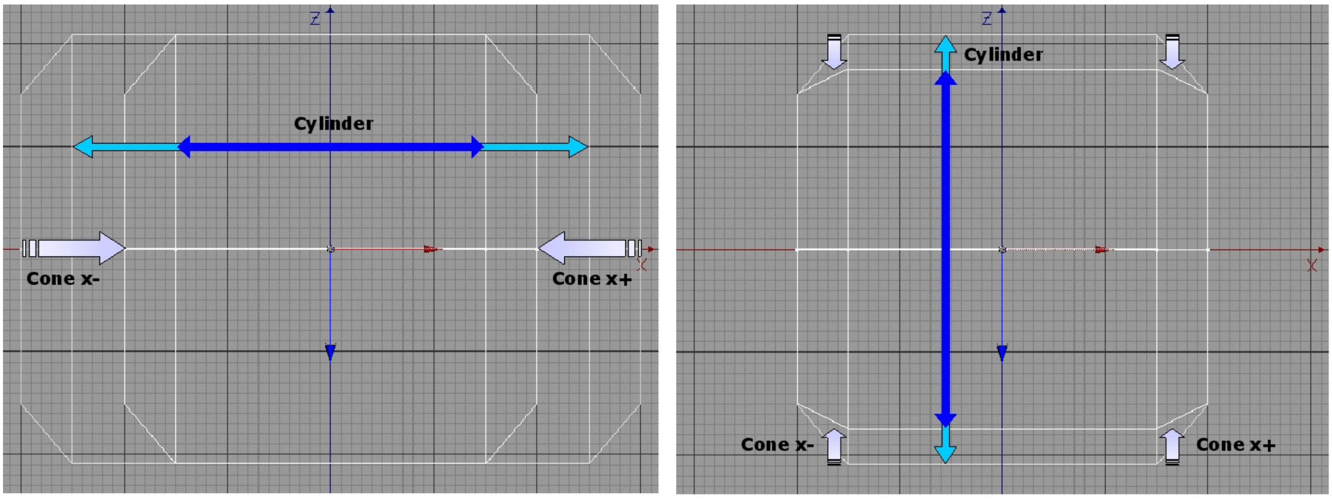
It is important to know that this connection is not only a simple geometric alignment of two bodies done once, but a true master-slave-connection between them. The moved body (the slave object) is virtually attached to the reference body (the master object). This means, if the user moves or rotates the master object, the slave will follow. However, the slave object cannot be affected directly in this way anymore without first disconnecting it from its master. This update of *truly* connected objects is performed by repeating the docking-routine automatically for every connected dockpoint of modified or consequently affected bodies.

Auto Geometrical Update

Another advantage arising from this concept of inter-dockpoint linkages is the possibility of maintaining a consistent geometry and proper topological distribution within a group of linked objects.

For creating a new space station configuration, the designer often wants to use existing hardware, namely previously created modules, and modify the length or the diameter for instance. Because the primitives "know" about each other through these dockpoint connections, a routine was implemented to automatically distribute changes from one object to the next. Figure 3.7 illustrates an example where the user changes the length of the central cylinder in the cone-cylinder-cone group (case a). After approving this change, the cylinder re-locates its default dockpoints automatically and re-initiates a docking-routine for each connected dockpoint. Like the positioning and rotation update above, now the docking operations are repeated iteratively to update all other affected objects. By using instances of primitive dockpoints for module-module connections, the same convenient automatic update-functionality applies, if changes to the modules' primitives occur.

In addition, primitives can optionally transmit geometric information through their connected dockpoints. E.g. with changing the diameter of a cylinder (case b, Figure 3.7) and if the connected objects are geometrically compatible (e.g. both are round: cone-cone, cylinder-cone) and have the same size at their interface initially, the new geometry will also be set for them. Thus, user time is saved because changing geometrical information manually at all connected bodies is not necessary. Both automatic update functions accelerate the modification process of existing modules enormously and help maintain consistent geometry throughout the design.



case a: topological update case b: geometry update

Figure 3.7: Automatic update of linked COMET primitives

3.2.4 Visualisation

Based on Cinema4D's professional visualisation functionality, some additional routines and sample scenes come with COMET, providing convenient 3D visualisation capabilities of spacecraft configurations and dynamics. Furthermore IRIS++ (see chapter 4.1) provides an output module for visualisation purposes. In summary, visualisations are currently used for:

- Configuration visualisation and verification
- Payload accommodation analysis
- Display of 3D trajectory data by importing simulation data coming from IRIS++ simulations
- Rendering of stills and animations

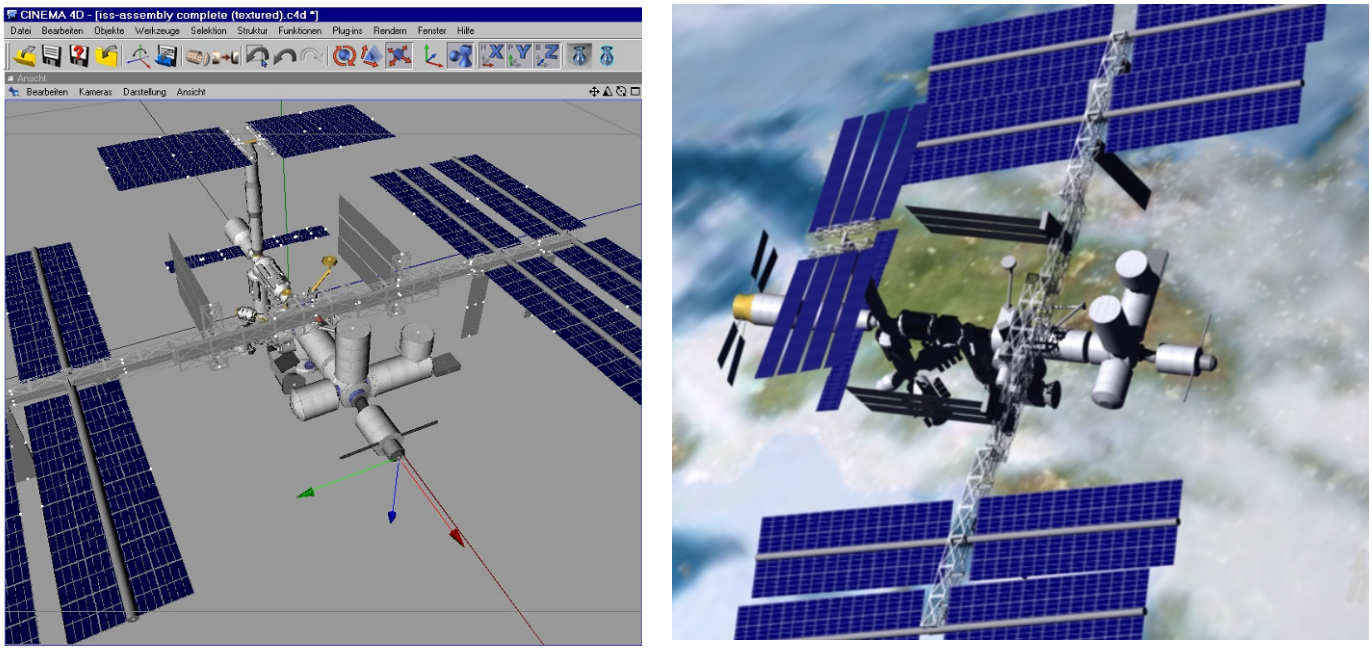


Figure 3.8: Applying visual properties and rendering of stills and animations with COMET

The following section will demonstrate these applications through examples.

Configuration visualisation: Rendering of stills and animations

Figure 3.8 depicts a COMET model of the International Space Station (ISS) with applied visual properties (e.g. texture maps and colours) to the primitives with respect to their functionality. Inserted into a *world scenery* including objects visible in the orbital environment, one can easily produce still renderings or animations to illustrate a conceptual design and its operation. Doing this the designer can principally also use actual computed data (i.e. orbit and attitude information produced by IRIS++) to visualise simulation results.

Payload accommodation analysis

One convenient method to analyse and verify the accommodation of external payload is offered by visualisation of the 3D scene. In this connection the geometrical spacecraft model is used to virtually attach an experiment or a device, such as an optical camera or an antenna, to the spacecraft hull or the truss of a space station. By setting up a Cinema4D camera with user-defined properties, the design engineer can easily obtain the view field of the instrument at the payload location and orientation. Hence, obstacles or other influencing components can be identified.

The operation of the *Global Time Services* (GTS) of the ISS communication experiment mounted on the Zvezda module, for instance, showed irregular behaviour with the acquisition of signal (AOS) when passing the GTS ground station in Stuttgart. Taking the actual ISS configuration and looking from the GTS perspective sitting on nadir side of ISS-Zvezda module, reveals the problem. As depicted in Figure 3.9 the view of the antenna is partly shadowed by the Docking Compartment One (DC1) and a docked Soyuz capsule. Each visible circle marks 10° of the view field. With this information paired with ISS orbit and attitude data the real AOS behaviour can be predicted accurately.

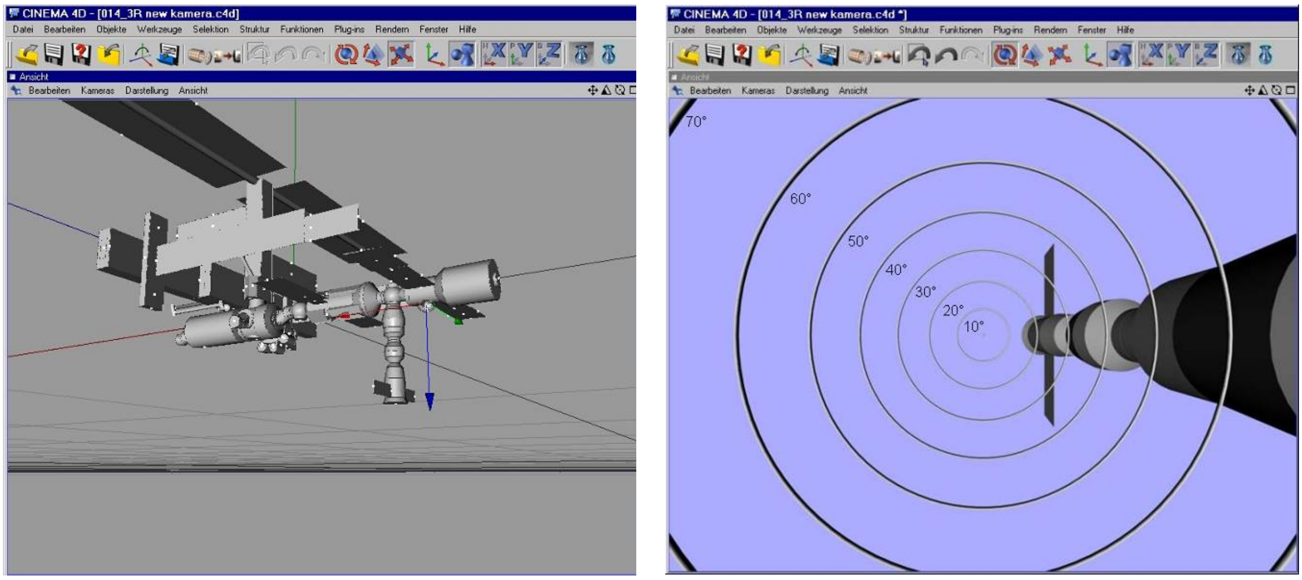


Figure 3.9: GTS view field analysis using COMET software

Data and Trajectory Visualisation

For data visualisation and analysis of simulation results, spreadsheet application Data Visualiser (DAVIS) based on MS Excel is used (Figure 3.10). However, this type of visualisation is limited to 2D plots, thus, some simulation parameters including trajectory data (position and orientation) cannot be visualised satisfactorily. Hence, a Cinema4D import filter was implemented allowing reading and processing of IRIS++ simulation data files and generating an animation scheme. This experimental COMET module provides appropriate visualisation of 3D information, including orbital trajectory evolution of spacecraft or celestial bodies (see Figure 5.3, page 85).

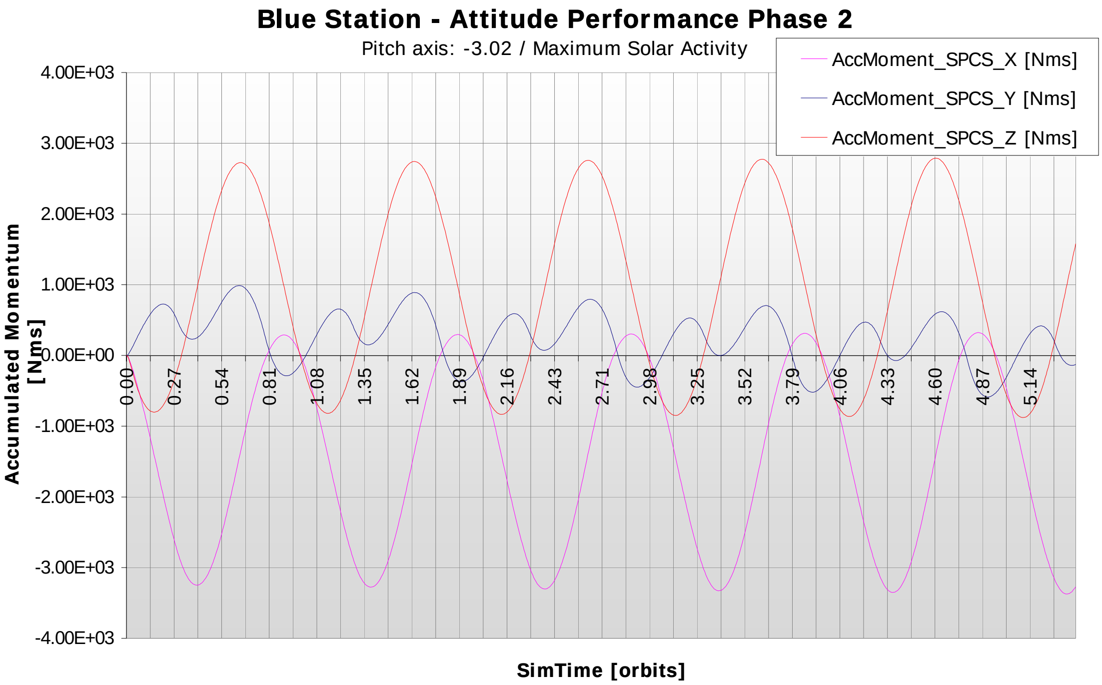


Figure 3.10: Visualisation examples: Data profile illustration, using DAVIS spreadsheet package based on MS Excel (data shown: TEA attitude performance of team Blue/SSDW2002)

3.3 Surface Modelling

For the simulation of spacecraft orbit and attitude dynamics and the calculation of environmental effects on utilization and subsystem operation, the surface-interface is of key importance. Irradiations of different kinds interact with the spacecraft surface and have an influence on various environmental parameters. These parameters depend primarily on surface irradiation conditions with respect to a "radiation source", i.e. the Sun, the aerodynamic flux of particles of the residual atmosphere, or the material flux of micrometeoroids and orbital debris (section 4.5). In all cases shadowing effects can also play a major role and must be taken into account.

Thus a surface model is a prerequisite for accurate spacecraft simulation that ideally enables and supports the fast computation of the following parameters at the same time:

- Projection area (effective irradiated area)
- Lightning conditions of single areas
- Location of the centre of pressure (CoP)
- Normal vector perpendicular to a surface element
- Contour of the projection area

Such a surface model was developed and integrated into the flight simulation software ([Yazdi1999],[Fritz2003]). This section summarises the approach, the chosen methodology and its benefits concerning the calculation of surface forces and torques, evaluation of the operational conditions of components such as the electrical power system (EPS) and the thermal control system (TCS), namely solar panels and thermal radiators, and obtaining impact probabilities of certain spacecraft surfaces. More information is given in [Yazdi2001].

3.3.1 Surface Model Requirements

Pairing the constraint of fast computation with the wide range of applications, the surface model must meet the following requirements:

Variable accuracy

In conceptual design an overall accuracy of approximately 10% can be taken as sufficient. Structural elements of a spacecraft complex can therefore be modelled with primitives as described in section 3.1. Thus, small elements of structures and attachments will be neglected. With the developed model, a possibility is obtained to control the level of detail of the surface description to enable fast estimation runs as well as more accurate, but little more time consuming, simulations.

Generic surface model

The method of surface discretisation must be applicable and the generated model must be accessible generically within the simulation software. This means all calculations requiring surface-related data should be able to use geometrical information of this one and only surface model. This should not be restrained by the discretisation-method. Furthermore, extending the range of simulated types of bodies should be possible.

Support of generic shadowing algorithm

Depending on the application, the focus of calculating irradiation or flow conditions will lie on single components of a complex structure (e.g. solar irradiance) or on the overall surface (e.g. aerodynamic flux). For the incorporated algorithms and methods this imposes that they are applicable to both types of calculations and that single surface elements can be identified after the shadowing computation.

3.3.2 Surface Discretisation

To access the surface of a random spacecraft configuration, its structure is modelled with the help of geometrical primitives (boxes, cylinders, cones and spheres). 3D surfaces are subdivided into a number of finite planar surface elements by an automatic discretisation process. These sub-surfaces approximate the three-dimensional and convex shapes of the original primitive bodies. A single primitive surface model is based on a finite number of vertices with a network of planar polygons in-between. The mathematical bodies created this way are polyeders (see Figure 3.11)

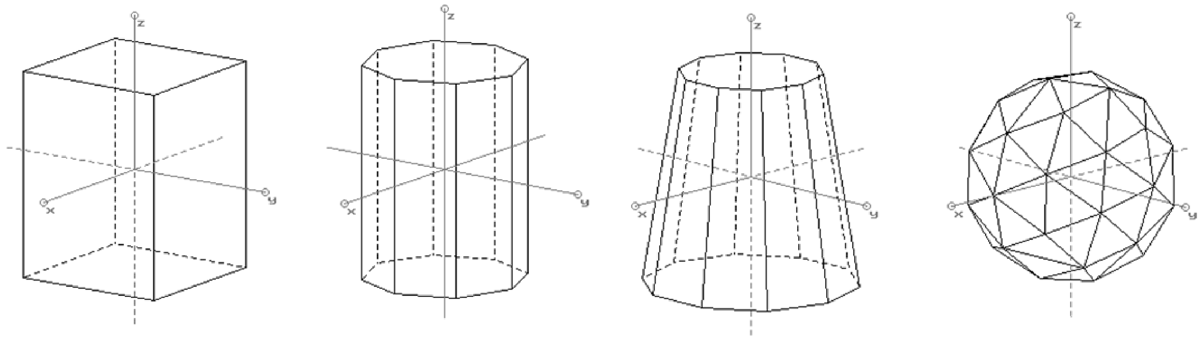


Figure 3.11: Surface models of considered primitive bodies

The more vertices and thus single surface elements (polygons) are taken, the better the shape and contour of the bodies is described by the discretisation. Increasing the number of vertices entails an increase of calculation time and computer memory allocation. To control the modelling accuracy, a variable grade of discretisation was chosen here. In addition to controlling overall accuracy, this also enables modelling of bodies with a lower level of detail when being small compared to the overall spacecraft and less important for shadowing computation.

Boxes

A box is the simplest primitive and consists only of planar surfaces. Therefore, its polygon surface model is described exactly. All box objects consist of 8 vertices, 6 polygons and 6 normal vectors indicating the outward facing side. Boxes are used to model solar panels and radiators.

Cylinders and Cones

Cylinders can be seen as an exceptional case of (truncated) cones. Because the results can be directly applied to cones, only cylinders will be discussed here. To discretise cylinders, one can create two identical polygons describing the abutting faces, whereby the circumference is divided into a certain number of identical segments with the generated points taken as vertices. Thus, the front faces are modelled with circular and regular p-point polygons and the coat face is modelled with the sum of p circumlocated quad-shaped polygons. The more points are used for the front faces, the better the cylinder-shape is described. With the number p of points for one cylinder front face, the model has a number of n vertices and f polygons and normal vectors:

(1)	$n = 2 \cdot p$	with	p	: number of points for each front face
(2)	$f = p + 2$		n	: number of vertices
			f	: number of polygons/normal vectors

Cylinder and cones are typically used to model pressurised modules and truss structures and are used very often in the case of space stations. Therefore, it was worth examining the relationship between accuracy and number of vertices and their configuration. An optimisation method resulted from this that applies to spheres analogously and is described in section 3.3.3.

Spheres

There are different possibilities to discretise spheres. The aim is to achieve a preferably good and homogenous description of the sphere's surface geometry with a minimum number of vertices. Unfortunately, an analytical method does not exist to equally distribute a random number of *points* over a spherical surface. For sphere-like objects this is possible in exactly five cases only, leading to the bodies described by PLATO. Hence a new process has been designed and implemented and which is illustrated in Figure 3.12. Firstly, a numerical-iterative method distributes a user-defined number of points on a sphere semi-equally by simulating repulsive forces between them. Then, a triangulation method described by BARNHILL and FOLEY [Barnhill1991] was adapted to generate a network of triangles wrapping the discretise sphere. This approach leads to a very accurate and flexible result, enabling convenient control of detail level via selecting the number of vertices freely. With a number n of vertices the number f of triangles equals:

(3) $f = 2 \cdot (n - 2)$

This time-consuming iterative surface generation process can be performed prior to simulations. Therefore, a separate small tool programme was used to create normalised spherical models with 8 to 64 vertices and to store them as surface model data (SMD) files read by IRIS++.

Spheres are typically used to model tanks and other spherical shapes. Other potential bodies are two-dimensional curved bodies like spherical or parabolic antennas ("dishes") for which the same method can easily be adapted. For details refer to [Yazdi1999].

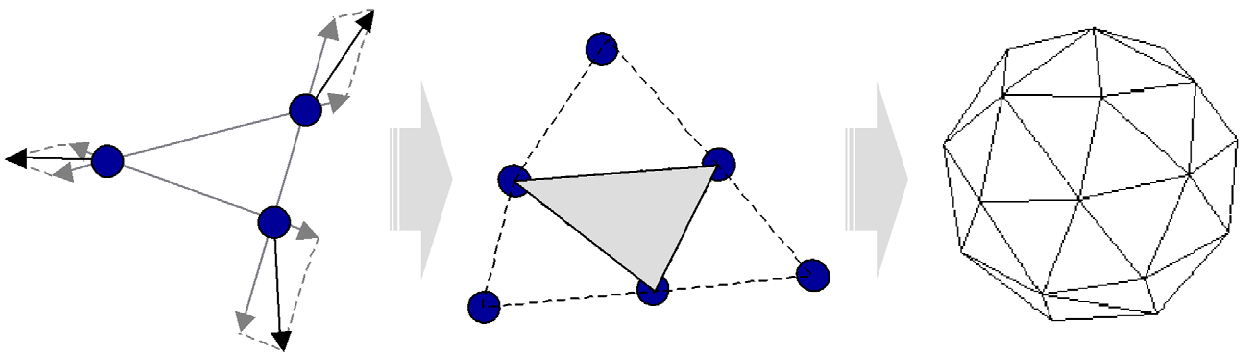


Figure 3.12: Surface model creation process (point distribution and triangulation)

3.3.3 Surface Model Optimisation

In this section the developed surface model optimisation method is described. It is used within the simulation software to simultaneously enhance cylinder, cone and sphere models in terms of the accuracy of the projected areas and reduce computer time. The underlying principle applies to all bodies of these types. Examining the relationship between the number of vertices used for discretising e.g. a cylinder reveals that the effective area with respect to a given direction, namely the projected area, depends on the relative rotation of the model in relation to the projection line. Figure 3.13 illustrates this variation, which increases by lowering the number of vertices.

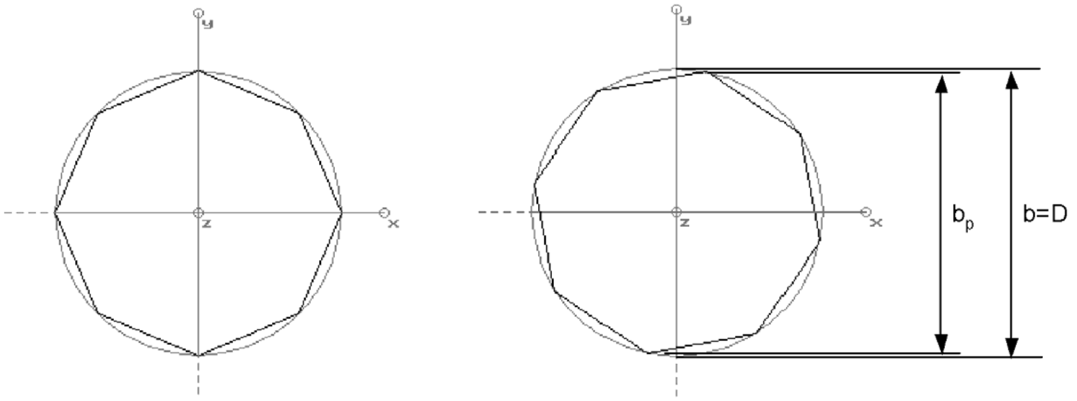


Figure 3.13: Effective height variation of a cylinder model with respect to a symmetric projection (left) and a slightly out-of-symmetry rotated projection (right)

The projected mantle as well as the front face of the model is obviously always smaller than the original, except in the special case of symmetrical projection (a). For sphere models this is also true. Here the polygons cut border volumes and border regions of the projected area. Therefore, the projected area of these bodies will always be smaller than their original primitives and will be accumulated over the total number of bodies used. This leads to a considerable deviation *to the unsafe side* when calculating surface forces, with effective areas and thus, the resulting forces too small. The problem increases with increasing complexity of the structure. A radius-adaptation rule is introduced to compensate this effect. Using a *radius correction factor* ***k***, the average projection area equals the projection area of the original body. Table 3.3 lists the correction formulas for which derivations can be found at the stated reference.

Table 3.3: Adaptation rules for the radius correction factor [Yazdi2001]

Cylinder/Cone	Sphere
(4) $k_{Cylinder}(n) = \dfrac{\sqrt{2\cdot\pi}}{3\cdot\sqrt{n\cdot\sin\left(\dfrac{2\cdot\pi}{n}\right)}} + \dfrac{2}{\left(\cos\dfrac{\pi}{n}+2\right)}$	(5) $k_{Sphere}(n) = \dfrac{F_{Circle}(r)}{\overline{F}_D(n,r_D)} = \dfrac{\frac{\pi}{2}\cdot r^2}{\overline{F}_D(n,r_D)}$ with $\overline{F}_D$: numerically calculated mean projection area of discretised sphere

Figure 3.14 and Figure 3.15 show the achieved enhancements for cylinder/cone and spherical surface models. With this correction the deviation falls below a given limit at far fewer vertices than without correction, thus leading to a reduction of surface model complexity. For example,

to achieve an accuracy of below 10%, 36 vertices are needed for a sphere without optimisation. With adaptation, 10 vertices are sufficient. Furthermore, Figure 3.15 shows that with the adaptation the deviation of the effective area distributes equally around the analytical correct value (some areas are too large, some too small) but vanishes statistically.

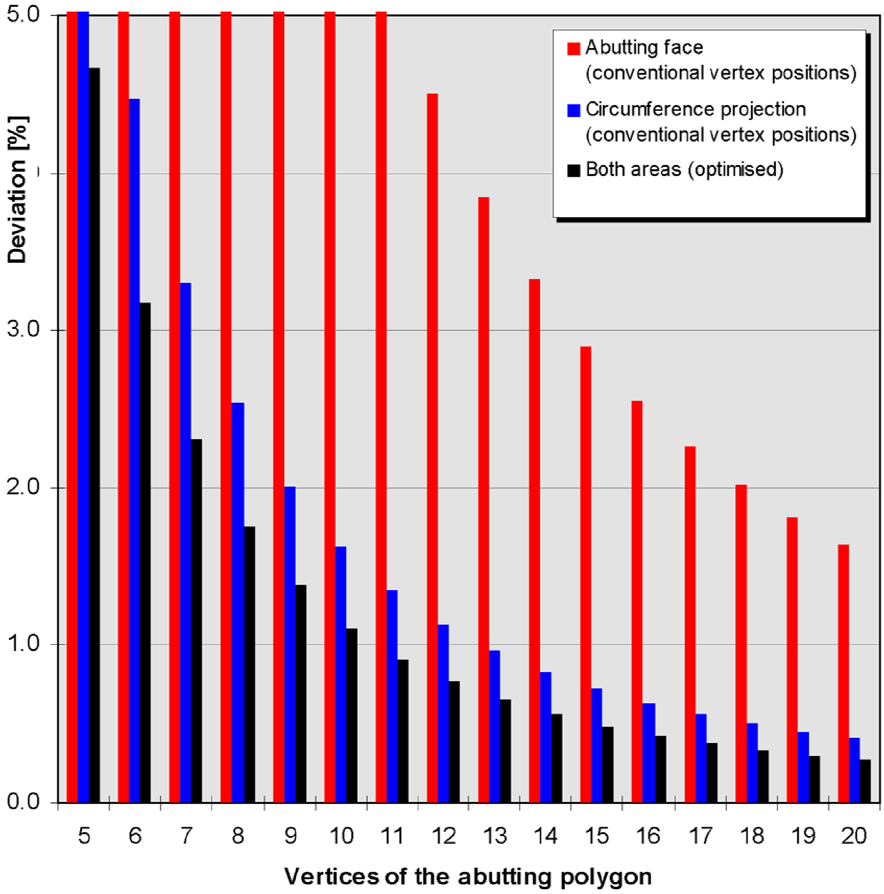


Figure 3.14: Deviation of cylinder models

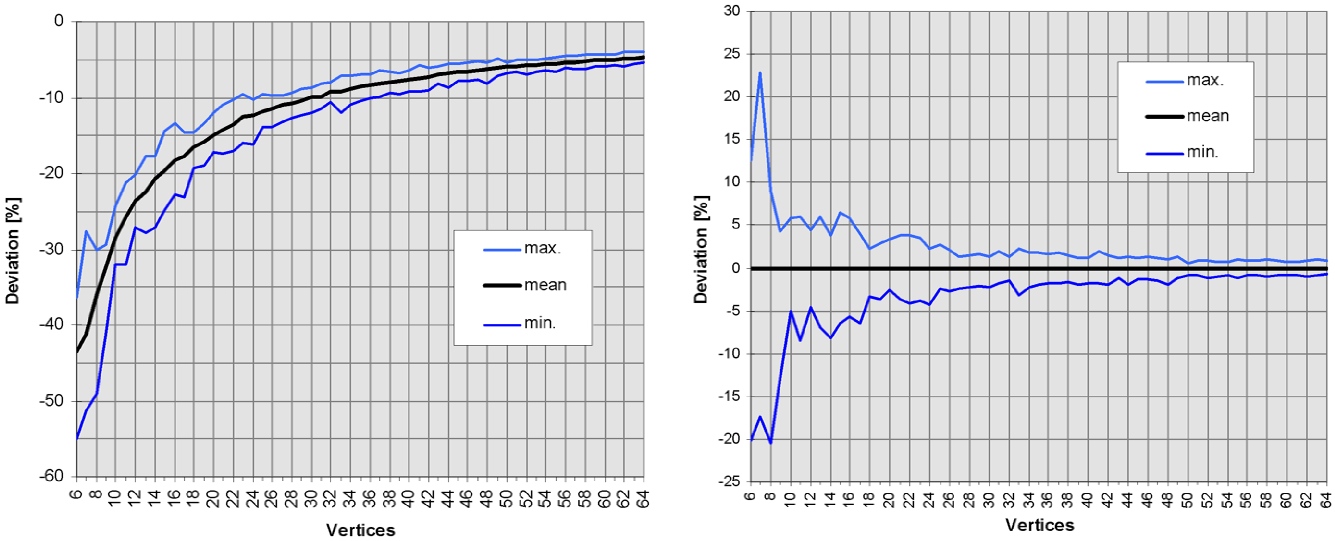


Figure 3.15: Deviation of sphere models (left: without, right: with adaptation)

4 Spaceflight Simulation

A spacecraft flight simulator is an important tool for the process of designing space systems. Given a sufficient range of simulation capabilities it enables the design engineer to preview the actual behaviour of the system concept within the projected mission scenario. In this connection, the spacecraft itself and the environmental effects are modelled at an appropriate level of detail. The simulation tool has to provide data output of all relevant variables in order to allow simulation analysis and thus mission evaluation.

Due to the large range of possible mission scenarios, the varying number of influencing factors to be taken into account to propagate the spacecraft system dynamics, and the limited computer performance, only the mission design and analysis departments of industry or agencies typically develop and use simulation tools. These tools are high-fidelity specific simulation software, mostly addressing a specific design problem (e.g. attitude control or trajectory optimisation), an individual mission scenario or phase (e.g. geostationary orbital mission or interplanetary cruise phase) or a to-be-planned manoeuvre (e.g. gravity-assist or aero-capture). Thus, for each specific problem a different tool is taken for analysis. Used by a small group of spacecraft dynamics experts, these tools do not generally feature convenient user interfaces.

Because of the changing space mission design methodology (i.e. concurrent engineering) more generic simulation tools are needed, covering a wide range of missions and used to analyse different problems. Due to increasing computer performance, such advanced tools have become feasible in recent years. Their development and application has begun, but currently existing tools still concentrate more or less on one of the disciplines: orbital mechanics, attitude dynamics or the spacecraft sub-systems.

System-level flight simulation software requirements

Dealing with conceptual design, the whole range of mission and system design needs to be addressed by an easy-to-use and highly generic simulation tool. Such a tool should cover all relevant scenarios within a family of missions (i.e. space stations and associated transfers) using one spacecraft model and reliably calculating all relevant mission and system design parameters at an appropriate level of detail. These parameters include top-level system and subsystem design parameters. In summary, spacecraft flight simulation preferably focuses on:

- using generic top-level system and subsystem models, and calculating
- orbital mechanics and
- attitude dynamics, both propagated with
- numerical integration methods, and all influenced by
- environmental models addressing perturbations and subsystem operations.

Altogether these elements determine the spacecraft state and operation as function of time. In order to support further development and enhancements of the included spacecraft and environmental models, this tool must be of modular design and easy to maintain. The flight simulation software developed within this thesis is IRIS++.

4.1 Flight Simulation Software IRIS++

IRIS++ version 2.0 is the spacecraft simulation programme developed and used here as command-line software for the analysis of dynamic space systems. Inputs are the geometry and mass distribution of a space station or platform, to be provided by the graphical modelling tool COMET (see section 3.2), and the simulation commands defining the simulated mission specification. Figure 4.1 illustrates the workflow when using IRIS++. Outputs are stored in data files for post-processing, e.g. in spreadsheet applications and 3D visualisation.

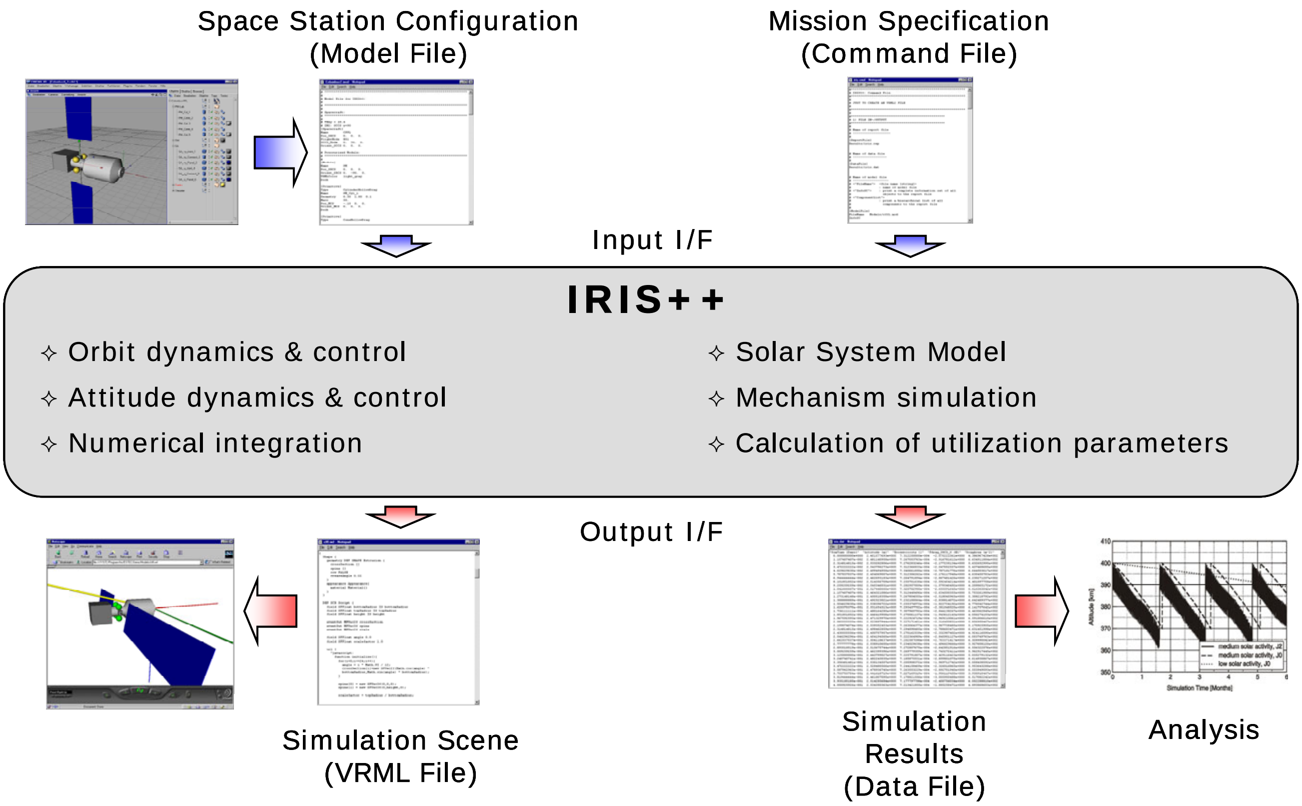


Figure 4.1: IRIS++ spacecraft flight simulator operation

Because IRIS++ is used for educational purposes (i.e. SSDW) and space systems research, it emphasises generic applicability to a wide range of spaceflight missions. Although emphasis lies on space station applications, IRIS++ provides a generic multi-spacecraft and multi-mission simulation environment. Today simulation scenarios dealt with include Earth, Moon and Mars orbital missions and cis-lunar as well as interplanetary transfer missions. The following features are currently included:

- Command interface to specify mission and control attitude and orbital operations
- Attitude dynamics, based on the numerical integration of 6D-Euler equations of motion and attitude control using actuators (momentum wheels and thrusters)
- Orbit dynamics, using cartesian and equinoctial parameters with orbit control, allowing control strategies using permanent and impulsive thrust
- Various numerical orbit integration methods featuring single, multi-step and step size control

- Perturbations, taking the following sources into account:
  - Planetary gravity potential field (up to 5th order)
  - Residual planetary atmosphere,
  - Third-body gravitational forces from multiple, arbitrary celestial bodies,
  - Solar radiation pressure, and
  - User-defined forces (e.g. thruster operations)
- Dynamic motion model of the solar system including all major planets and the Moon
- 3D surface model and shadowing algorithm for calculation of incidence areas (aerodynamic drag, solar radiation pressure) and subsystem environmental properties (illumination on solar panels and thermal radiators)
- Simulation of structural dynamics (tracking of panels)
- Top-level simulation of following subsystems: AOCS, EPS, TCS
- A Virtual Reality Modelling Language (VRML) output file containing a 3D-snapshot of the space station under analysis for quick configuration/attitude display and verification.

In addition to the state variables, perturbations and model information, specific output data include:

- Microgravity levels at user-specified locations within the spacecraft
- Shadowed ratio of solar/radiator panel active areas
- Accumulated momentums stored in momentum wheels
- Propellant consumption due to attitude or orbit control thruster operations

Subsystem parameters, such as consumable fluids in the ECLSS, are provided by ELISSA software (section 2.3).

4.2 Software Engineering

IRIS++ originates from the orbit simulation programme IRIS, which development begun in 1990 using Fortran77 programming language [Huber1990]. In the following years during further developments, the programme became more and more complex and opaque with every step, limiting practicability. In 1997/1998 a complete re-design and re-implementation of IRIS was studied, prototyped and dubbed "IRIS++" indicating the usage of the object-oriented computer language C++ and the associated software engineering progress [Yazdi1998]. The first implementation was completed as version 0.9 by HINÜBER in 2002 [Hinüber2002]. Since then, the software has undergone major enhancements and extensions in the framework of this thesis, reaching the current version 2.0.

4.2.1 Object-Oriented Design (OOD) and Programming (OOP)

Object-Oriented Design (OOD) is a relatively recent design approach in developing software. For designing IRIS++, the primary purpose was to model data and programming structures using real-world analogies. The practice of programming using OOD is called Object-Oriented Programming (OOP). The purpose is to design software in such a way as to make a simple interface, preventing the user from being affected by eventual changes to the internal operations of individual modules (e.g. classes). Furthermore, OOP helps to design software that is extendable. Object-oriented designs emphasise code concentration to points where new programmers can add new functionality to existing software with minimum effort and re-using as much previously coded software as possible.

To fully understand the programme structure introduced in section 4.2.1, this section gives a brief overview of OOD and is intended to emphasise its core features and associated advantages for software development and maintenance. On this occasion the fundamentals of *object-orientation, classes* and *objects* connected with OOP are introduced. Details and extensive language support have been documented by ECKEL [Eckel1989], SEED [Seed1996] and others.

Object-orientation

In contrast to the classical, sequential-linear design methodology of computer programming, object-orientation is associated with three keywords: *Encapsulation*, *Inheritance* and *Abstraction*. Each keyword is related to its application to objects and their properties, which are defined through the data and programme structure.

Encapsulation

Using classical programming languages (e.g. C, Fortran77 or PASCAL) one can define general data structures that contain different information through variables and data arrays. It is possible to associate these data structures with objects, created and deleted during run-time. The properties of these objects have to cooperate with the stored information respective to their data structure. This affects a large number of different routines typically distributed over the whole programme structure. Examples are routines for displaying information about objects on screen or saving them to a file. Other routines may seek specific data about objects in a database, etc. Thus, all programme routines have to be adapted to all these objects types in order to handle them properly.

Using object-oriented programming languages (e.g. C++), data structure and the associated programme structure are brought together. Such a joint structure is called *object class* or *class* for short. It declares not only the object variables (data) but also the proper routines and functions to handle them (methods). This leads to encapsulation of programme structures with specific data. The major advantage is that programmes become modular, easily extendable and maintainable.

Another aspect is the encapsulation of object data when defined as *private* or *protected*. Because these data structures are not freely accessible within the programme, access methods serve to monitor and control reading and writing access. This enables not only information validity checks but appropriate update functions to maintain data consistency as well.

Encapsulation leads, therefore, to enhanced reliable programs, especially if they are permanently extended and modified. Furthermore, benefits arise from the encapsulation of programme routines and object specific data from the perspective of copyright, because only necessary source code files have to be made available to a programmer with using the rest as a *black-box*.

Inheritance

During the object definition process, often a large similarity to other types of objects, which are already defined, is noticed. A large number of objects can even be slightly modified or extended versions of an existing object. Thus, object-oriented programming languages offer class definitions by inheriting object properties of previously defined classes. This means the properties of the new child class, i.e. its variables and methods, will be derived from a *parent class* and will be directly available. In addition, the programmer can include further variables and methods and can even modify or replace existing methods of the parent class by re-defining them. This is called *overloading of methods*. Changes to the parent class definition will directly affect all relevant parts of their child classes automatically. It should be remarked that in popular OOP-languages, neither the class structure nor the object structure has to be one-dimensional. Child classes can be derived from multiple parent classes and inherit their properties. Thus, inheritance is a powerful tool enabling efficient designing of complex hierarchical object class structures.

Abstraction

In a software programme often all objects of an object class tree must be managed and accessed centrally. This type of access is achieved in OOP-languages by declaring all necessary methods of the child objects of the tree within their common parent class. The definition is not included because these methods depend on the individual child class and are therefore encapsulated within. Thus, the functions are defined *virtually* only by their name, type and parameters. But such parent classes containing virtual methods become *abstract classes* and cannot be instantiated, i.e. no objects can be created from this class directly. Corresponding to object-orientation, abstraction makes it possible to create code that need not be concerned about *how* specific subroutines handle different object classes. They must only "know" that an object of this class has this specific method. How this method is implemented and does its job is defined in the child class.

Object class vs. Object

In the previous sections the terms *object classes*, *classes*, *objects* and their structures are mentioned. Often, these terms are not differentiated in literature, although they are of key-importance in understanding object-orientation and software development this way. Therefore this section briefly defines these terms.

Object classes, or *classes* for short, are definitions of properties and behaviours of an object or a group of objects within a programme. The class definition is therefore the specification prototype of the later objects. It includes variables and methods.

Objects are instances of object classes. This means, when an object XYZ of the class cXYZ is created, computer memory is allocated to store data corresponding to the variables defined within the class definition. Since this step at run-time, a data/programme complex (the object) called XYZ exists to work with.

An *object class structure* is often a hierarchical structure of classes that is built by inheritance from parent classes to child classes. Thus, the class structure is directly defined by the definition of the class during software development. It may also be called *(object-) class hierarchy*.

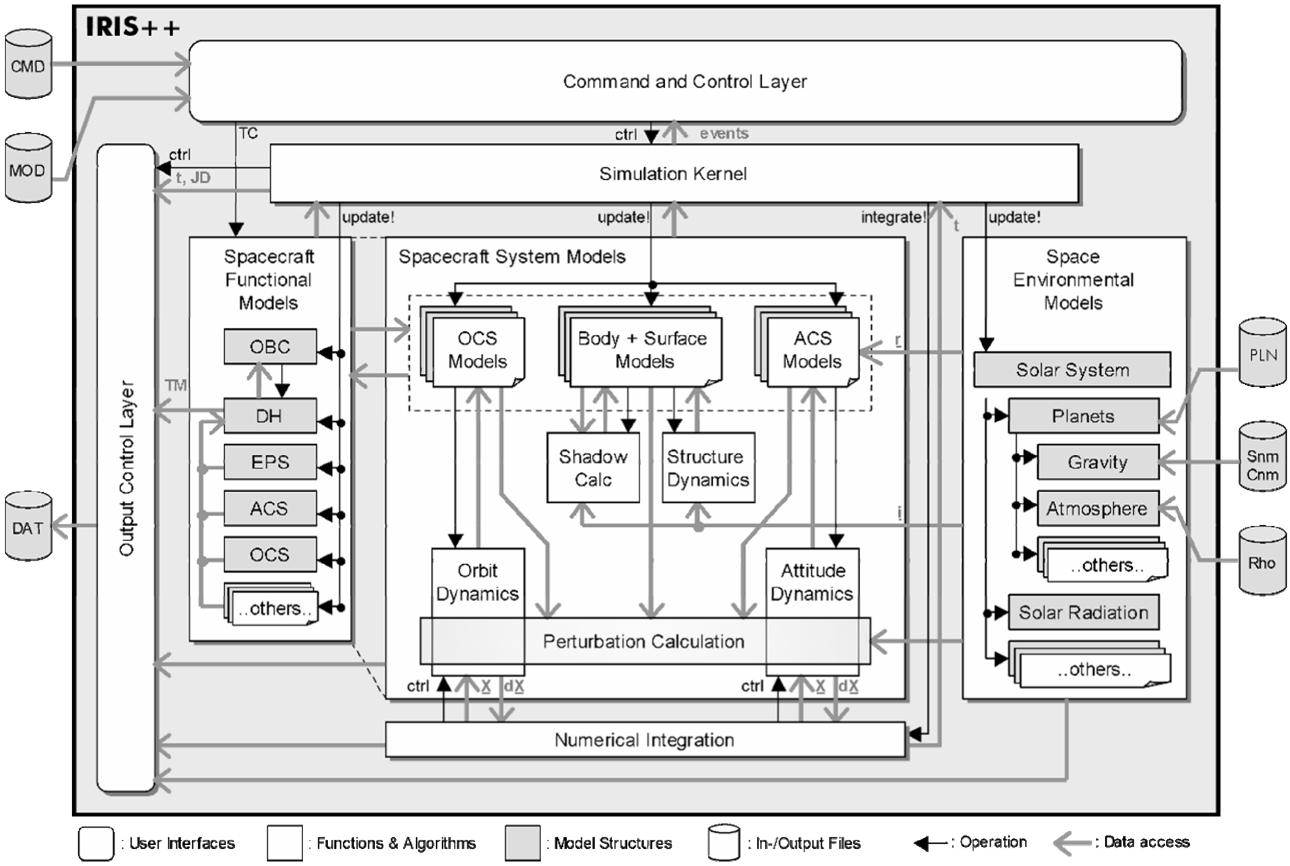


Figure 4.2: IRIS++ software architecture

In contrast to this, an *object structure* is a randomly linked structure of objects. By defining associated object variables (i.e. pointer to other objects) the programmer makes these links feasible between the objects (the so-called *relations*). The linkage itself will be made at run-time. An object structure with hierarchical character is often called object hierarchy or object tree.

To summarise, objects, defined by classes, are encapsulations of data and methods. Objects can be conceptualised in a manner similar to real world objects, or things. E.g. a car is an object that has data associated with it, such as the number of wheels and doors, colour, and size. A car also has operations that can be performed, like start, stop, accelerate, brake, turn left, turn right, and such operations like paint (for screen display). Data is usually hidden (not directly accessible) from the user or other programme parts but accessed using access methods. The internal representation can change without affecting how the user interfaces with the object and its encapsulated data within.

4.2.2 Programme Structure and Components

Figure 4.2 illustrates the functional structure of IRIS++, grouped into three main modules:

- Functional models: including the top-level representation of spacecraft systems/sub-systems:
    - o *On-board Computer* (OBC) model controlling autonomous and scheduling/triggering user-commanded operations performed by the spacecraft within the simulation run, e.g. Hohmann transfers for reboost manoeuvres.

  - *Data Handling* (DH) unit for storing system-level information and converting system data to output data provided as "telemetry" (TM)
  - *Electrical Power System* (EPS) model including e.g. information concerning the solar arrays, such as total solar array area, current power production level, etc.
  - *Attitude and Orbit Control System* (ACS, OCS) model with information on the number and types of AC/OC thrusters, propellant tanks, etc.
- System models: including all dynamics and component models that have to be simulated for spacecraft behaviour and internal and external state propagation:
  - Hierarchical physical body structure modelling of the mass properties
  - 3D surface model for physical-geometric representation
  - Shadowing algorithm supporting the environment-surface interactions
  - Structural dynamics routines performing solar tracking of structural components
  - Models of attitude and orbital control system (ACS, OCS) components, i.e. thrusters, tanks and momentum wheels
  - Attitude and orbit dynamics calculation routines
  - Attitude and orbit perturbation calculation routines (coupled)
- Environmental models: enabling and supporting the computation of the disturbances to the spacecraft orbit and attitude at the specific time and other specific calculations, depending on environmental information (e.g. orbit state representation, attitude control, tracking, shadowing). The planetary specifications are loaded from specification files at start-up.

A framework of support and control modules as well as in/output interfaces surrounds these models:

- The main control routine, called CCL in the scheme, reads and processes commands and configuration information from files to:
  - Build up the spacecraft element structure (bodies, subsystems),
  - Set up initial spacecraft state (orbit, attitude, subsystems)
  - Initialise all software components (e.g. date, environmental models; not shown),
  - Control simulation runs
  - "Tele-command" (TC) the spacecraft at specific events;
- The core simulation routine (kernel) includes the main processing loop with updating all models, controlling state propagation (integration) and data output.
- The output module is controlled by the simulation kernel and gathers all user-selected output variables and writes them a defined structure to an output data file for post-process analysis and visualisation. Data can be collected from all software components.
- The integration module manages the attitude and orbital state propagation operations, such as integration initialisation, method change if necessary, step-size control, etc.

Configurability and Extendibility

IRIS++ uses generic models that can be configured by parameters stored in external model specification files. Thus the user can change not only the simulated mission and spacecraft configuration, but the simulated space environment as well, without altering the source code and re-compiling the software.

Because the software is object-oriented and modularly designed, modifications and extensions to underlying models can conveniently be done within the source code. The process includes adding and implementing the new model class, defining a keyword for the specification file (CMD, MOD or PLN) and adding this keyword as designator in the specification file's interpreter routine. After connecting it with the new class and re-compiling the new model is available.

Even the simulation kernel can simply be replaced with other simulation or analysis schemes. For example, to obtain perturbations existing at specific locations in the solar system, a *perturbation scanner* programme class could quickly be integrated looping over the positions within the interesting region of space, all controlled by the command file. The new simulation class only has to take care of properly initialising the spacecraft state at each step because no *real* orbit and attitude dynamics and numerical integration are necessary. But it does not have to include initialisation of the time, date, output data and the environmental models because the entire functionality of the existing command file infrastructure can be used further.

4.2.3 Software Class Structure

To enable generic modelling and simulation of spacecraft systems for a wide-range of missions, a class structure of programme modules, routines and objects has been developed. The resulting programme structure is therefore modular and transparent and supports future extensions. This structure represents the implementation of the software architecture described in the previous section with most of the object classes ordered hierarchically:

- Parsing command, configuration and other model specification files (e.g. `CCommand`)
- Setup and control flight simulations (`CSimulate`)
- Structure of physical-geometrical bodies with mass properties (`CBody`)
- Structure of physical-geometrical surfaces (`CSModel`)
- Irradiation and shadowing computation (`CSurfaceCalcEngine`)
- Spacecraft functions and dynamics (`CSpacecraft`, `CAttitude`, `COrbit`)
- Orbit and attitude propagation (`CIntegrationProcedure`)
- The solar system, including the Sun, planets and moons (`CSolarSystem`)
- Celestial bodies ('planets'), including their rotation, gravitation and atmosphere (`CPlanet`)

Spacecraft Dynamics Model Classes

The UML diagram in Figure 4.3 provides an overview of major classes involved in orbit and attitude dynamics simulation and their interrelations. The individual classes are:

- `CCommand`: providing the command and control layer for the user (see Figure 4.2). The class is instantiated and initialised as global run-time object by the `main()`-routine. Processing the command and configuration file, it creates and configures all other programme components, thus controlling virtually every object in the system.
- `CSimulate`: includes the routines for setting up and performing the simulation runs, as configured by the user (via `CCommand`). It manages time, date, spacecraft state propagation, environmental model update and data outputs. Furthermore, it detects events that pause and stop the simulation, run and transfer them to command and control layer.

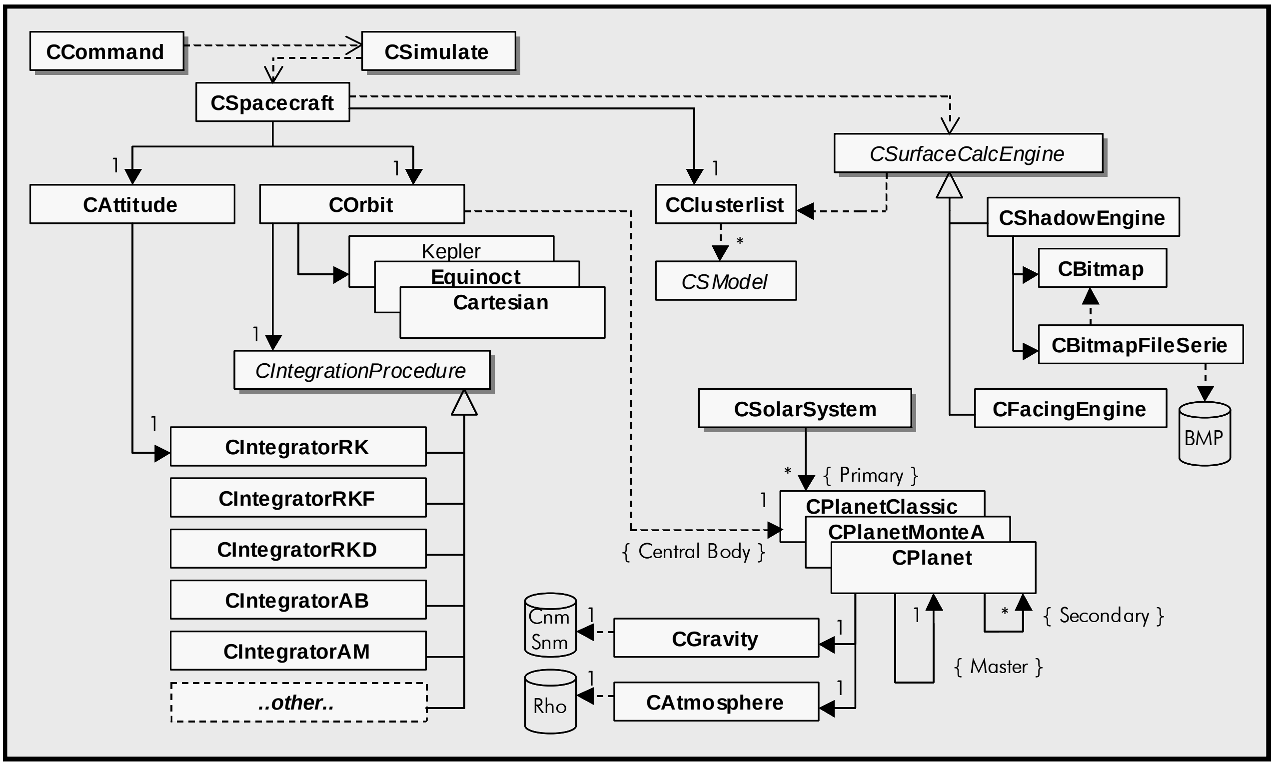


Figure 4.3: Spacecraft dynamics model class structure (excerpt)[1]

- `CSpacecraft`: modelling the space vehicle as a whole. Here the orbit and attitude dynamics are covered in an extra class `CAttitude` and `COrbit`. Concerning top-level update of perturbation influences, it manages the spacecraft surface model (`CClusterlist`) and initiates shadowing calculations. Other functions of the class are described below.
- `CAttitude`: includes all attitude dynamics and control methods. For state propagation it uses an RK4 numerical integrator.
- `COrbit`: includes all orbit dynamics and control methods. For state propagation it uses one of the implemented integrators as defined by the user. State representation can automatically be chosen or selected by the user. For perturbation calculation and dynamics evaluation, it accesses its current central body.
- `CIntegrationProcedure`: is the class through which the numerical integration methods are accessed.

[1] The *Unified Modelling Language* (UML) provides a standard representation scheme for class and object interaction diagrams. The following indicators help to read the relations in this diagram: Open arrowheads indicate an "is-a" relation between two classes. This means, the class at the arrow's root extends the class at the head (inheriting). Solid arrowheads indicate a "has-a" relation. This means an object of the class has another object of the class at the head stored or referenced to use it at run-time. A "1" at the root means, that "each-of" these objects has this relation. A number at the head marks the number of objects that the class at the base has. An asterix "*" means "every" object or an "unlimited" number of objects [UML].

- `CSurfaceCalcEngine`: is the abstract parent class of implemented function-classes for irradiation and shadowing calculations. Pure irradiation can be performed by using the `CFacingEngine` class and shadow calculations by `CShadowEngine.`
- `CSolarSystem`: controls the updates of the solar system environment including all planets and moons (`CPlanets`).
- `CPlanet`: models the motion of planets and moons and controls their incorporated environmental properties (`CGravity`, `CAtmosphere`) and possibly secondary planets (i.e. *moons*). The different sub-classes include different motion models of the planets and moons.

Spacecraft Physical-Geometrical Body Classes

The UML diagram in Figure 4.4 illustrates the hierarchy and interdependencies of the classes used for modelling a spacecraft system. Three basic classes exist: `CSpacecraft`, `CModule`, `CPrimitive`, all derived from the parent class `CBody` that defines a common spacecraft body and that is extended by its child classes (left). Two additional branches (on the right side) indicate the connections of the `CSpacecraft` class to a) the surface model for surface force calculations and b) the simulation of non-physical but functional components of the spacecraft.

Spacecraft Physical-Geometrical Surface Classes

To enable calculations of surface related parameters, such as surface area and the effective incidence area (necessary for `cSpacecraft`-level perturbation calculations), a numerical surface model must exist for each primitive class. Such surface models are defined with `cSModel` and its derived sub-classes (Figure 4.4, right). They use the basic geometrical object classes `cPoint`, `cPolygon` and more, to build up discretised polyeder models to approximate the surface of primitive bodies with variable levels of detail. The surface model and their discretisation method are described in section 3.2.4.

Spacecraft Functionality Classes

To include the functionalities of spacecraft sub-systems, another class of objects is introduced. The abstract *functionality class* `cFuncElements` groups all classes which describe properties and perform specific operations of different subsystems, such as a photovoltaic array of the *Electrical Power System* (EPS) or a propellant tank of the *Attitude and Orbit Control System* (AOCS). In doing this the objects are often mixed types with both, physical properties in addition to functional components.

Implemented functionality classes are:

- `CSolarpanel`, modelling panel-shaped photovoltaic arrays, thus inheriting the classes `CBox` and `CPhotovoltaic.` They calculate the current power level achieved by conversion of the direct solar irradiation to electricity on the active area.
- `CRadiatorpanel`, modelling panel-shaped thermal radiator, thus inheriting the classes `CBox` and `CThermalRadiator`

- `CTankSphere`, modelling a spherical tank volume for propellants or other fluids of different types. Tanks typically have a constant structure mass, a variable fluid mass and a mass flux depending on connected thruster activity.
- `CThrusterCone`, modelling a conical thrust nozzle. `CThruster` objects can be connected to tanks for propellant feed.

The *functionality part* of these multi-type objects is directly connected to the spacecraft object's corresponding subsystem part, which gathers their specific information. The method of *how* this information (e.g. produced power) is obtained is nested and encapsulated within the functionality classes. Thus, a future `CSolardynamic` object for instance, could replace a `CSolarpanel` object without altering the `CSpacecraft` functions related to the EPS.

To summarise, this generic spacecraft dynamics and functionality object class structure enables a high grade of flexibility for modelling and simulating different spacecraft during different missions.

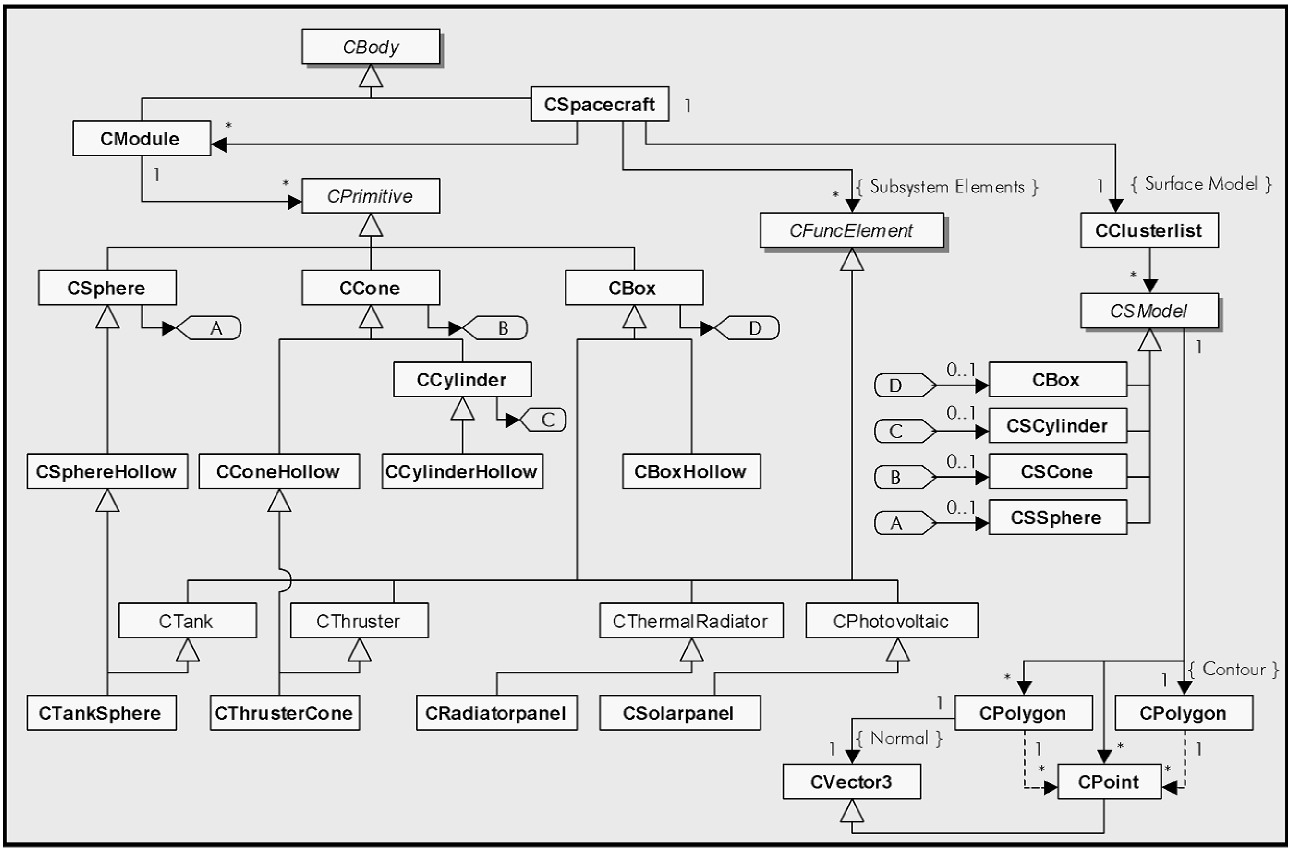


Figure 4.4: Spacecraft physical-geometrical model class structure (excerpt)

4.2.4 Verification

Setting up simulation software that *per definitio* shall represent reality in a specific and known order of magnitude, firstly one has to make sure that the selected models are appropriate and valid for the applied range of problems. Secondly, during software development, especially formulation of numerical methods, there is always the possibility of making errors, neglecting machine or language specific limitations and thus implementing improper routines. To control this threat and ensure correct transfer of the theory to machine executed code a thorough verification strategy was chosen.

This included verifications at different levels, at the *routine level*, which reveals faulty coding (e.g. specific equations), at the *programme module level*, to ensure proper description of a physical model for example, and *end-to-end* software verification tests, where a problem is selected and the result is analysed.

Following verifications were performed:

- All dynamic and kinematic equations, transformations and conversion routines were checked in reference to external, purely analytic solutions.
- The solar system motion model, including the orbits of the planets and the Moon were compared with numerical data given by ephemerides tables in literature for a given period of time ([Souders1970], [Standish2004]) and compared with accepted astronomical programs such as *Redshift 4* [Redshift] and *Guide 8.0* at certain well defined constellations (e.g. oppositions, vernal equinox and ecliptic plane passages).
- The implemented numerical methods were checked using specific problems that can also be solved analytically (e.g. two-body orbits). Because this does not ensure the method's integrity when perturbations are taken into account, two other tests were applied:
- Firstly, the methods were compared to each other, which gives a general impression whether a specific solver does drastically deviate with the others.
- Secondly, forward and backward integration were performed. This is a simple but quite powerful method because if the initial state is reached at the end with a specific accuracy, it is known, even without an external reference, that the solver itself works fine [Hechler2003].
- Finally the effective influences of various perturbations must be checked. This can only be done by adjusting a simulation to a situation well documented in literature or comparing simulation results with other tested and accepted software.

After successful performance of all previous tests the last is of course the most interesting one because it marks the overall validity. But this test is also the most challenging, not because of its technical difficulty but due to lack of information concerning the newly introduced perturbation effects. One typically does not have comparable software available (since this did not have to be created in the first place) and other sources of detailed simulation results are very limited. Indeed, the potentially best source in this case would be NASA and their trajectory experts at JPL because they did validate their tools using their experience from their actual spaceflight missions. Unfortunately, accurate simulation technology is classified as sensitive and as a restricted issue under US national export regulations, thus, no comparisons, not even discussions of results (beyond "looks reasonable" [Lo2003]) were possible!

Nevertheless, limited verification possibilities exist and were performed successfully: Simulation results of long-term effects could be compared with widely known "rule-of-thumb" deviations, especially the luni-solar disturbances of GEO satellites and the node drift of LEO satellites (including Sun-synchronous orbits). Secondly, gravitational dynamic Earth-Moon interaction results in libration points, which could be verified by setting up a simulation corresponding to the analytically available restricted three-body problem (i.e. Earth, Moon and the spacecraft with negligible mass).

4.3 Spacecraft Simulation Model

This section introduces the simulation model implemented in IRIS++. Time frames and conversions are discussed followed by an introduction of important coordinate systems and reference frames. Then the basics for attitude dynamics and orbital mechanics simulation are presented.

4.3.1 Time, Date and Coordinate Systems

When dealing with objects moving with high velocity through space, even small time differences can account for large differences in position. Thus, it is essential to know and use the right time frame and, therefore, it is necessary to define and relate relevant time frames that are used in astrodynamics and geodesy.

To define a specific point in time, one usually states "12:53 on April 17, 2004" for example. The individual wristwatch may deviate but nevertheless its time refers to the *Gregorian calendar* and the *Coordinated Universal Time* (UTC) transferred to the local time zone of the observer's longitude. The base unit "second" of the *Système International* (SI) is defined by atomic clocks, providing a uniform time base of high long-term stability through the hyperfine photon radiation of Cesium-133 atoms. These clocks provide us with the continuous *International Atomic Time* (TAI). Regular insertions of integer seconds to this TAI scale (i.e. *coordination*) yield to the UTC frame. In this connection UTC closely follows mean solar time, which is expressed by the *Universal Time UT* (i.e. UT1, the successor of GMT). UT is *derived* from the rotation of Earth expressed by the *Greenwich Mean Sidereal Time* (GMST), the Greenwich hour angle (GHA) of the vernal equinox. Although UTC is based on TAI, it is still a non-uniform time scale due to various influences included in UT, i.e. Earth's rotation, its motion around the Sun and relativistic effects. An atomic clock on Earth gives us the *Terrestrial Time* (TT), which is independent from Earth's rotation, but nevertheless it deviates from a sister clock sitting on or orbiting another planet or travelling though space. These differences are relevant for description of lunar and planetary ephemerides as well as solar system events within a general relativistic context. Therefore, an independent time scale is necessary by means of further time definitions, such as ephemeris, geocentric and barycentric dynamical time frames [AA2001]. This is why planetary motion models (such as the solar system model used here, section 4.4.6) typically use the *Barycentric Dynamic Time* (TDB) to obtain the position of the planets in respect to the solar system barycentre.

While IRIS++ users enter the time in UTC, the following conversion to TDB is necessary. For practical applications of the current scope the relativistic effects can be neglected [Montenbruck2000]:

(6) $$TDB = TT + \delta_r = TAI + 32.184s + \delta_r = UTC + \Delta_{leap} + 32.184s + \delta_r$$

with leap seconds $\Delta_{leap} = 32s$

and relativistic deviation $\delta_r \leq 0.002s \approx 0.0$

The given value for the leap seconds has been valid since January 1, 1999. Changes may occur at the end of June and/or the end of December and will be announced e.g. in "Bulletin C" of the International Earth Rotation Service (IERS) in Paris [IERS].

Julian Date

An instant in time (as defined by the sample date given above) is also called *epoch*, when the orbital state of a planet or spacecraft shall be specified. An alternate, more convenient calendar with which to do this is the Julian Date:

The *Julian Date* (JD) or its alternatively used derivate *Modified Julian Date* (MJD) is the major unit in astrodynamics to identify a specific date (epoch) or to measure time differences. JD provides a continuous scale and is defined as the number of days and day fractions which have passed since 12:00 UT on January 1, 4713 B.C.[2]. Because of the large numbers of Julian days, and since it is more convenient to start counting at midnight, the MJD was introduced:

(7) $$MJD = JD - 2\,400\,000.5$$

The equations for the conversion of Gregorian Date to a Julian Date and vice-versa are documented in publications, e.g. [Montenbruck2000].

Furthermore, the calculation of positions of celestial bodies often uses the *Julian Ephemeris Date* (JDE), counted in *Julian centuries* T since a specific epoch, today typically the *standard epoch J2000*, for 12:00 UT on January 1, 2000. This value is given by:

(8) $$T_{J2000} = \frac{JDE - JDE_{J2000}}{36\,525} = \frac{JDE - 2\,451\,545.0}{36\,525}$$

The different symbol indicates a difference of JDE to JD: While JD counts in terrestrial UTC-seconds, JDE refers to the TDB frame instead.

Coordinate Systems

Table 4.1 presents the definitions of the attitude and orbital coordinate systems used in IRIS++. The frames are all orthogonal, right-handed with the x-axis aligned to the stated principle direction and the x- and y-axis spanning the stated fundamental plane. The z-axis is always derived by the other axis and completes the right-hand system. The definitions always refer to the standard epoch J2000. Thus, the coordinate system's references are this epoch's vernal equinox and mean ecliptic or mean equator, respectively.

The reference frames of the first group are mainly used for spacecraft system related calculations, such as the mass properties and tracking operations. The spacecraft-bound origins identify the coordinate system origins that are used while modelling the space vehicle and its elements. In contrast to this, the centre-of-mass (CoM) could also be calculated during simulation. SPCS uses this CoM as origin and defines the principle axis frame of the inertia tensor of the actual spacecraft.

Orbit Coordinate Systems

The second group addresses orbit specific reference frames. The ORCS is defined according to the ANSI/AIAA convention [AIAA] and can be directly derived from the current orbital state of the spacecraft and is especially relevant for aerodynamic drag or rather non-spherical gravitational

[2] This epoch is determined from the combination of multiple calendars, which all shared this common year.

perturbation acceleration calculations. The LOCS is directly derived by rotation from the ORCS and is necessary for orbit integration using equinoctial elements.

Table 4.1: IRIS++ coordinate systems (note: with planet every celestial body can be associated with, be it the Earth, another actual planet, the Moon, or the even the Sun)

Coordinate System	Index	Origin	Principle Direction (X)	Fundamental Plane (XY)	Derived Direction (Z)
General/Spacecraft related					
Spacecraft Reference CS	SRCS	Spacecraft	(calculated)	(calculated)	(calculated)
Spacecraft Principle CS	SPCS	Spacecraft CoM	I_{11} Inertia axis (calculated)	I_{11}, I_{22} plane (calculated)	I_{33} Inertia axis (calculated)
Module-fixed CS	MCS	Spacecraft-Module object	(calculated)	(calculated)	(calculated)
Body-fixed CS	BCS	Spacecraft-Body object	(calculated)	(calculated)	(calculated)
Orbit specific					
Orbit Reference CS	ORCS	Spacecraft	IOP, transversal (approx. velocity)	Local Horizontal	Local Vertical (Nadir)
Local Orbital CS	LOCS	Spacecraft	IOP, radial (Zenith)	Orbital Plane	POP (North)
Planetocentric Equatorial	PCEQ	Central body	Vernal equinox	Equator	North
Planetocentric Planet-Fixed	PCPF	Central body	Main meridian (rotating)	Equator	North
Planetocentric Orbit-Fixed	PCOF	Secondary	Primary centre (rotating)	Secondary's orbital plane	Towards South (calculated)
Planetocentric Ecliptical	PCEC	Central body	Vernal equinox	Ecliptic	Ecliptic north
Heliocentric Ecliptical	HCEC	Sun	Vernal equinox	Ecliptic	Ecliptic north
Barycentric Ecliptical	BCEC	Barycentre of system	Vernal equinox	Ecliptic	Ecliptic north
Attitude specific					
Spacecraft Commanded CS	SCCS	Spacecraft	(user-defined with respect to flight mode)	(user-defined with respect to flight mode)	(user-defined with respect to flight mode)
Earth Oriented 1 flight mode	EO1	Spacecraft	IOP, transversal (approx. velocity)	Local Horizontal	Local Vertical (Nadir)
Inertial Oriented 1flight mode	IN1	Spacecraft	Vernal equinox	Central body's Equator	North
Inertial Oriented 2 flight mode	IN2	Spacecraft	Sun (ASL)	Central body's Equator	(free)

Next stated is PCEQ, the standard coordinate system using the planets equatorial plane as reference and in which the Keplerian orbital elements are defined and where orbit integration with cartesian elements takes place. In case Earth is taken as central body it corresponds to the documented common inertial geocentric coordinate system (GCI). Care has to be taken of the fact that this system is actually only "inertial" when no third-body perturbations are accounted for (see section 4.4.5). In the case of the Sun, the PCEQ system is actually an ecliptical system per convention (thus equal to HCEC), because it is common to describe interplanetary orbits with respect to the ecliptic instead of the Sun's equatorial plane.

Then there are two other planet-centred but non-inertial frames: The first, PCPF rotates with the referenced planet, therefore allowing identification of relative positions to surface locations and ground track calculation. The geo-potential field equations as well as the aerodynamic models of atmospheres are based on them (see section 4.4). The second rotating frame is PCOF, which is introduced to enable convenient analysis of spacecraft motions relative to the rotation of a two-body system, e.g. the Earth-Moon system.

The final planet-centred frame is PCEC, which serves as an interface between PCEQ and HCEC during the calculation of third-body perturbations, which are retrieved in HCEC. PCEC is also typically used for describing the motion of *secondary bodies* (i.e. moons of planets) in a planetary system (i.e. Earth-Moon system). In addition to HCEC, there must also be a barycentric frame (BCEC) for describing the orbital motion of planets, or rather the barycentre of planetary systems.

Attitude Reference Frames

There are several reference frames used in IRIS++ to describe orientations of objects. The above-mentioned SRCS describes the spacecraft's attitude state with respect to the ORCS and is by default identical to it in a circular orbit without attitude perturbations. The SCCS now determines the desired orientation of the spacecraft with respect to the user-selected *flight mode*. Latter can be chosen to be:

- *Earth Oriented 1* (EO1) equal to the ORCS, defining the local-horizontal-local-vertical (LHLV) attitude,
- *Inertial Orientation 1* (IN1) equal to the PCEC reference frame, and
- *Inertial Orientation 2* (IN2) aligned to the Sun (x: aligned to Sun line (ASL), y: in ecliptical plane, z: towards ecliptical north) and is strictly only semi-inertial due to the Earth's motion around the Sun

With attitude control activated during simulation, the used control algorithm will apply CMG or thrusters torques to keep the spacecraft's SRCS parallel to the SPCS.

4.3.2 Orbit Dynamics

The survey of orbital dynamics provides an understanding of the principles behind the spacecraft simulation framework. Beginning with an in-depth coverage of spatial state representations, it summarises the equations of motion as well as the perturbations to the ideal orbits. Finally, special issues accounting for orbit propagation are discussed.

Orbital State Representations

The orbital state is the description of the current three-dimensional trajectory of a spacecraft (or planet) relative to a defined reference frame/coordinate system. A representation of the orbital state is also necessary to formulate the dynamics of the orbital motion. The following two sections define common types of state representations and reference frames.

To fully specify the orbital state of a spacecraft at a given time at least six independent elements (*ephemeris*) must be provided. The widely used types, also implemented in IRIS++, are the *Keplerian Orbital Elements*, the *Cartesian Position and Velocity* and the *Equinoctial Elements*:

The most popular and figurative elements are the Keplerian Orbital Elements specifying the orbital shape and the orbital plane's orientation in space. The orbit shape is typically an ellipse, but depending on the spacecraft energy, it can also become parabolic and even hyperbolic in shape. Assuming an ellipse, Figure 4.5 illustrates the orbit elements: The *semi-major-axis* (a) and the *eccentricity* (e) define the shape, while the angles called *right-ascension-of-the-ascending-node* (RAAN) or (Ω), the *argument-of-perigee* (ω) and the *inclination* (i) define the orientation with respect to the inertial Earth-centred frame with the x-axis directing to the vernal equinox and the z-axis perpendicular to the Earth's equatorial plane (see PCEQ). The sixth remaining element is the *true-anomaly* (θ), an angle giving the position of the spacecraft on its orbit with respect to the perigee.

There are many individual parameters derived from the Kepler elements, also to be used alternatively for orbit specification, such as the orbital revolution period (instead of a) and the *radius-of-perigee* (r_p) or *–apogee* (r_a) stating the nearest or farthest point of the orbit, respectively (instead of e). BROWN [Brown1998] offers a comprehensive compilation of derived parameters and important equations that link the parameters to each other. Orbits around other celestial bodies are defined analogously by using their equatorial plane, except heliocentric orbits, where the ecliptic plane is chosen. Because Kepler orbits describe ideal, un-perturbed two-body orbital mechanics, the Kepler elements are only valid for a given epoch. To describe a realistic orbit over time, a series of Kepler elements set, the *osculating Kepler elements,* are necessary. Plotting the elements' evolution over time provides deeper insight into the orbital dynamics of a spacecraft. But Kepler elements incorporate singularities for the case of circular (e=0) and/or equatorial (i=0°) orbits. In the first case no perigee is defined and thus the true--anomaly reference vanishes. To avoid this problem of indicating the spacecraft position, alternatively the *argument-of-latitude* u is used which is the sum of the argument-of-perigee and the true-anomaly:

(9) $$u = \omega + \theta$$

In the other case of an equatorial orbit, the *line-of-nodes* connecting the ascending and descending node does not exist, thus the RAAN is undefined.

Since Kepler elements are especially descriptive they are used for data output and visualisation purposes only. Because of their singularities, two other state representations are preferable and used for numerical treatment within IRIS++. Therefore, a universal orbital state representation is given by the Cartesian Position and Velocity vectors. Here the radius vector and its first derivate $\dot{\bar{r}} = \bar{v}$ provide the six necessary elements:

(10) $$\bar{r} = x \cdot \bar{e}_X + y \cdot \bar{e}_Y + z \cdot \bar{e}_Z = \begin{bmatrix} x \\ y \\ z \end{bmatrix}$$

(11) $$\dot{\bar{r}} = \begin{bmatrix} \dot{x} \\ \dot{y} \\ \dot{z} \end{bmatrix} = \bar{v} = v_X \cdot \bar{e}_X + v_Y \cdot \bar{e}_Y + v_Z \cdot \bar{e}_Z = \begin{bmatrix} v_X \\ v_Y \\ v_Z \end{bmatrix}$$

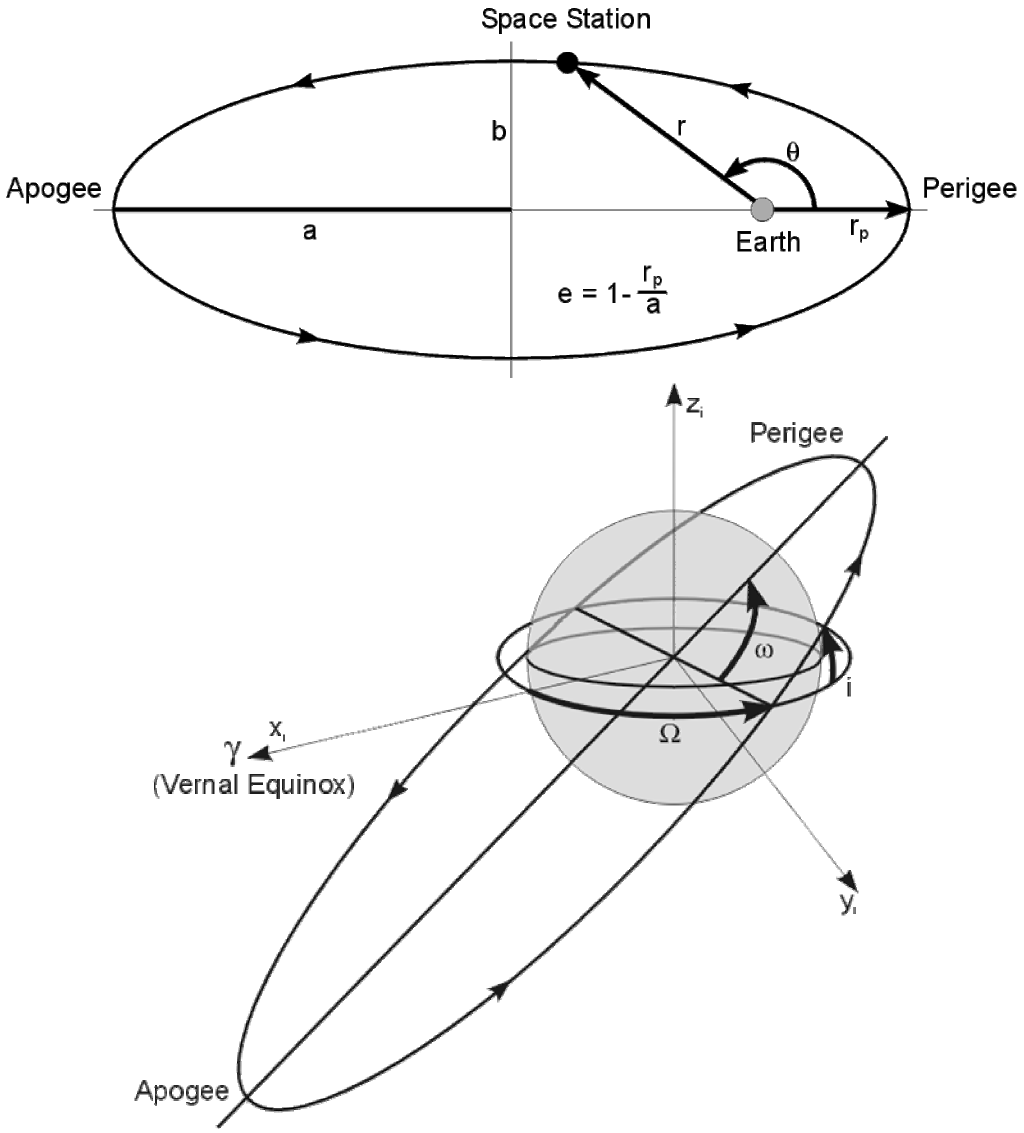


Figure 4.5: Kepler orbital elements (in the planetocentric equatorial frame) [Hinüber2002]

The vector components can be given and used in any of the major reference frames, defined by cartesian coordinate system axes expressed by the unit vectors $\bar{e}_x, \bar{e}_y$ and $\bar{e}_z$. Evaluating the state vectors with respect to the equatorial coordinate system PCEQ of the dominating celestial body and taking this body as reference for a two-body problem, the instantaneous orbital elements can be computed by a conversion scheme (e.g. [Montenbruck2000]).

Another set of parameters for state representation is BROUCKE's Equinoctial Elements supporting numerical stable integration of circular and low-eccentricity orbits. They are closely based on the classical Keplerian orbital elements, but are lacking in their singularities at equatorial and circular orbits. Instead their singularities lie in harmless conditions of rectilinear orbits (zero angular momentum or parallel position and velocity vectors) and reverse equatorial orbits (inclinations $i = 180°$) [Battin78]. These six equinoctial elements can be derived to and from the classical elements (including the mean anomaly M) by the following equations:

(12) $$P_1 = e\sin(\Omega+\omega)$$

(13) $$P_2 = e\cos(\Omega+\omega)$$

(14) $$Q_1 = \tan\frac{i}{2}\cdot\sin\Omega$$

(15) $$Q_2 = \tan\frac{i}{2}\cdot\cos\Omega$$

(16) $$l = \Omega+\omega+M$$

(17) $$a = a$$

(18) $$e = \sqrt{{P_1}^2+{P_2}^2}$$

(19) $$\Omega = \arctan\frac{Q_1}{Q_2}$$

(20) $$\omega = \arctan\frac{P_1}{P_2}-\Omega$$

(21) $$i = 2\arctan\sqrt{{Q_1}^2+{Q_2}^2}$$

(22) $$M = l-\Omega-\omega$$

(23) $$a = a$$

But equinoctial elements cannot be used universally for all orbits. In fact they are not applicable to very high eccentricity, parabolic or hyperbolic orbits. Therefore, they cannot be chosen as the single numerical state representation for simulation, but use cartesian elements for the other cases too.

Equation of Motion

For predicting the time evolution of the spacecraft's orbital state relative to the dominating celestial body, the dynamic equation of motion of the spacecraft body must be formulated, by accounting for all relevant (perturbation) accelerations, and integrating them numerically over time. This means the orbital state in one of the mentioned representations is taken to evaluate the variation of the state parameters due to the accelerations.

For cartesian representation, the inverse-square law of orbital motion [Bate71] is taken by adding the perturbation terms (formulated for spacecraft with $m \vee M$):

(24) $$\ddot{\bar{r}} = -\frac{GM}{r^3}\cdot\bar{r}+\bar{a}_{pert}(\bar{r},\bar{v})$$

to the universal gravitational constant G, central body mass M, position vector $\bar{r}$ forming the point-mass attraction of the central body, and the total perturbation acceleration vector $\bar{a}_{pert}$ of various sources, which can of course depend on the spacecraft's state as well (see next section). State and variation vectors are then given by:

(25) $$\bar{y} = \begin{bmatrix} x \\ y \\ z \\ v_x \\ v_y \\ v_z \end{bmatrix}$$

(26) $$\dot{\bar{y}} = \begin{bmatrix} v_x \\ v_y \\ v_z \\ a_x - \frac{GM}{r^2}\cdot\bar{e}_x \\ a_y - \frac{GM}{r^2}\cdot\bar{e}_y \\ a_z - \frac{GM}{r^2}\cdot\bar{e}_z \end{bmatrix}$$

and form the state equation $\dot{\bar{y}} = f(t,\bar{y})$, formulated in the PCEQ frame.

Analogue equations exist for the other state representation linking the perturbation term in cartesian coordinates. The variations of classical elements (given by *Gauss' Equations*) and equinoctial elements are provided by BATTIN [Battin1987]. Here the perturbation vector must be transformed into the LOCS system.

Orbit Perturbations

The term $\bar{a}_{pert}$ used above in the dynamic equations of orbital motion includes major perturbation and external control forces applied when operating a spacecraft:

(27) $$\bar{a}_{pert} = \bar{a}_{non-spherical}(\bar{r}) + \frac{1}{m}\bar{F}_{drag}(\bar{r},\bar{v}) + \frac{1}{m}\bar{F}_{srp}(\bar{r}) + \bar{a}_{3rd-body}(\bar{r}) + \frac{1}{m}\bar{F}_{thrust}$$

with spacecraft mass m and force $\bar{F}_i$ to a corresponding perturbation i. Sources of these disturbances are:

- Environmental influences, such as the:
  - Non-spherical shape of the gravitational potential field of the celestial body orbited
  - Aerodynamical drag
  - Solar radiation pressure (SRP) torque
  - Third-body gravitation
- Additional external accelerations may be caused by:
  - Orbit control thrusters
  - Electromagnetic tether devices

Except for the tether devices, all perturbations mentioned are included in the IRIS++ simulator. Depending on the mission scenario some sources become more or less important. An overview of the perturbation accelerations of different sources is presented in chapter 4.4.

Orbit Integration

The equations of motion and the variational equation lead to a six-dimensional state equation $\bar{y} = f(t, \bar{y})$. This system of six ordinary differential equations (ODE) can be directly integrated with any appropriated numerical integration method.

Worth emphasizing here is that the numerical properties of (near-) circular and non-circular (especially high-eccentric) orbits deviate greatly. It is well known that during the short period of pericentre passage of high-eccentric orbits, the orbital motion is very dynamic and for the rest of the orbit rather slow and smooth. Therefore, the accuracy of orbit integration will basically depend on the integrators' accuracy during these short phases, where powerful accelerations and high spacecraft velocity are paired with a fast rotation of the velocity and acceleration vectors. Integration methods (single vs. multi-step) can differ greatly in their ability to cover this demanding behaviour, but in general a high numerical resolution and therefore a small integration step size is necessary during the pericentre flight phase. Because simulating the whole orbital motion under this pericentre-dictated condition is very time consuming and therefore inefficient, integration methods with step size control are the first choice for orbit integration. IRIS++ o implemented several integration methods (i.e. Runge-Kutta and derivates by Fehlberg and Dormind-Prince; Adams-Bashforth and -Moulton) of different orders.

4.3.3 Attitude Dynamics

An attitude is used to define the orientation of a body-bound frame to a reference frame, such as a spacecraft body-fixed frame with respect to an orbit reference or Earth-fixed frame. Attitude dynamics are the time-variation of this spacecraft attitude with respect to another reference frame due to external torques and are described by the dynamic equations of motion. To describe an attitude or attitude kinematics and dynamics, the set of used reference frames and the methods for representing the orientation of these frames with respect to another are defined in this section. The subjects of kinematics and dynamics of these frame rotations and of the computation of attitude dynamics are discussed.

Attitude State Representations

There exist various attitude representations such as: *Euler Axis and Angle*, *Three Rotation Angles*, *Direction Cosine Matrix* (DCM), *Quaternions* and others. A comprehensive survey is available from SHUSTER [Shuster1993] including a convenient set of conversion equations. IRIS++ uses mainly Rotation Angles and DCM for attitude description and Quaternions for kinematics, routines which where implement by HINÜBER [Hinüber2002]. Important features of these types of attitude representation are as follows:

- A Direction Cosine Matrix (DCM) is a simple way to describe and represent transformations with a 3×3 matrix of values, a rotation matrix. Applied to a given 3×1 vector it transforms a vector from one coordinate frame a to another frame b according to equations (28) and (29):

(28) $$\bar{v}^{b} = \bar{\bar{R}}^{ba} \cdot \bar{v}^{a}$$ with

(29) $$\bar{\bar{R}}^{ba} = \begin{bmatrix} \cos\Phi_{x_b x_a} & \cos\Phi_{x_b y_a} & \cos\Phi_{x_b z_a} \\ \cos\Phi_{y_b x_a} & \cos\Phi_{y_b y_a} & \cos\Phi_{y_b z_a} \\ \cos\Phi_{z_b x_a} & \cos\Phi_{z_b y_a} & \cos\Phi_{z_b z_a} \end{bmatrix}$$

  The values $\cos\Phi_{I_b J_a}$ are the cosine of the angles between axis I in frame b and axis J in frame a. Because such rotations can be combined by multiplication of their corresponding matrices, one can inversely regain the angles of a known sequence of principle rotations.

- Three rotation angles define a sequence of rotations around the principle axes x (Φ), y (Θ) and z (ξ). If the first and third rotation are around the same axis (e.g. 3-1-3 meaning: firstly around z, then around x, finally around z again), the angles are called *Euler Angles;* and if all axes are used, *Cardan Angles*. The latter often correspond with the angles *roll*, *pitch* and *yaw* used in flight dynamics. Note that the order of the axes used is important. With equation (30) the DCM is obtained from Cardan angles of a 1-2-3 rotation sequence:

(30) $$\overline{\overline{R}}^{ba} = \begin{bmatrix} c3\cdot c2 & s3\cdot c1 + c3\cdot s2\cdot s1 & s3\cdot s1 - c3\cdot s2\cdot c1 \\ -s3\cdot c2 & c3\cdot c1 - s3\cdot s2\cdot s1 & c3\cdot s1 + s3\cdot s2\cdot c1 \\ s2 & -c2\cdot s1 & c2\cdot c1 \end{bmatrix}$$

with substitutions $\begin{matrix} s1 = \sin(\Phi) \\ s2 = \sin(\Theta) \\ s3 = \sin(\xi) \end{matrix}$ and $\begin{matrix} c1 = \cos(\Phi) \\ c2 = \cos(\Theta) \\ c3 = \cos(\xi) \end{matrix}$

Another important aspect to consider is that because several rotation sequences may achieve the same result, the transformation from one reference frame to another is not unique. Furthermore, singularities can occur in the sequence of three rotations. Thus rotation angles are often impracticable for numerical treatment within simulations. Nevertheless, they are useful for output and visualisation purposes and therefore available in IRIS++.

- Quaternions (or Euler symmetric parameters) are specific attitude representations used for numerical computations. They reduce the computation time considerably, do not possess any singularities of the rotation angles and need no evaluation of trigonometric functions during coordinate transformations. Thus they become a favoured scheme for attitude integration and on-board navigation systems [Wertz78]. The four quaternions build a four-element structure $\overline{\hat{q}}$ with three elements of a vector $\overline{q}$ and an additional scalar q_4 [Hinüber2002]:

(31) $$\overline{\hat{q}} = \begin{bmatrix} q_1 \\ q_2 \\ q_3 \\ q_4 \end{bmatrix} \quad \text{with} \quad \overline{\hat{q}} = \begin{bmatrix} \overline{q} \\ q_4 \end{bmatrix} \quad \text{and} \quad \overline{q} = \begin{bmatrix} q_1 \\ q_2 \\ q_3 \end{bmatrix}$$

satisfying the following constraint equation:

(32) $$q_1^2 + q_2^2 + q_3^2 + q_4^2 = 1$$

The four parameters can be obtained from a given DCM $\overline{\overline{R}}$ by:

(33) $$q_1 = \frac{1}{4\cdot q_4}\left(\overline{\overline{R}}_{23} - \overline{\overline{R}}_{32}\right)$$

(34) $$q_2 = \frac{1}{4\cdot q_4}\left(\overline{\overline{R}}_{31} - \overline{\overline{R}}_{13}\right)$$

(35) $$q_3 = \frac{1}{4\cdot q_4}\left(\overline{\overline{R}}_{12} - \overline{\overline{R}}_{21}\right)$$

(36) $$q_4 = \frac{1}{2}\sqrt{1 + \overline{\overline{R}}_{11} + \overline{\overline{R}}_{22} + \overline{\overline{R}}_{33}}$$

Vice-versa the DCM can be calculated by:

(37) $$\overline{\overline{R}} = \begin{bmatrix} q_1^2 - q_2^2 - q_3^2 + q_4^2 & 2(q_1\cdot q_2 + q_3\cdot q_4) & 2(q_1\cdot q_3 - q_2\cdot q_4) \\ 2(q_1\cdot q_2 - q_3\cdot q_4) & -q_1^2 + q_2^2 - q_3^2 + q_4^2 & 2(q_2\cdot q_3 + q_1\cdot q_4) \\ 2(q_1\cdot q_3 + q_2\cdot q_4) & 2(q_2\cdot q_3 - q_1\cdot q_4) & -q_1^2 - q_2^2 + q_3^2 + q_4^2 \end{bmatrix}$$

Equations of Motion

For predicting the time evolution of the spacecraft orientation (attitude propagation), IRIS++ uses dynamic modelling of the attitude dynamics by formulating the *dynamic* and *kinematic equations of motion* and integrating them numerically over time.

The dynamic (or kinetic) equations of motion, given by Euler equations, characterise the absolute angular velocity vector in an environment subject to external disturbances. In this connection, the spacecraft must be seen as a rigid-body or non-rigid body, depending on whether it incorporates angular momentum storage devices (such as reaction or momentum wheels) for attitude control. In accordance with WERTZ [Wertz78] the Euler equations are formulated by the following equations, for

a) rigid spacecraft:

(38) $$\frac{d\overline{L}}{dt} = \overline{M}_{ex} - \overline{\omega} \times \overline{L}$$

(39) $$\overline{L} = \overline{\overline{I}} \cdot \overline{\omega}$$

(40) $$\overline{\overline{I}} \frac{d\overline{\omega}}{dt} = \overline{M}_{ex} - \overline{\omega} \times (\overline{\overline{I}} \cdot \overline{\omega})$$

b) non-rigid spacecraft:

(41) $$\frac{d\overline{L}}{dt} = \overline{M}_{ex} - \overline{\omega} \times \overline{L} - \overline{\omega} \times \overline{h} + \overline{M}_{in}$$

(42) $$\overline{L} = \overline{\overline{I}} \cdot \overline{\omega} + \overline{h}$$

(43) $$\overline{\overline{I}} \frac{d\overline{\omega}}{dt} = \overline{M}_{ex} - \overline{\omega} \times (\overline{\overline{I}} \cdot \overline{\omega} + \overline{h}) - \overline{\omega} \times \overline{h}$$

(44) $$\frac{d\overline{h}}{dt} = \overline{M}_{in}$$

with the angular momentum $\overline{L}$, defined by the moment of inertia tensor $\overline{\overline{I}}$ and the angular velocity $\overline{\omega}$. The time derivative is taken and the vectors are resolved in a body-fixed coordinate system (i.e. SRCS). The term $\overline{M}_{ex}$ includes all external torques due to perturbations, thruster operations and $\overline{M}_{in}$ the control torques applied to a momentum or reaction wheel to change its angular momentum $\overline{h}$. The latter transfers angular momentum from the rigid part of a spacecraft to a wheel or vice-versa (equation (42)). This means, with a torque $\overline{M}_{in}$ alone the total angular momentum of the spacecraft stays constant. Equations (40) and (43) are obtained by taking equations (38) and (41), substituting $\overline{L}$ with equations (39) and (42), respectively and using the additional *control momentum equation* (44) addressing the additional degree of freedom.

To fully describe spacecraft attitude dynamics, the kinematic equations of motion are necessary. These describe the current change of orientation of the spacecraft body, thus the current rotational motion. The formulation of this *variational equation* in quaternion representation is given by WERTZ [Wertz78]:

(45) $$\frac{d}{dt}\overline{q} = \tfrac{1}{2} \cdot \Omega \cdot \overline{q}$$ with the skew-symmetric matrix $$\Omega = \begin{bmatrix} 0 & \omega_3 & -\omega_2 & \omega_1 \\ -\omega_3 & 0 & \omega_1 & \omega_2 \\ \omega_2 & -\omega_1 & 0 & \omega_3 \\ -\omega_1 & -\omega_2 & -\omega_3 & 0 \end{bmatrix}$$

Attitude Perturbations

The term $\overline{M}_{ex}$ used in the dynamic equations of motion above includes major perturbation and external control torques existing while a spacecraft is orbiting celestial bodies or during transfer in-between:

(46) $$\overline{M}_{ex} = \overline{M}_{gg} + \overline{M}_{drag} + \overline{M}_{srp} + \overline{M}_{thrust} \;\; (+ \overline{M}_{mag} + \;\; ... \;\;)$$

Sources of torques to be taken into account are:

- Environmental influences, such as the:
    - Gravity-gradient torque
    - Aerodynamical drag torque
    - Solar radiation pressure (SRP) torque
    - (Magnetic field interactions)
- External attitude stabilization and control torques caused by:
    - Attitude thrusters
    - (Magnetic-coil torquers; if applicable)

External forces $\overline{F}_i$ due to environmental influences with a point of attack $\bar{r}_i$ lying off-centre with respect to the CoM $\bar{r}_{CM}$ cause the torques:

(47) $$\overline{M}_i = \overline{F}_i \times (\bar{r}_i - \bar{r}_{CM}) \qquad \text{with} \qquad i = drag, srp \; (, gg)$$

Depending on the perturbation i, the point-of-attack $\bar{r}_i$ can be the position of CoP of the aerodynamic drag or solar radiation pressure. Although the resulting gravity force attacking at the *centre-of-gravity* (CoG) could be derived from the gravitation field and the spacecraft mass distribution, the gravity-gradient torque $\overline{M}_{gg}$ is calculated directly from the spacecraft inertia tensor $\bar{\bar{I}}$ and the orbital radius R [Hinüber2002]:

(48) $$\overline{M}_{gg} = \frac{3\mu}{R_{CM}^3}\left[\bar{e}_R \times \left(\bar{\bar{I}} \cdot \bar{e}_R\right)\right] \underset{\text{Earth Oriented}}{=} \frac{3\mu}{R_{CM}^3}\begin{bmatrix} -I_{yz} \\ I_{xz} \\ 0 \end{bmatrix}$$

with the Earth's gravitational parameter $\mu = 3.989 \cdot 10^{14} \frac{m^3}{s^2}$.

The external control torque due to thruster operations can be obtained similarly:

(49) $$\overline{M}_{thrust} = \sum_n \left(\overline{F_n} \times (\bar{r}_n - \bar{r}_{CM})\right)$$

Here the number of thrusters is n, which are located at $\bar{r}_n$ with the thrust vector $\overline{F_n}$. All measurements take place at the body-fixed frame with random origin, e.g. the geometrical centre of the spacecraft. The interaction of the spacecraft with the planet's magnetic field generates a magnetic moment $\overline{m}$. This includes disturbances due to permanent or induced magnetisms of spacecraft components and spacecraft-generated current loops [Wertz78]. The latter can also be produced intentionally with magnetic coils for attitude stabilization and control including desaturations of reaction wheels instead of using thrusters. The instantaneous magnetic torque due to the spacecraft's effective magnetic moment $\overline{m}$ [Am²] and the planet's local magnetic flux density $\overline{B}$ [Vs/m²] is given by:

$$\overline{M}_{mag} = \overline{m} \times \overline{B} \tag{50}$$

Except for $\overline{M}_{mag}$, which is only relevant for certain LEO missions, all mentioned perturbations are available in IRIS++. As a result the drag and SRP force as well as the associated CoP positions can be influenced by shadowing effects and, therefore have to be taken into account. The calculation of the perturbation forces $\overline{F}_i$ is done with environmental models implemented in IRIS++. Section 4.5 addresses how shadows are quantitatively taken into account within the flight simulation software for the computation of all surface forces and torques.

Attitude Integration

Written in Cartesian components, four kinematic equations from (45) and three dynamic equations of motion from (40) are obtained, totalling seven attitude state variables and functions, which are to be integrated for attitude simulations of a rigid spacecraft. Implemented in IRIS++ is a non-rigid spacecraft equipped with reaction wheels, therefore one integrates (45), and analogously (43) and three control momentum equations from (44) in addition, making ten functions. The written components equations and further attitude control equations used in IRIS++ have been documented in a previous thesis [Hinüber2002]. For propagation of these attitude state functions, IRIS++ uses a Runge-Kutta numerical integrator of fourth order (RK4).

A point to pay attention to while setting-up a spacecraft flight simulation integrator engine is the principle coupling between attitude and orbit integration due to each perturbation depending on the others' current state. To accurately compute the solar radiation pressure force and torque vectors for instance, the spacecraft's current position relative to the current positions of the Sun and possible eclipse conditions must be evaluated during attitude integration <u>and</u> during orbit integration. The current incidence area depending on the spacecraft attitude and tracking state must be evaluated, too. This is especially important if the spacecraft geometry and thus orientation is used for orbit control (e.g. solar sail applications). More related information is available from WOODBURN [Woodburn2001].

4.4 Space Environmental Models

Space is an environment of extremes. The spacecraft, especially its outer surface in near vacuum, is exposed to a range of environmental influences: from kilowatt solar irradiation in direct Sun light at Earth distance to the black coldness of 3 K in the shadow, from the erosive action of atomic oxygen in LEO to the permanent flux of charged particles, from the aerodynamic drag of the planetary atmosphere leading to orbital decay, to the solar radiation pressure possibly used for solar attitude control, and finally the constant threat of meteoroid and orbital debris impacts and exposure to the radiation during *solar flare* events.

Mission and system design deals primarily with the task of coping with these environmental effects, taking them into account during system element design (e.g. shielding against impacts and radiation), using them as acceptable tools or controlling them as perturbations. Identifying and quantifying the effects is accomplished by flight simulations.

This section addresses the environmental models implemented in IRIS++ and summarises the most relevant equations. In doing this it focuses on the effects necessary to calculate the perturbation influences affecting the attitude and orbit dynamics.

4.4.1 Environmental Effects and Perturbations

For mission design the most relevant environmental effects originate from the orbited planet itself and the Sun affecting different mission elements and subsystem design. The most relevant environmental influences, apart from gravitation, are described in this section. Other influences, such as the planetary magnetic field, the *solar wind*, radiation and relativistic effects are not considered.

The residual atmosphere of the orbited planet

In low Earth orbits to an altitude of approximately 500 km (Mars: 260 km), the aerodynamic incident flow of particles of the residual atmosphere and their interaction with the spacecraft surface impose major influences. This leads to the deceleration and thus altitude reduction and produces disturbing momentums. For maintaining the nominal range of altitude and attitude this aerodynamic drag force must be compensated, necessitating propellant needs of the attitude and orbit control system (AOCS). Furthermore, they affect the position of the *torque equilibrium attitude* (TEA), which determines – together with the drag-deceleration – the prevailing microgravity conditions (μg) on-board [Messerschmid2000]. Moreover with the aerodynamic incident flow spacecraft surface elements are exposed to chemically aggressive atomic oxygen.

Above an altitude of 1000 km (Mars: 400 km) all aerodynamic effects can be neglected.

Solar irradiation

The radiation of the Sun is in multiple ways highly relevant for the operation of spacecraft:

- Power supply due to light irradiation on photovoltaic arrays or solar dynamic collectors
- Thermal balance and light/thermal irradiation on heat radiators
- Solar radiation pressure

With about 1.4 kW/m^2 the solar irradiation is an important energy source in space and is crucial for the design of thermal radiators and solar panels. If irradiating conditions on radiators and solar panels do not correspond with specified values, this may yield significantly low performances. Thus, solar irradiation including possible shadowing must be considered during sizing of these systems and their accommodation during configuration design of space stations.

Another effect comes from the dynamic interaction of the irradiation with the surface. Somewhat similar to the aerodynamic effects, the sunlight leads to a non-conservative force and torque.

Furthermore, the Sun has a heating effect on the planetary atmosphere in respect to its activity cycle of about 11 years [Messerschmid2000]. While the day/night cycle has only a rather small effect, the "breathing" of the atmosphere in the rhythm of the solar cycle determines the density at certain altitudes significantly.

Planetary albedo and thermal irradiation

Besides direct solar irradiation, the orbited planet reflects sunlight (albedo) and emits infrared radiation itself. Both play a considerable role for thermal radiators but dwarf against direct sunlight. Albedo acts also as a radiation pressure source. Because the radiation pressure influence is relatively small at low altitudes compared with other perturbations (especially drag) and of minor importance beyond LEO due to the farther distance, it is only interesting for analysis of low altitude orbit of celestial bodies without an atmosphere, e.g. the Moon. Thus, the implementation of albedo radiation is currently under analysis but not implemented within IRIS++ yet.

Impact of meteoroids and orbital debris

For safety analysis the estimation of impacting hazards on critical spacecraft components is an important task. Close to Earth, radar techniques can identify only parts larger than about 10 cm [Hess1998]. Thus the smaller objects (metal parts and lacquer fragments) down to about 1 cm form the largest danger, since their trajectory cannot be predicted and their energy might be sufficient to penetrate the spacecraft's hull. Pairing this with their extraordinarily high velocities of up to 15 km/s unfolds their potentially devastating effect. However, models exist in order to describe statistically the distribution and the most frequent directions [MASTER]. Together with a computation of geometrical "incidence" of debris on space station surface elements one can identify suitable or preferable accommodation locations for external payloads and sensitive external system components. Such application was demonstrated for ISS' originally planned *Crew Return Vehicle* (CRV) [Langholf1999] using the surface model and shadowing algorithm of IRIS++.

Perturbation Effects

The effects caused by environmental influences are called *perturbations*, which disturb the ideal KEPLER orbital trajectory. Perturbations to the spacecraft orbit can principally be divided into two types: *internal* perturbations including all gravitational influences, and *external* perturbations, including all other influences, especially surface forces and thrust operations. In order to take environmental effects into account that are related to surface interactions, an appropriated spacecraft surface modelling (see section 3.2.4) as well as a shadowing algorithm (see section 4.5) are necessary.

4.4.2 Aerodynamic Drag Force

Spacecraft at low altitudes are decelerated due to the aerodynamical drag induced by the residual atmosphere. Therefore, especially below 500 km Earth altitudes, or 260 km for Mars, permanent or periodical reboosting is mandatory, involving a relatively high amount of delta-v. Orbit decay, lifetime and delta-v are best calculated via numerical simulation, but analytic estimation formulas do exist [Messerschmid2000].

In the orbit perturbation acceleration equation (27) and in the attitude perturbation torque equation (47) one finds the drag force term $\overline{F}_{drag}$. The product of the *dynamic pressure* with the *drag effective area* A_{drag} and the *drag coefficient* c_D gives this term:

$$(51) \qquad \overline{F}_{drag} = -\overline{e}_v \cdot \underbrace{\frac{\rho}{2} \cdot v^2}_{\text{dynamic pressure}} \cdot c_D \cdot A_{drag}$$

Thus, the drag force is proportional to the current density and current drag area and directs opposite to the velocity vector $\overline{v}$. Latter is valid in most cases [Montenbruck2000] because one can assume that the atmospheric particles are completely adsorbed by the surface, thus, transferring their total kinetic impulse in the direction of the flux.

While the shadow algorithm (see section 4.5) provides a calculation of the effective drag area A_{drag}, the drag coefficient c_D cannot be specified well a priori. The value 2.0 is valid for a sphere and values from 2.3 to 2.5 can be chosen for most non-spherical and convex spacecraft ([Hinüber2002],[Montenbruck2000]).

Density of the residual atmosphere

Equation (51) contains the spacecraft surrounding density at a given location and time. To obtain this density information, atmospheric models of the planets are used. These empirical models contain approximation formula with several parameters, including the PCPF position (as geographic longitude, latitude and radius), the solar activity factor $F_{10.7}$ stating the Sun's spectral radiation density at 10.7 cm wavelength and the geomagnetic activity factor A_p (if applicable). The influence of the Sun is by far the strongest, as demonstrated in Table 4.2.

Table 4.2: Density variations for a circular Earth orbit at 400 km altitude [Hinüber2002]

Solar Activity (11y cycle)	Local Time	Local Latitude	Geomagnetic Activity	Semi-Annual Variations	Solar Activity (daily)	Local Longitude
1165 %	115 %	60 %	60 %	50 %	5 %	5 %

Although these models come typically as FORTRAN code and could be used as library in C++ programs, they have not yet been included directly in IRIS++ to reduce computation time. In any case, for conceptual design purposes it is sufficient to use density information at representative conditions, say at low, medium and high solar activity. Thus several look-up tables were calculated and stored in so-called density files (*.RHO). IRIS++ reads these tables and interpolates the density as necessary. This is available in all three dimensions, but the tables currently used are solely altitude-dependent, produced by averaging globally (i.e. over all longitudes and latitudes) at specific conditions.

The atmosphere models and conditions used within IRIS++ are:

- MSISE93 model for Earth [Chabrillat1995] at:
  - Low activity: $F_{10.7}$ = 76.9, A_p = 12.7 (corresponding date: 01.01.2008)
  - Medium activity: $F_{10.7}$ = 121.1, A_p = 18.7 (corresponding date: 01.01.2010)
  - High activity: $F_{10.7}$= 231.7, A_p = 21.0 (corresponding date: 01.01.2012)
  - Average: $F_{10.7}$= 107.8, A_p = 13.4 (mean value by 2007-2016)

- Mars-GRAM 2000 model for Mars [Justus2000] at:
  - Low activity: $F_{10.7}$ = 76.9 (corresponding date: 01.01.2008)
  - Medium activity: $F_{10.7}$= 121.1 (corresponding date: 01.01.2010)
  - High activity: $F_{10.7}$ = 231.7 (corresponding date: 01.01.2012)
  - Average: mean value by 2008-2018

4.4.3 Solar Radiation Pressure

When a spacecraft is exposed to solar irradiation, the interaction of the electromagnetic particles or photons with the surface produces a small force, called solar radiation pressure. The amount and direction of the resulting force vector depends on:

- Irradiated energy
- Illuminated surface area
- Surface's optical properties, namely the absorption/reflection capacity

Mathematically the total force equation due to solar radiation pressure in the SRCS frame is given by the following equation (derived from [Montenbruck2000]):

$$\overline{F}_{srp} = -p_{sr} \cdot k_{ecl} \cdot \left(\underbrace{\overline{e}_{sun} \sum_i \left((1-\varepsilon_i) \cdot A_i^{sr} \right)}_{\text{due absorption}} + \underbrace{2 \sum_i \varepsilon_i \cdot \overline{n}_i \cdot (\overline{n}_i \cdot \overline{e}_{sun})}_{\text{due reflection}} \right) \tag{52}$$

with

A_i^{sr}	: effective Sun-illuminated area of i -th surface element
$\overline{e}_{sun}$	: unit vector towards the Sun (ASL)
ε_i	: emissivity of i -th surface element
$\overline{n}_i$	: normal vector of i -th surface element
k_{ecl}	: eclipse factor $k_{ecl} \in (0;1)$

and the solar radiation pressure:

(53) $$p_{sr} = \frac{\Phi}{c} = \frac{P_{Sun}}{4\pi \cdot s^2 \cdot c} \underset{\substack{\uparrow \\ 1\,\mathrm{AU}}}{=} \frac{1367\ \mathrm{W/m^2}}{c} = 4.56 \cdot 10^{-6}\ \mathrm{N/m^2}$$

with Φ : solar flux
c : vacuum light speed
P_{Sun} : gross output power of the Sun

Typical radiation pressure properties, namely reflectivity and absorptivity, are provided in Table 4.3. For spacecraft with large solar arrays directing towards the Sun, one can neglect the off-directing surface elements, thus assuming the effective normal vector directing to the Sun [Montenbruck2000]. In this case equation (52) simplifies to:

(54) $$\overline{F}_{srp} \approx -\, p_{sr} \cdot k_{ecl} \cdot C_R \cdot \overline{e}_{sun} \cdot A_{total}^{sr}$$

with $C_R = 1 + \varepsilon$: radiation pressure coefficient

Table 4.3: Optical properties of sample spacecraft components [Montenbruck2000]

Coefficient	**Solar panel**	**High gain antenna**	**Solar sail (aluminium coated Mylar)**
ε	0.21	0.30	0.88
$1-\varepsilon$	0.79	0.70	0.12
$C_R = 1+\varepsilon$	1.21	1.30	1.88

4.4.4 Non-Spherical Gravitation

With the two-body description of spacecraft orbits the orbited planet (or central body) is treated as a point mass with a spherical-shaped gravitation potential field. In reality planets are not spheres with homogeneous mass distribution, thus the spacecraft acceleration deviates from this central force simplification. For orbit prediction or trajectory determination purposes, an accurate representation of the central body is necessary. According to ESA [ECSS-E1004] the *geopotential* field of a planet is represented by the central force augmented by a series of spherical harmonics as:

(55) $$U(r,\phi,\lambda) = \frac{GM}{r} + B(r,\phi,\lambda)$$ with the non-spherical term:

(56) $$B(r,\phi,\lambda) = \frac{GM}{r} \sum_{n=2}^{N} \sum_{m=0}^{n} \left(\frac{R}{r}\right)^n \cdot \overline{P}_{nm}(\sin\phi) \cdot \left[\overline{C}_{nm}\cos(m\lambda) + \overline{S}_{nm}\sin(m\lambda)\right]$$

with G : universal constant of gravitation
M : central body's mass
R : semi-major axis of the gravitational reference ellipsoid (commonly taken as the equatorial radius of the central body)
r : spacecraft's radial distance from central body's centre of mass
ϕ : planetocentric latitude
λ : planetocentric longitude
n,m : degree and order of the harmonic term
N : maximum degree of the potential equation

$\overline{C}_{nm}$, $\overline{S}_{nm}$	: spherical harmonic coefficients (normalised)
$\overline{P}_{nm}$	: associated Legendre functions of the first kind (normalised)

The latter are expressed by:

(57) $$\overline{P}_{nm}(u = \sin\phi) = \underbrace{\sqrt{\frac{(2n+1)k(n-m)!}{(n+m)}}}_{normalization} \cdot \frac{(1-u^2)^m}{2^n n!} \cdot \frac{d^{n+m}(u^2-1)^n}{du^{n+m}}$$

with $k = 1$ for $m = 0$
$k = 2$ for $m \neq 0$

In the orbit perturbation acceleration equation (27) the perturbation acceleration term regarding the non-spherical shape of the central body is:

(58) $$\overline{a}^{PCPF}_{non-spherical} = \nabla B = \begin{bmatrix} \frac{\partial B}{\partial r} \\ \frac{1}{r}\frac{\partial B}{\partial \phi} \\ \frac{1}{r\cos\phi}\frac{\partial B}{\partial \lambda} \end{bmatrix}$$

with the partial derivations [Drodzek2002]:

(59) $$\frac{\partial B}{\partial r} = -\frac{GM}{r^2}\sum_{\mathrm{n}=2}^{N}\sum_{m=0}^{n}\left(\frac{\mathrm{R}}{\mathrm{r}}\right)^n \cdot (n+1)\cdot \overline{P}_{nm}(\sin\phi)\cdot\left[\overline{C}_{nm}\cos(m\lambda) + \overline{S}_{nm}\sin(m\lambda)\right]$$

(60) $$\frac{\partial B}{\partial \phi} = \frac{GM}{r}\sum_{\mathrm{n}=2}^{N}\sum_{m=0}^{n}\left(\frac{\mathrm{R}}{\mathrm{r}}\right)^n \cdot \left[\overline{C}_{nm}\cos(m\lambda) + \overline{S}_{nm}\sin(m\lambda)\right] \cdot \left[\sqrt{(n-m)(n+m+1)(1+\eta)}\cdot \overline{P}_{n,m+1}(\sin\phi) - m\tan\phi\cdot\overline{P}_{nm}(\sin\phi)\right]$$

(61) $$\frac{\partial B}{\partial \lambda} = \frac{GM}{r}\sum_{\mathrm{n}=2}^{N}\sum_{m=0}^{n}\left(\frac{\mathrm{R}}{\mathrm{r}}\right)^n \cdot \overline{P}_{nm}(\sin\phi)\cdot m\cdot\left[\overline{S}_{nm}\cos(m\lambda) - \overline{C}_{nm}\sin(m\lambda)\right]$$

with $\eta = 1$ for $m = 0$
$\eta = 0$ for $m \neq 0$

Here the evaluation of the acceleration components depends on the coordinate representation. The original equations are formulated in the rotating PCPF coordinate system using spherical coordinates, thus it must be transformed and derived for usage in cartesian and equinoctial coordinates (see below).

To apply these formulae for individual celestial bodies, their spherical coefficients are needed. Typically they are gained through gravimetric models and tabulated to a certain degree and order. Within IRIS++ the following models are implemented, each up to an order and degree of 5:

- EGM96 model for Earth [EGM96]
- GMM-2B model for Mars [GMM2B]
- JGL075G1 model for the Moon [JGL75]
- MGN120 v1.0 model for Venus [MGN120]

Cartesian coordinates

The orthogonal transformation matrix for converting to cartesian coordinates and transforming them into the inertial PCEQ system is [Hwang1998]:

(62) $$\bar{a}^{PCEQ}_{non-spherical} = \begin{bmatrix} \frac{x}{r} & \frac{-xz}{r\sqrt{x^2+y^2}} & \frac{-y}{\sqrt{x^2+y^2}} \\ \frac{y}{r} & \frac{-yz}{r\sqrt{x^2+y^2}} & \frac{x}{\sqrt{x^2+y^2}} \\ \frac{y}{r} & \frac{\sqrt{x^2+y^2}}{r} & 0 \end{bmatrix} \cdot \bar{a}^{PCPF}_{non-spherical}$$

Evaluated, this equation is used as the orbit perturbation acceleration term due to non-sphericity for orbit integration in cartesian representation. Note, that the spherical coordinates in PCPF used in equations (58) to (61) are gained from inertial cartesian coordinates in PCEQ by:

(63) $$\phi = \arcsin\left(\frac{z}{r}\right)$$

(64) $$\lambda = \arctan\left(\frac{y}{z}\right) - W$$

(65) $$r = \sqrt{x^2 + y^2 + z^2}$$

The term W is one of the rotational elements of the planet (section 4.4.6), indicating the location of its main meridian at a given date.

Equinoctial coordinates

For orbit integration using the equinoctial orbit representation, we must convert the perturbation acceleration given with equation (62) to the local orbital coordinate system (LOCS). This is done according to the following transformation [Drodzek2002]:

(66) $$\bar{a}^{LOCS}_{non-spherical} = \bar{\bar{T}} \cdot \bar{a}^{PCEQ}_{non-spherical}$$

with $$\bar{\bar{T}} = \begin{bmatrix} \cos u \cos\Omega - \sin u \cos i \sin\Omega & \cos u \sin\Omega + \sin u \cos i \cos\Omega & \sin u \sin i \\ -\sin u \cos\Omega - \cos u \cos i \cos\Omega & -\sin u \sin\Omega + \cos u \cos i \cos\Omega & \cos u \sin i \\ \sin i \sin\Omega & -\sin i \cos\Omega & \cos i \end{bmatrix}$$

and the orbital elements given with equations (9) and (19) to (21) on page 52.

4.4.5 Third-Body Gravitation

Another gravitational perturbation arises due to the gravitational field of celestial bodies near the orbited central body. Because these bodies are typically rather far away, a simple point mass model according to NEWTON is applicable. Some care is required here, however, because the origin of the coordinate system describing the spacecraft orbit (typically PCEQ) is now not inertial and at rest anymore, but itself subject to an acceleration. The resulting equation, taking the accelerations by other celestial bodies into account, is given by [Montenbruck2000]:

$$\bar{a}_{3rd-body}(\bar{r}) = \sum_i GM_i \cdot \left(\frac{\bar{s}_i - \bar{r}}{\left|\bar{s}_i - \bar{r}\right|^3} - \frac{\bar{s}_i}{s_i^3} \right) \tag{67}$$

with

G	: universal constant of gravitation
M_i	: i -th celestial body's mass
$\bar{s}_i$	: i -th celestial body's radius vector from the centre of mass of the central body
$\bar{r}$	: spacecraft radius vector from the centre of mass of the central body

To use this term, the radius vectors are formulated in the PCEQ system of the current central body. Thus, the coordinates of all involved celestial bodies are needed and derived from a solar system motion model (see section 4.4.6).

4.4.6 Solar System Motion Model

A motion model, describing the positions of major celestial bodies of the solar system as function of time, is necessary for several reasons. Firstly, the position of the Sun relative to the spacecraft is a prerequisite to calculate solar influences, be it sunlight irradiation, shadowing, eclipses, or Sun's gravitational influences. Depending on which body is currently dominating for orbit description (i.e. the central body) also its motion and its possible primary body (e.g. the Earth in case of the Moon as the central body) are necessary. Other planets may also be relevant for trajectory propagation if missions at regions of low acceleration shall be examined (Libration point orbits, weak stability boundary (WSB) missions). Furthermore, the prediction of the rotational motion of the celestial bodies is crucial for evaluation of atmospheric and non-spherical gravitational influences as well as ground-track calculations.

Thus, a generic approach for a solar system model capable of describing the motions of *any* relevant body benefits flexibility and extendibility. To enable this for IRIS++, the following approach was chosen:

- The solar system planets' motions are modelled via a polynomial approximation of fourth order [Meuss1998]. The orbital ephemerides of individual bodies are calculated with a set of characteristic coefficients.
- The Lunar orbital ephemerides are currently modelled with series expansion formulae providing the geo-centric lunar longitude and latitude with an accuracy of several arc-minutes and about 500 km in the lunar distance.
- The rotational motion of the planets is modelled with a common approach of *rotational ephemerides* that also uses a set of characteristic coefficients for each body [Montenbruck2001].
- All body-specific coefficients are stored in a planet file (*.PLN).

This approach is beneficial in allowing editing or exchanging of motion models without altering the simulation programme itself. A model indicator in the planet file lets the user select the preferred model, or rather input format. Therefore, for short-term analysis, the user can alternatively provide the orbital elements as Kepler orbital elements.

If requirements for higher accuracy exist, the JPL Ephemerides should be used [DE405]. These include high-precision ephemerides of all major solar system bodies in a specific time span and are based on various observation databases and numerical integration of the n-body problem.

4.5 Shadow Computation

All environmental influences originating from interactions with the spacecraft surface depend on the geometry and topology of the spacecraft configuration. Typically these influences have an irradiation or rather a "flux" character, thus shadowing can play a major role in investigations. Analysis has proven that neglecting shadowing causes typical relative errors of up to 80% for calculations of the effective area [Yazdi2001]. In summary, relevant influences include:

- aerodynamic forces and torques due to particle flux of the residual atmosphere,
- solar radiation pressure forces and torques,
- solar irradiance on solar arrays and thermal radiators defining their efficiency,
- probability assessment of meteoroid and space debris impacts.

The prerequisite for any shadow computation is a surface model. In section 3.3 the surface model designed for this purpose was described. It represents the spacecraft surfaces as a set of polyeders, i.e. closed, convex and not penetrating objects consisting of convex opaque polygons between a mesh of vertices. A valid simplification for all the above-mentioned influences is that the irradiance can be treated to be parallel, where the rays, or rather the particles, do not interfere with each other. Thus, penumbrae do not occur and the same approach in computing shadow effects can be chosen.

In order to compute shadow-dependent perturbation forces and torques, a generic algorithm was developed. This section summarises its principles and application.

4.5.1 Requirements of Shadow Computation for Spacecraft Simulation

In general, the task of a quantitative shadowing algorithm is to compute the amount of the shadowed or lightened (i.e. non-shadowed) area in respect to the total area. Furthermore, the centre of pressure (CoP) of the lightened area is required for torque calculations.

Depending on the application, the investigation focuses on the calculation of the irradiance on individual components (i.e. solar or radiator panels) or the irradiance of the whole spacecraft surface (i.e. for perturbation calculations). Thus, a further requirement of the algorithm is flexibility to allow for both types of calculations and to identify individual surface element irradiation properties afterwards. The latter is also essential when the resulting surface force vector and the centre-of-pressure have to be found. Furthermore, this routine has to take the distribution, orientation and optical properties of the individual surface elements into account to be able to calculate the surface perturbation impulse correctly.

The irradiance calculations, with a potentially highly complex surface model involved, obviously have to be performed repeatedly during simulation because of the permanently changing spacecraft attitude and orientation relative to the irradiance vectors. Thus, shadow calculation can take a considerable amount of computer time and must be designed to be as fast and efficient as possible.

4.5.2 Design of the *Priority-List Shadow-Buffer Algorithm*

The developed shadowing algorithm calculates quantitatively the lightened (i.e. the non-shadowed) part of individual surface elements with respect to a random irradiance vector. To do so, two subtasks have to be performed: a) finding the *shadowing order* in respect to an irradiance vector and b) finding the *shadowed parts* of the surfaces of a scene.

The computer graphics domain offers a wide range of concepts to identify *hidden-line* and *hidden-surface* elements ([Newman1986],[Sutherland1976]). In this connection, we have an observer instead of an irradiation source illuminating the scene. The objective of the involved methods is the generation of more or less realistic images. Finding out *how much* of each object is hidden (or shadowed analogously) is irrelevant. For this numerical simulation problem only few approaches exist. One found is a method used to estimate the influence of shadowing on the solar panels of the ISS [Chung1995]. This method does not rely on a common surface model, but approximates the projection areas of all actual ISS bodies with rectangles, which were resized and oriented explicitly towards the Sun. In summary, this method is too inaccurate to be of practical use and not generally applicable. Thus, an appropriate method had to be developed. For this purpose some principles of computer graphics algorithms were selected and combined coherently with new components and new algorithms. Its name is derived from the two key elements, which are a *priority list* for finding the shadowing order and a *shadow buffer* for calculation of the shadowed areas of each surface. In this section the main components and principles of the algorithm are summarised. For further information the reader is referred to a previous publication [Yazdi1999].

Priority List

Based on a surface model of plane polyeders, the algorithm treats all surface elements (i.e. individual polygons) of one body as a *cluster*. Because the polyeders are presumed to be closed and of convex shape, the polygons of one cluster cannot cast shadows on each other. Off-facing polygons, i.e. polygons with normal vectors directing away from the source of irradiance, are completely non-lightened. Furthermore, if one cluster A is *in front of* another cluster B, all its surfaces are in front of the surfaces of B. Thus, the effort of finding the shadowing order drastically decreases because one has only to sort the clusters in the order of occurrence with respect to the irradiation vector (so called *inter-cluster priority list*) and not all surfaces. In doing this, the priority marks the order such that a cluster of high priority can shadow a cluster of low priority. This priority, or rather the order of one cluster in respect to all other clusters, must be found every time when the spacecraft orientation changes considerably with respect to the irradiation. For this task separating planes and a *Bubble-Sort* method are utilised.

Search for Separation Planes

In order to distinguish whether cluster A is in front of cluster B or vice-versa, separation planes can be used, when no cluster penetrations occur. In doing this, the product of the plane's normal vector and the irradiation vector determines the order. But the task of finding these planes between arbitrary shaped polyeders can be rather complex, and a mix of three succeeding approaches is applied; first the easiest and fastest, then the methods of increasing complexity.

Figure 4.6 illustrates how an arbitrary plane is checked for its *separation property*. We see that if the maximum distance of reference-body vertices along the plane normal vector is lower than the minimal distance of the check-body vertices, the plane separates these bodies. Furthermore, from this figure we can intuitively find that the unit vector, which is parallel with the connecting line of both origins each located approximately at the geometrical centres, may have a separation property. In fact, this is often true if the *surrounding spheres* (defined by the maximum radial distance of the vertices from the centre) do not penetrate, thus the bodies do not lie in close vicinity. But this is often not sufficient and the next check, or rather search method, applies.

In most cases experience has shown that one of the clusters' surfaces is a separating plane. Therefore, the second method simply has to check the normal vectors of all surfaces. These normal vectors are pre-calculated during surface model generation and therefore available anyway. If this test should fail also, the last method will always succeed if the objects are indeed separated. It is based on the hypothesis that the plane defined by the nearest surface connection of two convex and not penetrating clusters separates them. With the knowledge of the body geometries (primitives such as box, cylinder, cone, etc.) an iterative process can be designed starting with the centre-to-centre connection and ultimately finding this nearest connection. Figure 4.7 illustrates the process of finding a separation plane typically after 3 to 5 iterations.

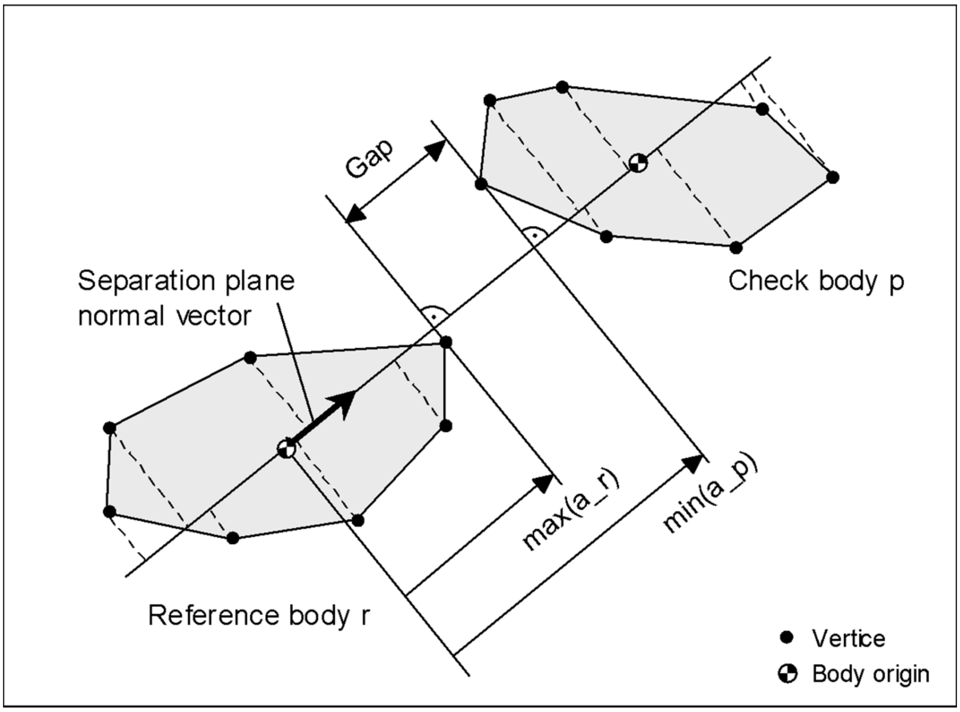


Figure 4.6: Surface model separation test

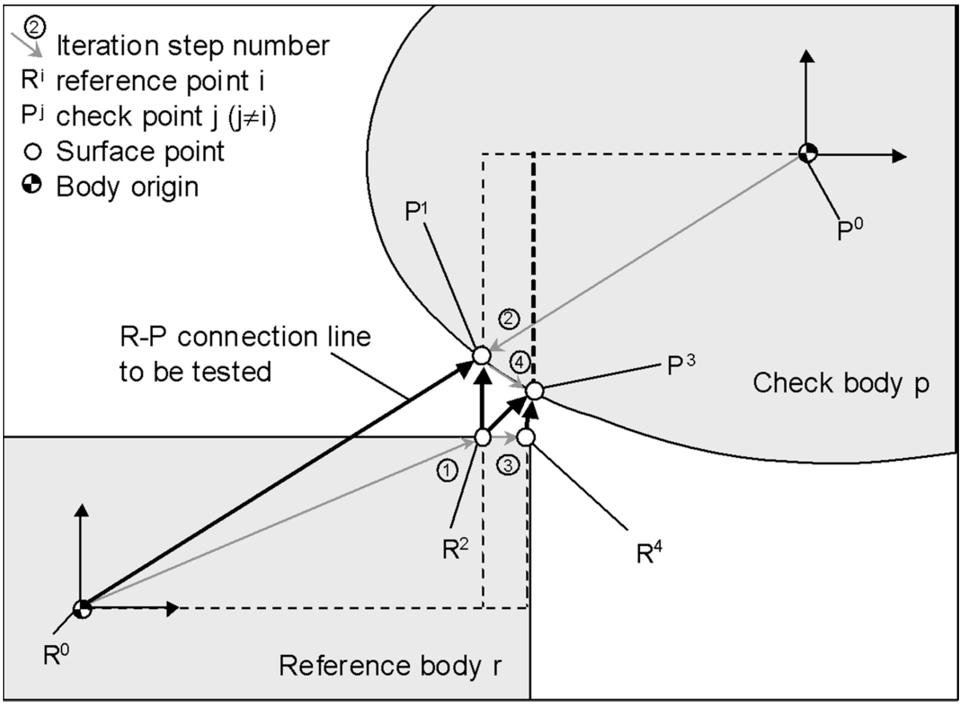


Figure 4.7: Iteration process for finding a separating plane

Note that the process of finding separation planes is only done once at programme start and a look-up table (an *anti-symmetric separation plane matrix*) is used during the simulation. If structural dynamics (e.g. panel tracking) change the scene considerably, the few involved planes are checked for validity and updated if necessary.

After sorting the clusters in order of their priority, the quantitative calculation of the shadowed areas of each surface is next. For this task grid-oriented and vector-oriented approaches can be envisioned. Because we deal typically with rather complex scenes with a large number of objects, i.e. is hundreds or thousands of surface elements, variable, user-definable accuracy allowing fast execution is of primary importance. Therefore, a so-called *shadow buffer* method was developed and is currently in use.

Shadow Buffer

A shadow buffer is a dynamical two-dimensional array containing shadowing information of the area of the projection plane (perpendicular to the irradiance) where shadowing currently might occur. The shadow buffer frame (Figure 4.8) defining this area is determined by *Minmax-tests* between relevant clusters that can potentially cast shadows on other surfaces. The frame location and dimension changes during simulation as the shadowed areas and the selection of involved bodies changes. In contrast to the total projection area defined by the bounding-box of the scene, using a shadow buffer saves computer time and memory because only the area of this frame must be initialised and processed during shadow cast calculation.

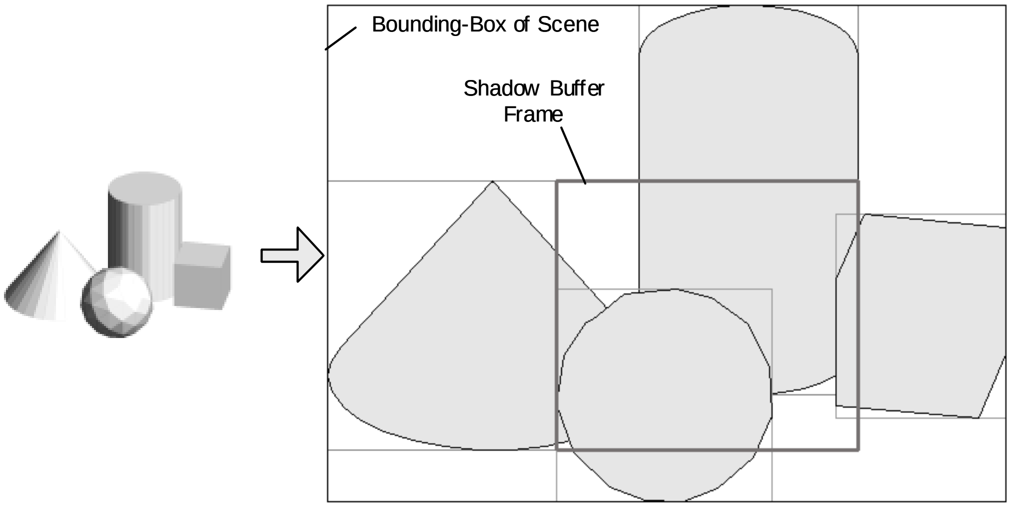


Figure 4.8: Shadow buffer frame including all possible shadowed areas

Using a shadow buffer involves casting projections of clusters, or rather surfaces, to the grid defined with the shadow buffer. The size of this grid defines the *resolution* or the number of *pixels* of the shadow buffer and can be specified by the user in pixel per meter. The process of setting pixels corresponding to the surface projection is called *scan conversion*. For this a *polygon fill algorithm* is used, which is based on computer graphics scan-line algorithms [Newman1986] and an enhanced BRESENHAM line drawing algorithm to identify the span of each scan line to be filled. The effective projection area of a surface is therefore represented by a certain number of pixels. To identify how many of these pixels are shadowed, casting is started by clusters of highest priority respectively their surfaces on to the buffer. If a pixel is already set during scan conversion (i.e. by a surface of higher priority) the surface currently processed is shadowed at this place. The ratio $r_i^{lightend}$ of the non-shadowed (i.e. lightened) surface i is therefore given by:

(68) $$r_i^{lightened} = \frac{N_i^{total} - N_i^{shadowed}}{N_i^{total}}$$

with N_i being the indicated number of pixels of i-th surface. The CoP is determined by calculating the average x- and y-coordinates of all non-shadowed pixels during scan-conversion.

4.5.3 Effective Irradiated Surface Area

The effective irradiated area A_i of each surface element is obtained with the *facing ratio* and the lightened ratio obtained before:

(69) $$A_i = A_i^{total} \cdot \underbrace{(\bar{n}_i \cdot \bar{e}_{rad})}_{r_{facing}} \cdot r_i^{lightened}$$

with

A_i^{total}	: total area i -th surface element
$\bar{n}_i$	: normal vector of i -th surface element
$\bar{e}_{rad}$	: unit vector parallel to irradiance direction (pointing away from source)

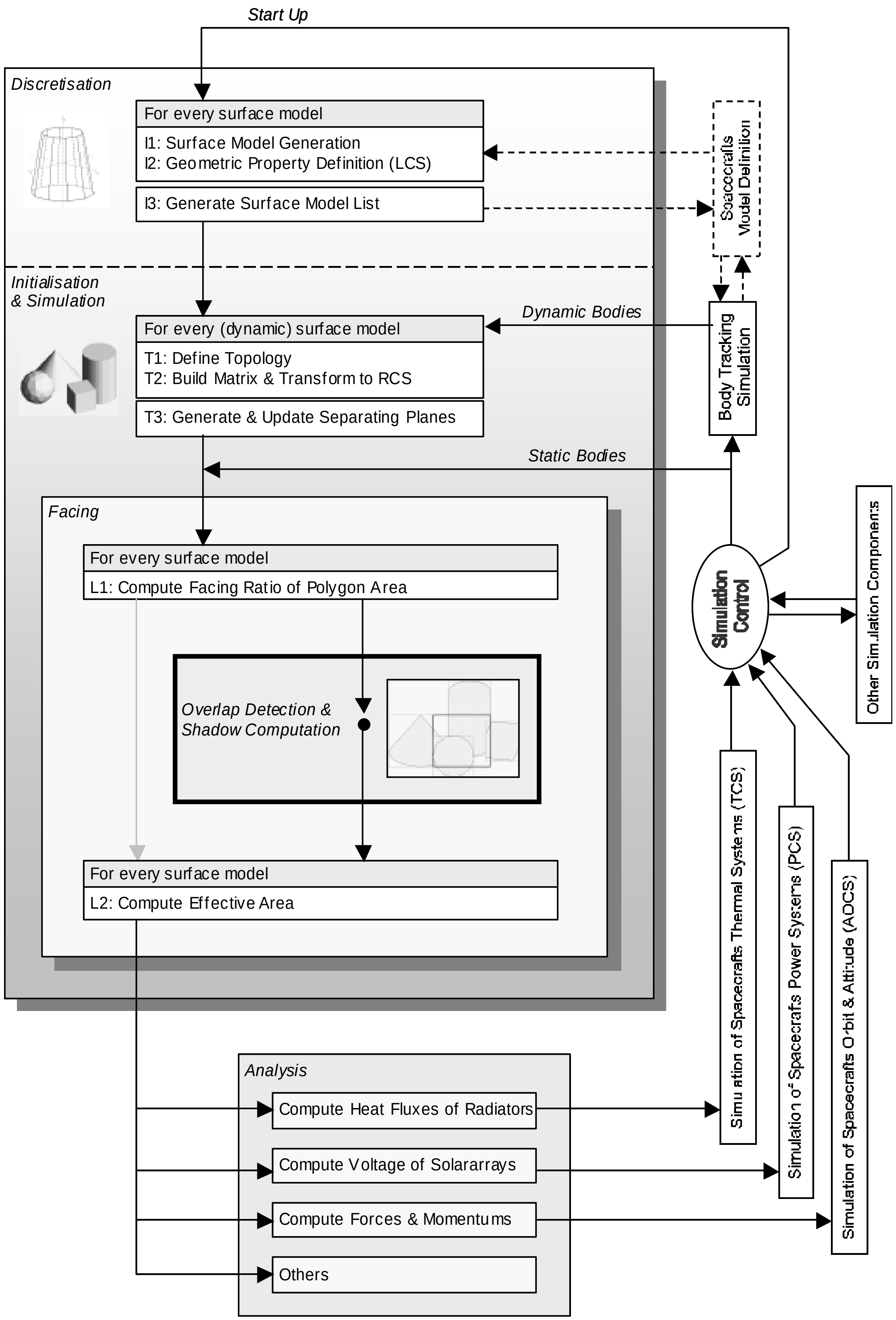


Figure 4.9: The shadow algorithms embedded in the simulation process

5 Application to a Lunar Space Station Scenario

Current human spaceflight plans envision missions to the Moon and Mars. This offers multiple system and mission design tasks, ranging for example from staging stations and transfer vehicles to surface base concepts. One specific example regarding the NASA Mars Design Reference Mission [DRM3] is the design of a space station, or rather transfer vehicle, called Earth Return Vehicle (ERV) serving as a "mother-ship" for orbital rendezvous and flight back to Earth of a returning crew from the surface of Mars. This spacecraft would stay up to 3.5 years in Mars orbit. A first numerical characterisation analysis of such Mars orbital space stations was performed during this work by DROZDEK [Drodzek2002].

Similarly, a lunar space station for orbital support of lunar surface activities has been chosen here as an example of a more near-term scenario. This chapter exemplifies the modelling and simulation tools developed within this dissertation and how they can be applied to a spaceflight mission scenario beyond LEO. As a point of departure a lunar space station scenario is defined with a minimum configuration station, serving as a stepping-stone and gateway to further lunar activities. The scenario addresses ESA's preparatory programme *Aurora* [ESA2001], indicating Europe's long-term intentions for the next 30 years, and the recently formulated new US *Space Exploration Policy* of NASA with focus on human exploration of the solar system and primarily targeting the Moon and Mars [Bush2004]. The visions foresee a human return to the Moon surface by 2020 and the first human expedition to Mars by 2030. Activities in the Earth-Moon system are seen as the "next logical step" for the development of space and to prepare successive human space operations beyond.

Although no details were known at the time this was written, the actual range of possible utilizations of a lunar space station is manifold and shall be addressed in the beginning of this chapter. This is followed by a discussion of the associated challenges regarding mission analysis, planning and simulation. Finally, the sample lunar exploration architecture is presented including results demonstrating tools application to a sample conceptual design problem.

The Challenge of Space Station Design beyond LEO

Conceptual system design of space stations addresses issues concerning station configuration as well as its systems and subsystems. The most relevant issues are summarised in Table 5.1. Because non-LEO space stations must be operated and eventually also assembled at rather far distances from Earth, the transfer legs to and from the station for crew and cargo transport become similarly mission relevant as the station system design. Therefore both, the transfer vehicles and the space station, are discussed. Table 5.2 summarises mission aspects of major importance during conceptual design. Compared with LEO space stations, mission design aspects become more complex. The complexity increases with increasing distance from Earth and extending stays at locations far away from Earth. Systems and mission concepts become more interdependent in terms of the increased number of mission and system elements, transfer times and logistical constraints, whereas spacecraft and transfer vehicles pass or stay at low acceleration regions. Regarding mission analysis and simulation, the rather few dominating effects of a central body in low altitude orbits are being replaced by a mix of various influences, which have to be taken into account. This is demonstrated by a survey of perturbations within section 5.3. Because not all relevant issues can be represented in sufficient detail within the frame of this dissertation, for practical reasons a concept summary is presented.

Table 5.1: System design characteristics and parameters of space stations beyond LEO

Area	Characteristics and Parameters
Configuration	• Total mass • No. and type of modules • Configuration design/architecture • Safety (e.g. dual egress & dual ingress, safe haven)
Crew	• Crew size • Life support strategy (in/outputs, recycling, radiation protection) • EVA and robotics • Human factors (incl. habitability, safety)
Subsystems	• Attitude and orbit control • Environmental control and life support • Power • Propulsion • Structures and mechanisms • Synergies
Logistics	• Storage and transport masses (re-supply, disposal) • Diversity of transport goods

Table 5.2: Mission design characteristics and parameters of space stations beyond LEO

Area	Characteristics and Parameters
Station assembly	• Assembly sequence and schedule (no. of launches, launch rate) • Safe operability at every stage (station keeping, life support) • Launcher accessibility (departure orbit) • Transfer stage constraints (mass, dimensions) • EVA and robotics requirements • Docking and berthing requirements
Station orbital dynamics	• Orbit acquisition and control delta-v (station-keeping) • Actuator system (propulsion) • Actuator active times (manoeuvre scheduling) • Ground station coverage
Transfer vehicle orbital dynamics	• Departure/arrival locations • Launcher accessibility • In/outbound transfer trajectories • Transfer window periods, contingency crew return • Transfer delta-v • Orbit manoeuvres delta-v • Return path constraints (aero-capture, aero-braking/re-entry) • Ground station coverage
Attitude dynamics and control (all space segment elements)	• Flight mode • Attitude control strategy and delta-v • Solar tracking • Antenna tracking
Utilization and operation	• Life time • Early utilization • Full operational utilization tasks • Crew size • Re-supply and disposal (mass, diversity of transport goods) • Logistics (flight time, periods, scheduling) • Environmental influences (eclipses, radiation, debris) • Ground station communication link times

5.1 The case for the Moon

The Moon, the next celestial neighbour to Earth, is more accessible than all other bodies in the solar system, thus human infrastructures will be far better maintainable and utilizable than on any other celestial body by access of shorter period and with higher frequency. Furthermore, the Moon offers a long-term perspective for economical exploitation of space (in terms of propellant and raw material production as well as tourism) and offers simultaneously excellent early utilization possibilities in scientific research and technological developments for successive steps of mankind into the universe, be it exploration or colonisation.

Science rationales

By extending human lunar activities science will benefit greatly because compared to automated probes any human space mission is capable of transporting considerable mass and diversity of scientific payloads and the resulting built-up infrastructure fosters research activities that a purely scientific budget could not afford and that would not otherwise take place [Crawford1998]. The most compelling arguments supporting the scientific case for a return to the Moon are based on geosciences, life sciences, astronomy and solar physics [ESA2003].

Beside its good accessibility, the Moon features no considerable atmosphere, has no tectonics and rotates rather slowly. Its surface is extremely old, preserving a record of the early evolution of terrestrial planets and the near-Earth cosmic environment [ESA1992]. This record includes the time period before 4.5 to 3.8 billion years that is not present on any other terrestrial body, except Mercury [Crawford2004]. But this "period saw the origin (or at least the establishment) of life on … Earth" and "it also covers an interesting period in the evolution of the Sun". Our current understanding of planetary and solar system history and evolution bases primarily on lunar surface observation and the samples brought back by the Apollo missions from only six landing sites in the vicinity of Moon's equator on the near-side (see Figure 5.1). Therefore, any approach for a better understanding ultimately will require thorough surface analysis including sample identification, collection and analysis on a far larger scale than Apollo [Crawford2003].

In summary, the following scientific rationales exist:

1. <u>Geophysical science of terrestrial planets and solar system history</u>

Dating and composition analysis of samples from the Moon's extreme ancient surface is the only sure way of obtaining the age of meteorite impact and the cratering rate. Similar to meteorites originating from Mars found on Earth, the Moon might have also preserved meteorites due to large impacts on celestial bodies like Mars, Venus and even the early Earth containing also very valuable information. If they could be located and recovered these samples would provide insight into the origin and evolution of the solar system and would generate significant advances in the geophysical science of the Earth, Moon and other terrestrial bodies [Crawford2003].

2. <u>Lunar geology and the history of lunar water</u>

The surface samples brought back by Apollo were extremely dry. In contrast to this, photographs of the surface show features that may have been created by fluids, including the possibility of water [Peplow2004]. Furthermore, investigations of the Moon's south pole region with the *Lunar Prospector* spacecraft has shown deposits of considerable amounts of hydrogen in dark and cold sinks ('cold-traps'). Although the chemical state is unknown yet, one possible explanation is water ice that could have been formed by impacts of asteroids and comets [Lucey2004].

3. Solar physics

Solar wind ions and cosmic ray particles are efficiently trapped in lunar regolith. Furthermore, volcanic activity on the young Moon and impact ejections created layers of former surface material. These layers provide "snapshots of the lunar surface environment at particular epochs billions of years ago" conserving "a unique record of the charged particles environment" as well as information on the flux and composition of the interplanetary dust in the early solar system. This historical data on the strength and composition of the solar wind will allow development and verification of models of solar evolution [Crawford2003] and its influence on Earth.

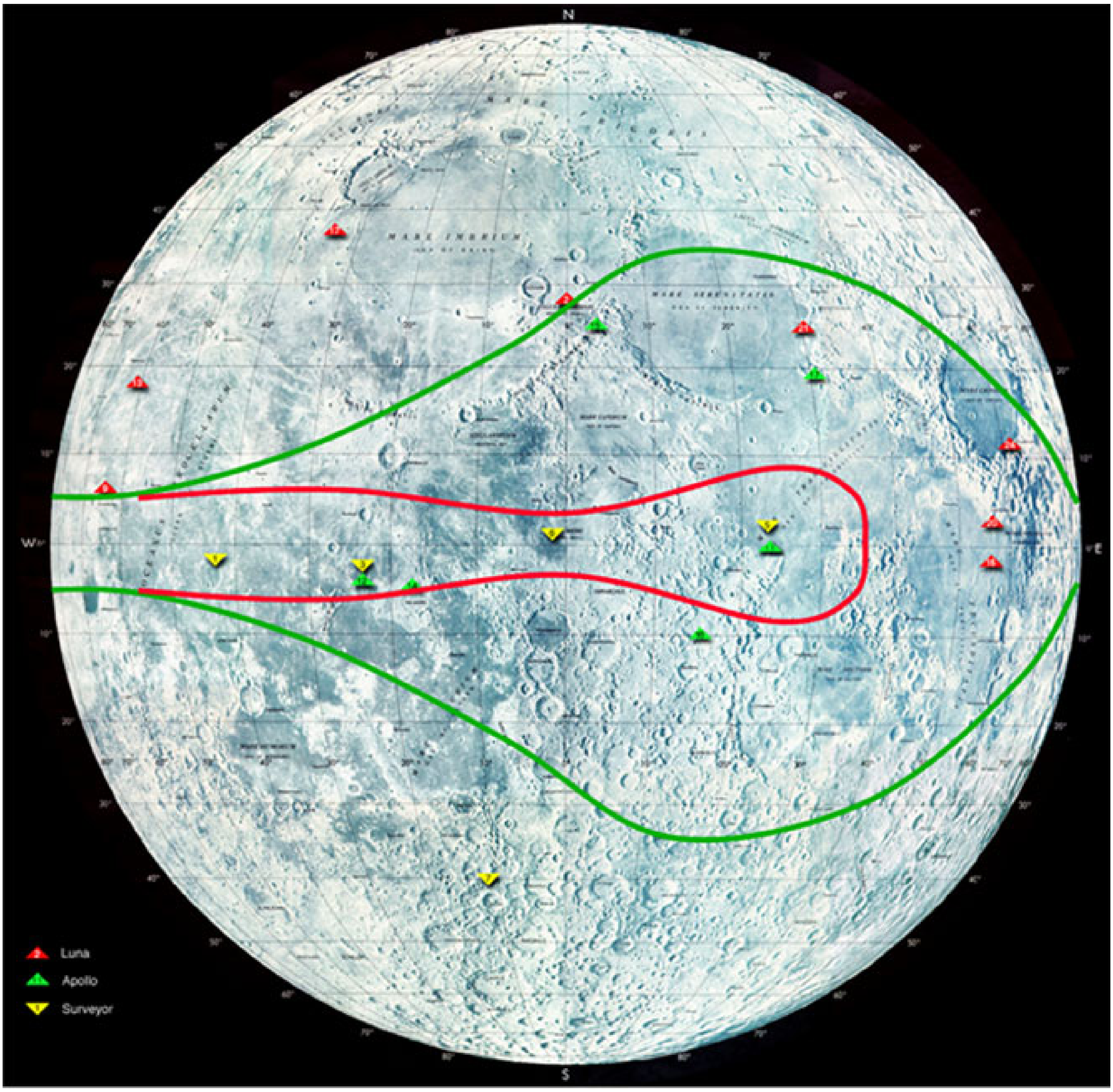


Figure 5.1: Map of previous lunar landings

Marked areas indicate reachable surface: Red: Initial free-return (Apollo 8, 10, 11) and modified free-return (12, 13, 14) trajectories with up to ±5° latitude; Green: Powered-return trajectories (15, 16, 17) with up to ±40° latitude

4. Astronomy

The Moon offers many advantages as a platform for astronomical observation [ESA1992]. And once the basic infrastructure is available one can expect that astronomers will begin exploiting it for their instruments [Crawford2003], just as they are now using the LEO environment for satellites (e.g. HST, XMM and many more) and possibly the external structure of the ISS (e.g. XEUS). The following key features in particular are present on the Moon for astronomical utilization ([Crawford1998],[ESA2003]):

- Stability of Moon's surface allows long-baseline optical and infrared interferometers.
- Slow rotation period offers the possibility of a very long integrations time on a single object with large telescopes.
- Extreme radio quietness of Moon's far-side will offer probably the best site for radio astronomy anywhere in the solar system.
- Cold, permanently shaded craters at the poles offer advantageous locations of large cryogenic telescopes.

These properties could revolutionise astronomy, including direct observation and analysis of extra-solar planets.

5. Life sciences, human physiology and psychology

Extended space activities including living and working in the hostile environment of space or on other planetary surfaces will create intentional or incidental advances in the knowledge of the human body and life in general. Although abstract and not well determinable in advance, they will certainly be considerable and will not only foster the expansion of terrestrial life to worlds beyond but will have a strong influence on biology, medicine and psychological disciplines on Earth.

Programmatic and long-term rationales

Current human spaceflight plans envision missions to the Moon and Mars. While spaceflights of astronauts to Mars would possibly capture more imagination and thrill as compared to a mission to the Moon, no relevant space agency and only a few experts favour a mission to Mars first. This is scientifically well founded with the search for existing or extinct life on the red planet but it is also driven by the belief in a higher level of feasibility and affordability of a *Mars Direct* mission [Zubrin1996], i.e. not requiring the ISS or the Moon as staging technology development basis or as competitor for institutional funds, which are deemed necessary. But the Moon is not only scientifically important as described above; it could also be the "most valuable real estate in the solar system" [Lucey2004] because of its location, properties and resources. The main arguments are based on the utilization and exploitation of lunar resources and the use of the lunar environment as a test-bed for the human exploration of Mars and the solar system:

1. Resources of the Moon

Similar to Earth, the Moon is rich in various minerals and metals that would support construction of surface and spaceflight structures, thus, lunar infrastructure will foster human spaceflight in a long-term perspective. From a near term point of view, the most relevant commodities are the considerable amounts of oxygen and hydrogen. Lunar rocks and soil contain about 45%

chemically bound oxygen, e.g. as ferrite-oxide (FeO) and titanium-dioxide (TiO_2). Various physical and chemical processes are available to extract this oxygen from the soil [Eckart1999][Taylor2004]. Hydrogen in some form must also be present, especially at the poles [Lucey2004]. This includes the possibility of over 10 billion tons of water ice [ESA2003]. Both, hydrogen and oxygen (or water) are assets for future space programmes due to their application in life support systems and for in-situ propellant production.

Furthermore, the Moon offers energy: Besides supply by regenerative means, the future energy supply on Earth will potentially reside on thermonuclear fusion reactors. Relevant reactor fuels are deuterium-tritium (DT) and deuterium-helium-3 (D^3He), whereas the latter has distinct environmental advantages. Deuterium is abundant on Earth, but terrestrial helium-3 is practically not available, but on the Moon, where it is generated by continuous bombardment of the lunar surface by the solar wind over billion of years. The mass fraction is hereby only about 13 mg/ton [ESA1992], thus, a large area must be processed to obtain a considerable mass of helium-3. However, only one kilogram of helium-3 is necessary to produce the same amount of energy as 10 billion kg of fossil fuel [ESA2003].

Addressing the US Congress, the geophysics scientist SPUDIS spoke about the Moon as "a scientific bonanza" and appended that the Moon is also "an economic treasure trove … that can revolutionise our [the US] national strategic and economic posture in space and at home." [Spudis2004]. This emphasises the geopolitical aspect that the Moon will possibly have in the second half of this century, in a world of scarce fossil fuels. The question is who will control the abundant lunar material and energy resources and what will this mean for the socio-economic stability on Earth?

2. Test-bed for the human exploration of the Solar System

Although lunar activities may not be mandatory for a human mission to Mars, the exploration of Mars and other destinations will greatly benefit from a return to the Moon. With having an operational lunar programme running, this operational environment can be used for demonstration of techniques and processes "in relatively safe reach to Earth" [NASA2004] including demonstration and verification of

- in-situ resource utilization (ISRU),
- landing, ascent and surface mobility,
- habitat and laboratory facilities, and
- drilling and sample treatment.

A prominent location for a lunar base is the south pole of the Moon. The possible availability of most valuable *in-situ* resources together with the relatively "stable and .. human-friendly temperature range of –30°C to –50°C" at the nearly perpendicular illuminated crater rims is tempting this region as a first lunar base location [Spudis1996].

Conclusion

While "exploration and colonization are the best reasons for having people up there [on the Moon], rather than science" [Jones2004], science will greatly benefit from a human lunar programme. Not only science is obviously linked to exploration, but scientific exploitation will demand a considerable amount of fieldwork for collecting more, and more diverse rock and soil samples, drilling and retrieving drilling cores from possibly up to hundreds of meters deep [Crawford2003] and returning the most promising to Earth. Furthermore, scientists will be interested in operating instruments (e.g. seismic profiling devices, telescopes, interferometers) at different places on the Moon's surface. The other major objectives will be determining what lunar resources indeed exist, proving their accessibility and verifying how they can be utilised and exploited.

Besides the geopolitical meaning and the scientific arguments above, the Moon is (visually) present in the every day life of all humans, thus providing great potential for public attention and support that is crucial for every future space programme. In contrary to other opinions, this could also accelerate the exploration of Mars, especially when we follow an integrated Moon/Mars exploration approach with taking first steps early.

5.2 Rationales for Space Stations in the Earth-Moon System

The previous section explained what reasons we have for proceeding with exploration of the Moon for a sustained development of space. To enable this endeavour and for conducting scientific experiments on and at the Moon rapidly, space-based infrastructural installations in the Earth-Moon system offer interesting opportunities and could be realised in the rather near future as a next step in an evolutionary step-by-step approach to the Moon's surface and other destinations. A space station or a man-tended space platform installed at the Moon, or rather in the Earth-Moon system could particularly serve as:

- A staging platform in a programmatic sense for verification of technologies, procedures and human factors and for demonstration of technological expertise and maturity;
- A gateway station and transfer platform, favourably with global access to the lunar surface, simultaneously providing a *safe haven* for the crewed missions;
- A research facility utilizing the specific environmental characteristics of near-interplanetary space with extreme low perturbations;
- An observation platform with an unobstructed view of Earth, the Moon, near-Earth space and all astronomical objects;
- A maintenance and servicing platform for other facilities installed in cis-lunar or surrounding space (e.g. at the Sun-Earth libration point two) or on the surface of the Moon including the far side. This could include astronomical facilities and Earth orbiting satellites (i.e. large GEO telecommunication and scientific satellites).
- A space hotel for tourism and public outreach activities.

In a sustained approach such space stations will be based on currently available or upgraded technology and would enable development and usage of common transportation vehicles (for launch, transfer and return) presumably reducing overall cost, even when the total velocity increment (delta-v), or rather propellant mass, of individual missions were not optimised.

Therefore, a space station could play a significant role in near-term space activities. Nevertheless most past and recent studies done on lunar architectures concentrate on surface bases ([Eckart1999][Koelle2003][Duke2003][CDF23A]) and only very few studies have included non-surface installations or elements such as lunar space stations or platforms. Examples are a large lunar orbiting station called *Space Operation Centre* (SOC) envisioned by KOELLE

[Koelle2002], the rotating *Clarke Station* conceptually designed in a student project [Akin2001], and a gateway-like station in a high eccentric Earth-Moon cycling orbit drafted in a student project during the ISU's summer session programme 2003 [ISU-SSP2003].

5.3 Earth-Moon System Analysis

This section presents a selection of results gained from the analysis of the Earth-Moon system regarding mission design. Beginning with section 5.3.1 an overview of the perturbation environment and its effects on the orbits of spacecraft is presented. Section 5.3.2 addresses the orbital motion of the Moon and section 5.4 discusses mission aspects, namely transfer window periods (TWP), flight time and delta-v requirements for transfer flights when navigating through cis-lunar space.

5.3.1 Perturbations in Cis-lunar Space

In low Earth orbits (LEO) a few dominating perturbations determine the evolution of the orbit and attitude of a spacecraft. With increasing distance from Earth, however, these primary perturbations are replaced by a mix of various effects existing in the Earth-Moon system, which have been quantitatively analysed and discussed earlier [Yazdi2003c].

Figure 5.2 depicts the acceleration environment spacecraft experience up to the Moon orbit (compare with literature for altitude up to GEO [Brown1998],[Montenbruck2000]). These were calculated by perturbation scanning of the Earth-Moon system with IRIS++ software. Several facts can be identified: At LEO, the deviation of the Earth gravity field from a spherical shape (expressed in the simplest form by the geo-potential coefficient J2) and the aerodynamic drag constitute the major perturbations to an ideal Kepler ellipse. While the atmospheric effects decrease rapidly with increasing altitude, J2 effects are the major disturbing source up to the geo-stationary Earth orbit (GEO). Beyond GEO, the influence of the Moon gravity takes over and at higher distances in cis-lunar space, gravity of the Sun and the solar radiation pressure gain a significant share and must not be neglected in mission design and planning. In the following sections only effects relevant for conceptual mission design will be discussed.

Effects on mission design

At LEO, mainly the J2 perturbation and the strategy for drag compensation has to be considered (i.e. permanent or periodical reboosting; [Messerschmid2000]). The main effects of the J2-term are the regression of nodes (node drift $\dot{\Omega}$), which alters coverage, ground track repetition, eclipse duration, etc.), but also the slight oscillation of the semi-major axis of about 10-11 km with orbital period. Luni-solar influences only induce very small periodic fluctuations on the orbit elements, except some minor secular alterations, which increase with greater distance but can be generally neglected.

Approximately at GEO the J2 influence equals the effect of the Moon (and beyond also the Sun) in terms of acceleration level. Perturbations to the orbital elements reach a minimum at this distance and increase beyond GEO again, now dominated by third-body effects.

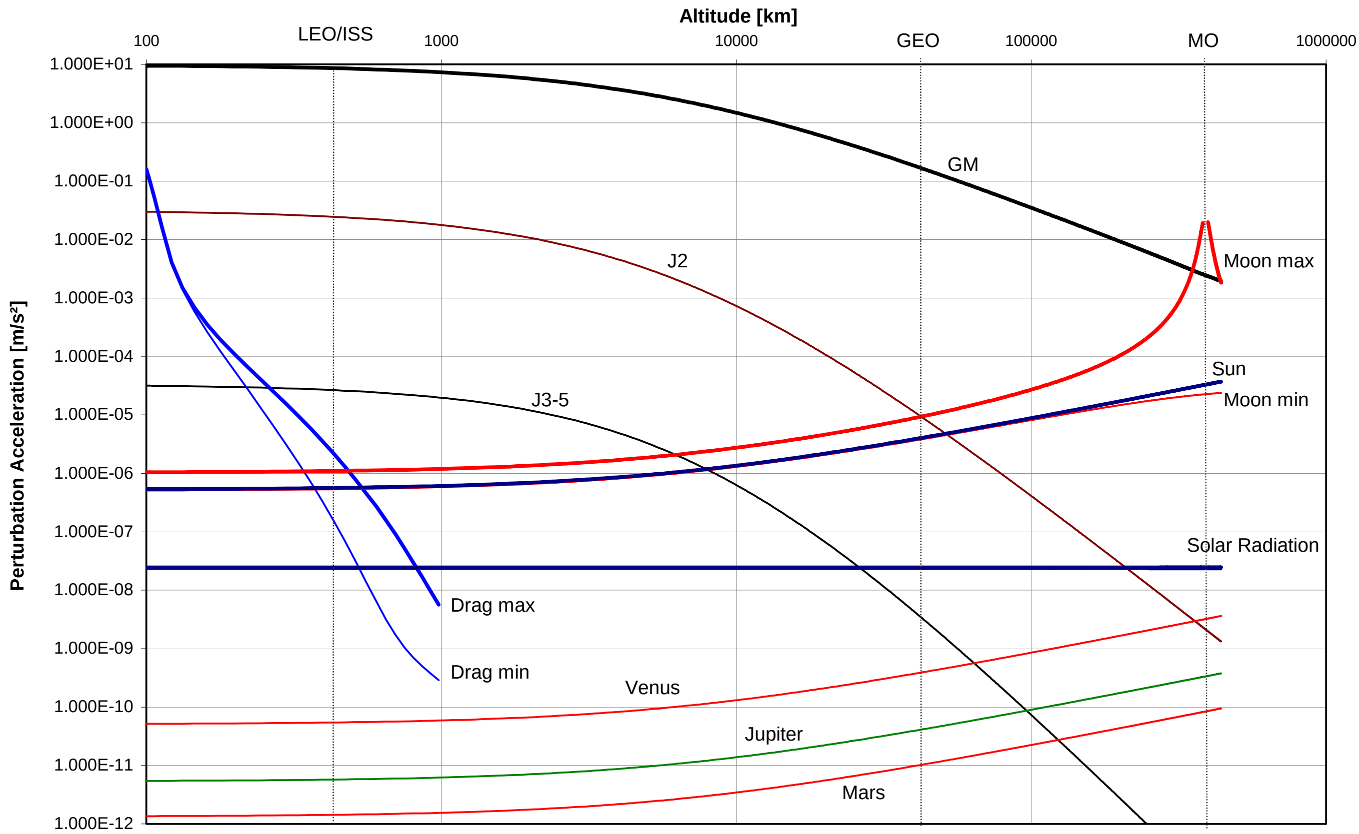


Figure 5.2: Acceleration levels of various perturbation sources for a spacecraft in a geo-centric orbit[3] (other influences i.e. albedo, tidal forces and relativity not shown)

These influences lead to an inclination change, a rotation of the line-of-nodes and a drift of the perigee. In the case of geo-stationary satellites with high-accuracy navigation requirements, the location dependent East-West drift (originated by the non-spherical geo-potential) and the North-South drift (luni-solar inclination change of about 1° per year) have to be corrected. Though inclination changes increase beyond GEO and could eventually be used by unmanned lunar transfer vehicles to save delta-v, they will not be considered here any further.

Very high Earth orbits from approximately 250000 km onwards lead to an orbital track that from time to time brings the spacecraft close to the Moon so that its gravity pull can cause accelerations or decelerations with severe changes to the spacecraft geo-centric orbit. Especially the perigee altitude and inclination can be altered. Ballistic trajectories often show chaotic behaviour and have a good chance impacting on either the Earth or Moon. Due to this, in general, analytic rules-of-thumb cannot be applied in this region. This means that transfer missions which stay for long period of time beyond LEO (i.e. low-thrust missions) and/or orbits with low velocities at apogee in high cis-lunar space (e.g. direct transfer to LL1 or stationary high-eccentric orbits) must be carefully designed by taking these perturbations into account.

Table 5.3 summarises the influences of the different perturbation sources for Earth and Moon-bound orbits. The elements are ordered by the strength of the influencing effect.

[3] Assuming a mass-to-effective-area ratio of 230 kg/m²

Table 5.3: Effects of various perturbations on the orbital elements in the Earth-Moon system

Perturbation	LEO	MEO/GEO	HEO	LLO
Residual Atmosphere	Δa, Δe, (Δi)	-	-	-
Planets non-spherical shape	$\Delta\Omega$, $\Delta\omega$, Δe	$\Delta\Omega$, $\Delta\omega$	-	Δe, $\Delta\Omega$, $\Delta\omega$ Δi
3rd-body influences	-	$\Delta\Omega$, $\Delta\omega$, Δi	$\Delta\Omega$, $\Delta\omega$, Δe, Δi, Δa	(Δi)
Magnetic influences	(Tether)	-	-	-
Solar radiation pressure	-	Δe	Δe	Δe

Here the aerodynamic forces can only lead to an inclination change if the spacecraft shape has an effective surface area with its normal vector pointing off the atmospheric incidence direction. In this case, the spacecraft surface produces a lift force that could be used for atmospheric orbit transfer vehicles that change their orbit inclination aero-assisted by temporarily entering upper atmospheric layers.

At HEO, spacecraft move in a rather low-acceleration environment most of the time, where the gravitation of the Earth, Moon and Sun dominate orbit evolution. Thus, individual trajectories can differ much, depending on the time and date of the mission and the resulting individual orbital track relative to these bodies.

Somewhat similar are missions utilizing one of the Moon-synchronous Lagrange points. These libration points are located stationary in the rotating Earth-Moon system (see section 5.4.1). They are embedded per definitio in a low-acceleration region with the Moon and Sun perturbations dominating. In this connection, the eccentricity and irregularity of the orbital motion of the Moon (section 5.3.2) and other third-body influences, especially the Sun, are crucial to investigate the spacecraft orbital stability and station-keeping efforts.

Finally, Moon-bound orbits such as low lunar orbits (LLO) are mainly influenced by the non-spherical shape of the Moon, or rather the selenopotential field, and with increasing lunar altitude also considerably by the Earth and Sun gravitation. Long-term analyses often show decay of the pericentre, ultimately leading to surface impacts after ballistic orbital lifetimes ranging from over 1 year down to 40 days from an initial 100 km perilunar altitude [Meyer94]. These calculated lifetimes strongly depend on the initial orbital state, such as inclination, eccentricity, RAAN and argument of pericentre. Due to the lack of accurate data regarding the Moon's gravitational field, today long term predictions of lunar orbital evolutions are rather unreliable. However, one may conclude that lunar orbits are heavily perturbed and an elaborated control strategy will certainly be necessary for station keeping.

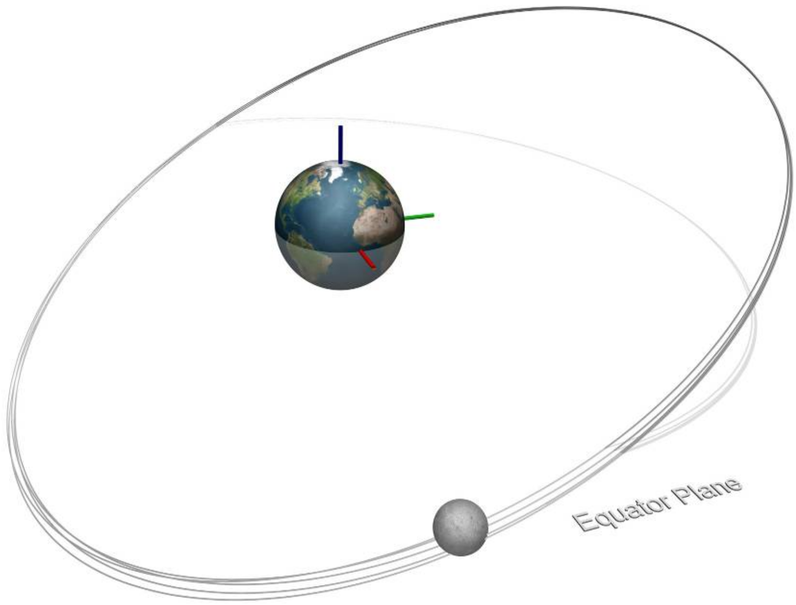


Figure 5.3: The Moon's orbital motion around the Earth
(sample trail during 4.5 months; inertial geo-centric frame; PCEQ)

5.3.2 The Moon's Orbital Motion

The Moon orbits the Earth in 27.317 days at a mean distance of 384400 km with an orbital eccentricity of 0.0549, thus the distance from Earth varies from 363300 km to 405500 km. Furthermore, the orbit is not inertial in space but highly influenced by the effects discussed before. Therefore, the orbital elements vary strongly more or less periodically making both the orbital motion models of the Moon and the Earth-to-Moon transfer a challenging task. Figure 5.3 depicts a plot of the Moon orbit obtained with IRIS++ and visualised with the COMET orbit data visualisation module. It demonstrates the Moon trajectory over approximately 4 months and shows a visible orbital variance even from one revolution to the next. Long-term analysis reveals that the ascending node travels around the Earth once in 18.6 years yielding to a fluctuation of the Earth-equatorial inclination from 18.3° to 28.6° with this period [Bate1971].

5.4 Cis-lunar Navigation

For navigating between locations in cis-lunar space, a large range of influencing factors must be taken into account. Especially for the calculation and optimisation of low-thrust trajectories, e.g. when using solar-electric propulsion (SEP) or solar-sails, very accurate environmental models are a prerequisite for trajectory design and simulation. These models have been presented in chapter 4.4. This section will firstly identify specific locations within the Earth-Moon system representing typical destinations and departure locations, and then addresses the transfer requirements and constraints.

5.4.1 Lunar Locations and their Properties

In this report all typical destination locations of lunar missions are referred to as "lunar location" or locations "at the Moon". This incorporates not only the actual lunar surface or a low lunar orbit (LLO), but also other places in lunar or lunar orbit vicinity that could play a role in the future. Summarised these are:

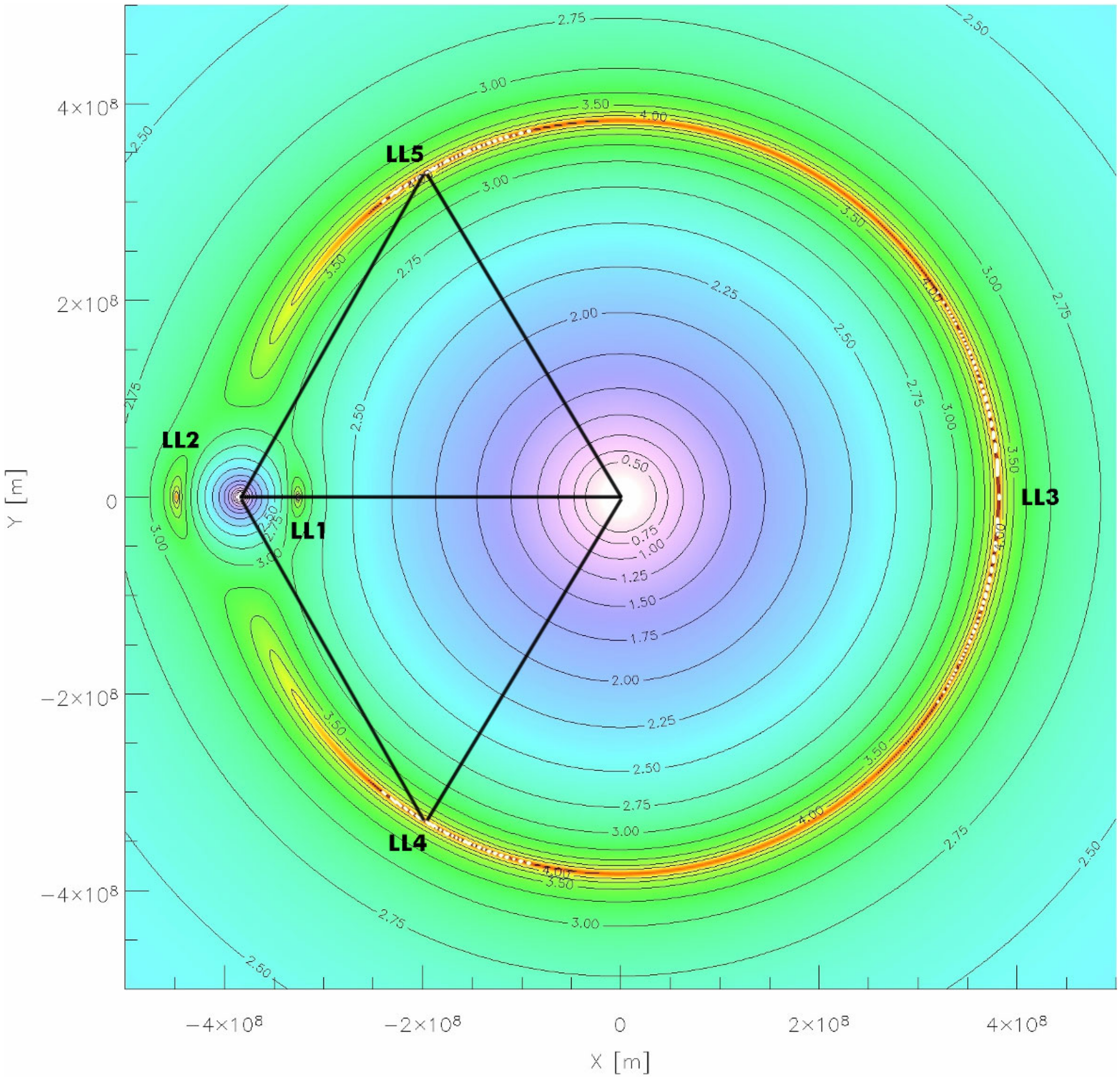


Figure 5.4: Acceleration level in the Earth-Moon system plane due to Earth and Moon gravitation (rotating frame; values given in [10^{-z} m/s²])

- Lunar Surface Base (LSB) locations
- Low Lunar Orbit (LLO)
- Moon-synchronous Lunar Lagrange Points (LLP)
- Other Moon-periodic orbits, such as High Earth Orbits (HEO) and cis-lunar/trans-lunar High Eccentric Earth Orbits (HEEO), including cycler orbits [Uphoff1993].

Lunar Lagrange Points (LLP) in the Earth-Moon System

In the following, the focus will be on LLPs. In order to visualise LLP locations Figure 5.4 shows instantaneous gravitational acceleration levels (z value) of the Earth and Moon acting on a spacecraft standing still in the rotating PCOF frame (synchronous with the Moon). The data was obtained with IRIS++ in the perturbation scanner mode and shows that in the vicinity of the Moon orbit regions of very low acceleration levels can be found, where rather low amount of delta-v is sufficient for station-keeping ([Hoffman1993],[Euler1971]) and departure to other destinations ([Farquhar1969], [Bond1991]).

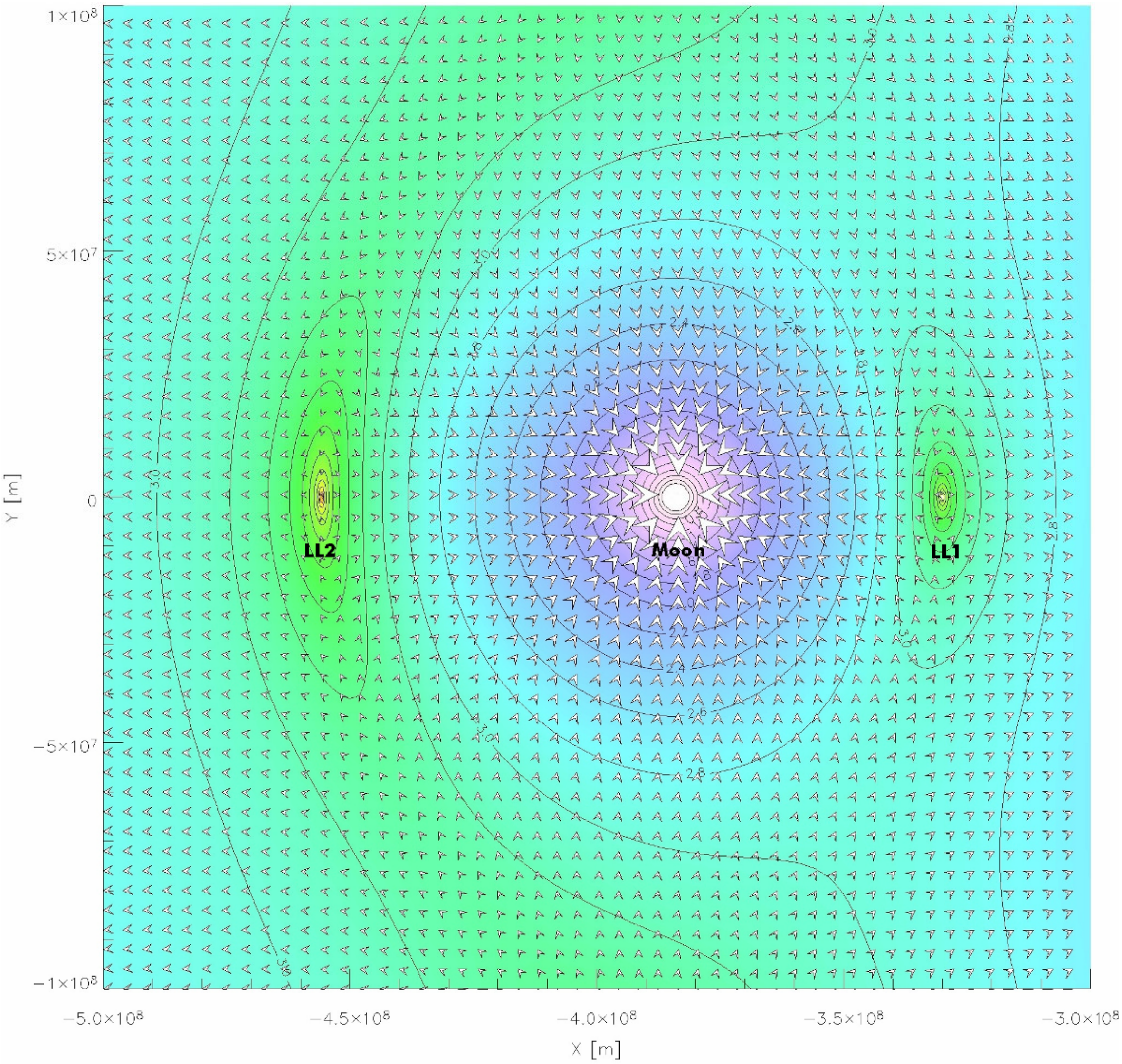


Figure 5.5: Acceleration environment around the Moon including LL1 and LL2 (Moon rotating frame; values given in [10^{-z} m/s²])

In fact, five individual regions lying on the lunar orbital plane (LOP) named libration or *Lagrange Points* are identified in the circular restricted three-body problem:

- LL1 is the cis-lunar (near-side) LLP lying between the Earth and Moon at 83.69% of the Earth-Moon distance (321710 km mean distance from Earth; 62690 km from Moon).
- LL2 is the translunar (far-side) LLP lying behind the Moon at 115.57% of Earth-Moon distance (mean distance of 59743 km from Moon).
- LL3 is the trans-Earth LLP lying behind the Earth at 100.51% of Earth-Moon distance (mean distance of 386345 km from Earth), and the

LL4 and LL5 are the librations points leading respectively trailing the Moon on its orbit at 60° angular distance to the Moon seen from Earth (384400 km mean distance from both).

The collinear points LL1 to LL3 are *semi-stable* points, namely stable in both transversal directions but unstable radially. Thus, an infinitesimal deviation of the position of an uncontrolled (ballistic) spacecraft from these points in the transversal directions leads to acceleration back towards these points resulting in an oscillation known as *Lissajous orbit*.

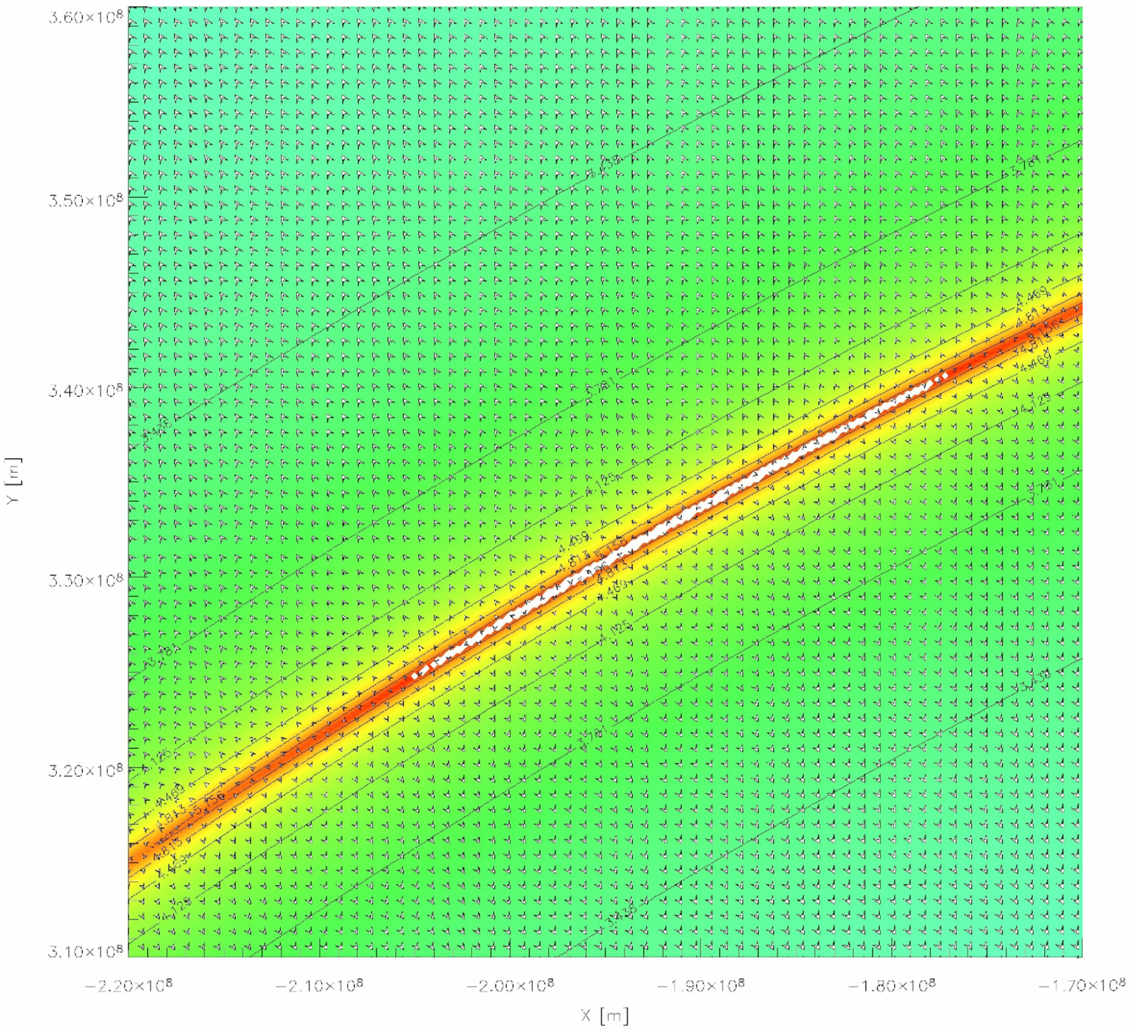


Figure 5.6: Acceleration environment around the Lunar Libration Point LL5 (Moon rotating frame; values given in [10^{-z} m/s²])

In contrast to this, a radial deviation leads to acceleration towards either the Moon or the Earth respectively, resulting in an unstable motion with respect to these points. This "saddle-point" character can be seen in Figure 5.5 depicting the acceleration environment of LL1 and LL2 in the lunar orbital plane.

Regarding the equilateral-triangle points LL4 and LL5, literature often irritatingly state that these points are stable. Indeed, the Earth-Moon system, namely the Earth-Moon mass ratio, fulfils the analytical stability criterion defined in the restricted three-body problem [Farquhar1970].

In this context it is correct to state that spacecraft would enter stable orbits around these points if only the Earth and Moon gravity were present. For example Figure 5.6 depicts the acceleration environment around LL5, showing the acceleration vectors directing away from the LL5 region. Because of the *Coriolis* forces a spacecraft radially accelerated towards Earth would also accelerate in a tangential direction with increasing the geo-centric orbit and vice-versa, thus leading to a periodic orbital motion around the LLPs (*halo orbit*). In reality, however, other influences such as the Moon orbital eccentricity and the Sun's gravity potential are present, allowing an uncontrolled spacecraft to escape from these LLP-orbits after a number of revolutions. Thus, in any case an orbit control approach is certainly necessary. FARQUHAR

[Farquhar1970] and others (e.g. [Hoffman1993],[Euler1971]) addressed the station-keeping effort at the collinear points. It turned out that a simple control strategy is sufficient in order to maintain a stable orbit with a delta-v requirement of 0.5 to 36 m/s per year, thus in the same magnitude of the station-keeping delta-v of geostationary satellites.

5.4.2 Earth Orbits for Staging (Departure Locations)

Assuming a target space station location "at the Moon" as defined above, several orbits and trajectory types principally come into question in support of parking and staging. To benefit from the existing infrastructure, currently used orbits and derived orbits are the preferred choices:

- Low Earth orbits (LEO) ranging from 300 to 500 km altitude and 0° to 60° inclination of nearly circular shape
- Geo-stationary Earth orbits (GEO), circular with a 42164 km radius, coplanar with Earth's equatorial plane or inclined up to 30°
- GEO transfer orbit (GTO), eccentric orbit with an eccentricity of about 0.72, inclinations from 0° to 30° and perigee altitudes of about 350 to 600 km

At first glance, the following orbit properties could additionally be of special interest due to their outstanding theoretical benefits:

- Equatorial and near-equatorial orbits impose payload efficient launch possibilities
- Coplanar orbits with the lunar orbital plane (LOP) would allow permanent, low-cost and safe departure and arrival for in/outbound lunar transfers.
- Moon-synchronous orbits would offer similar benefits for departure or arrival, saving transfer delta-v by permanently avoiding the necessity of plane changes.

This list leads to a range of various orbits considered in this investigation (Table 5.4). Other orbits and trajectories for transfer and/or space station location, e.g. high eccentric Earth orbits (HEEO) and cycling orbits, shall not be discussed further.

Table 5.4: Types of LEO-based parking/staging orbits

Orbit	Altitude [km]	Inclination [°]	Eccentricity
LEO/ISS	350 – 420	51.6	~0
LEO/STS	300 – 400	28.5	~0
LEO/LOP	300 – 400	18.3- 28.6	~0
LEO/A5	300 – 500	7	~0
LEO/E	300 – 500	0	~0
GTO/A5	350 – 35800	7	~0.72
GTO/LOP	350 – 35800	18.3 - 28.6	~0.72
GEO/E	350 – 35800	0	0
GEO/LOP	35800	18.3 - 28.6	~0

5.4.3 Transfer Window Periods (TWP)

TWPs shall be defined as the time period between two successive events when departure to the Moon is possible with an impulsive transfer injection manoeuvre tangentially to the spacecraft's current orbital plane. Such an event is defined with Moon's zero orbital declination to the spacecraft's orbit at the time of arrival (Figure 5.7). Otherwise an extra delta-v necessary to change the inclination equal to this will-be orbital declination of the Moon at arrival must be added. Note that the spacecraft does not have to travel within the lunar orbital plane (LOP).

The questions this section deals with are, when, how long and how often these transfer windows open. To answer this, attention is necessary because neither the spacecraft nor the Moon orbits are inertial in space (see also section 5.3.2). The Moon travels with approximately 13.2° per day around Earth and completes one revolution in 27.3 days. Besides, the orbital plane of all (save equatorial) Earth orbits rotate as well with one revolution of their RAAN in 40 to 70 days, depending on their inclination and altitude. These two motions primarily dictate Moon's orbital declination profile, its amplitude, period and length of the associated transfer windows.

Assuming circular and equatorial orbits, lunar transfer windows open every 13.6 days on average[4] (Figure 5.8) corresponds to half of the Moon's sidereal period of 27.3 days. As this figure also shows, circular inclined orbits, if not altered, would principally have the same mean departure window periods. In reality, however, orbits are perturbed and the RAAN difference between departure and LOP is not constant but varies with the RAAN drift rate $\dot{\Omega}$ which is a function of inclination i and Earth distance (semi-major axis a and eccentricity e). Results of drift calculations show none of the considered orbits offer a RAAN drift rate equal to the mean motion of the Moon. Because this orbit could be *synchronous* with the Moon, it would therefore offer a permanent departure or arrival opportunity[5].

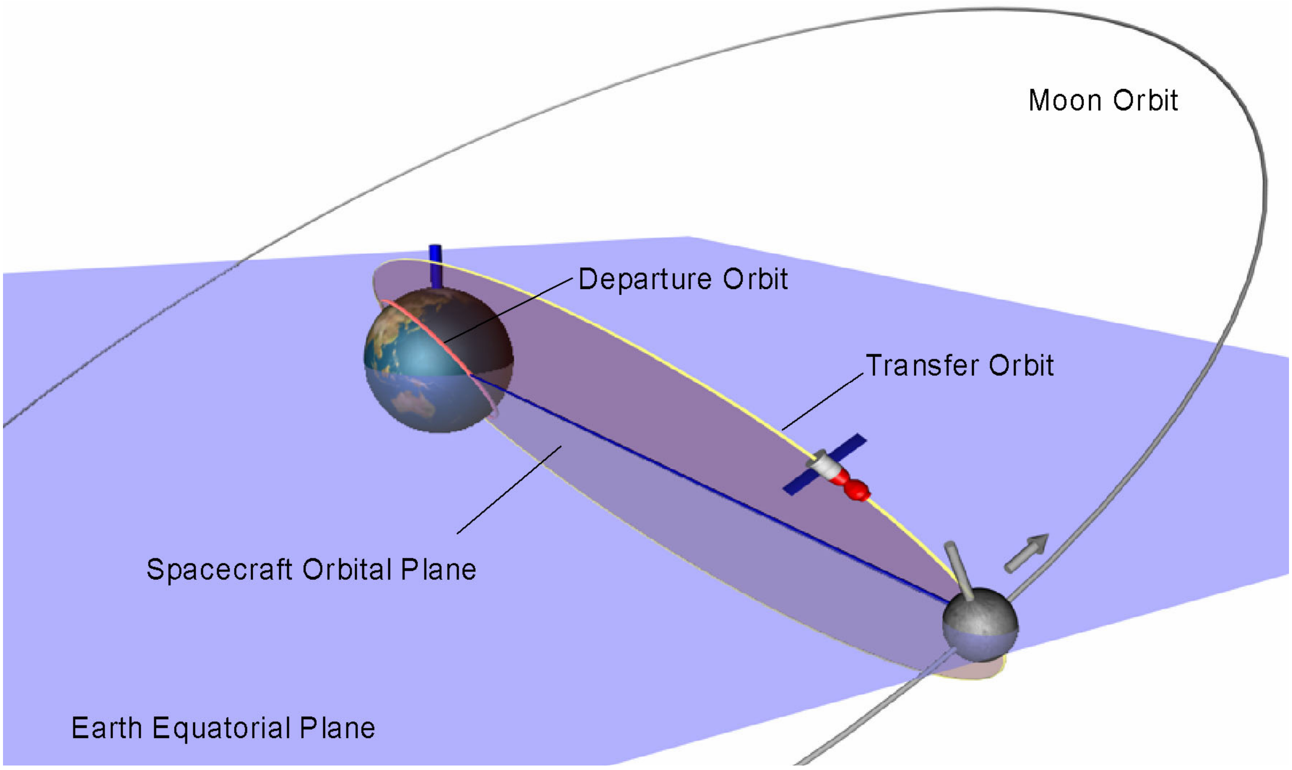


Figure 5.7: Open transfer window conditions at Moon arrival

[4] The period is alternately 12.9 and 14.4 days due to the Moon's orbital eccentricity.

[5] This orbit would not be coplanar with LOP, but would direct to a target point on the lunar orbital track, where a transfer vehicle would reach the Moon after the transfer time.

Details may be found in a previous investigation [Yazdi2003c]. With the mean motion of the Moon $n_{Moon} = 13.2 \,{}^{\circ}\!/_{d}$ one can roughly estimate the mean TWP-length to be

(70) $$\overline{P}_{\delta=0^\circ} = \frac{180^\circ}{n_{Moon} - \dot{\Omega}}$$

with $\overline{P}_{\delta=0^\circ}$: mean length of the transfer window period (TWP)

n_{Moon} : mean orbital motion of the Moon

$\dot{\Omega}$: mean RAAN drift rate of the spacecraft's orbit

The real time between two consecutive passes of the Moon through the departure plane deviates from this value considerably due to the drifting RAAN that involves a fluctuation of the *effective inclination* between the departure plane and LOP. The effective inclination is the angle between both planes and is a function of time as it depends on the current RAAN difference of both planes. Its value can be calculated with the scalar product of the normal vectors of both planes and is a sinus-curve for constant RAAN drift rates.

The effective inclination defines the amplitude $\hat{\delta}$ of the oscillation of orbital declination and, therefore, is also a function of time. For a given unperturbed constellation with a departure orbit inclination i, the amplitude is constant (as it is in Figure 5.8) and the orbital declination δ oscillates steadily between two possible extremes $\hat{\delta}_{\min}$ and $\hat{\delta}_{\max}$ (Figure 5.8-b min and max). These extreme amplitudes, or rather effective inclinations, are defined by the (equatorial) inclination of both orbits:

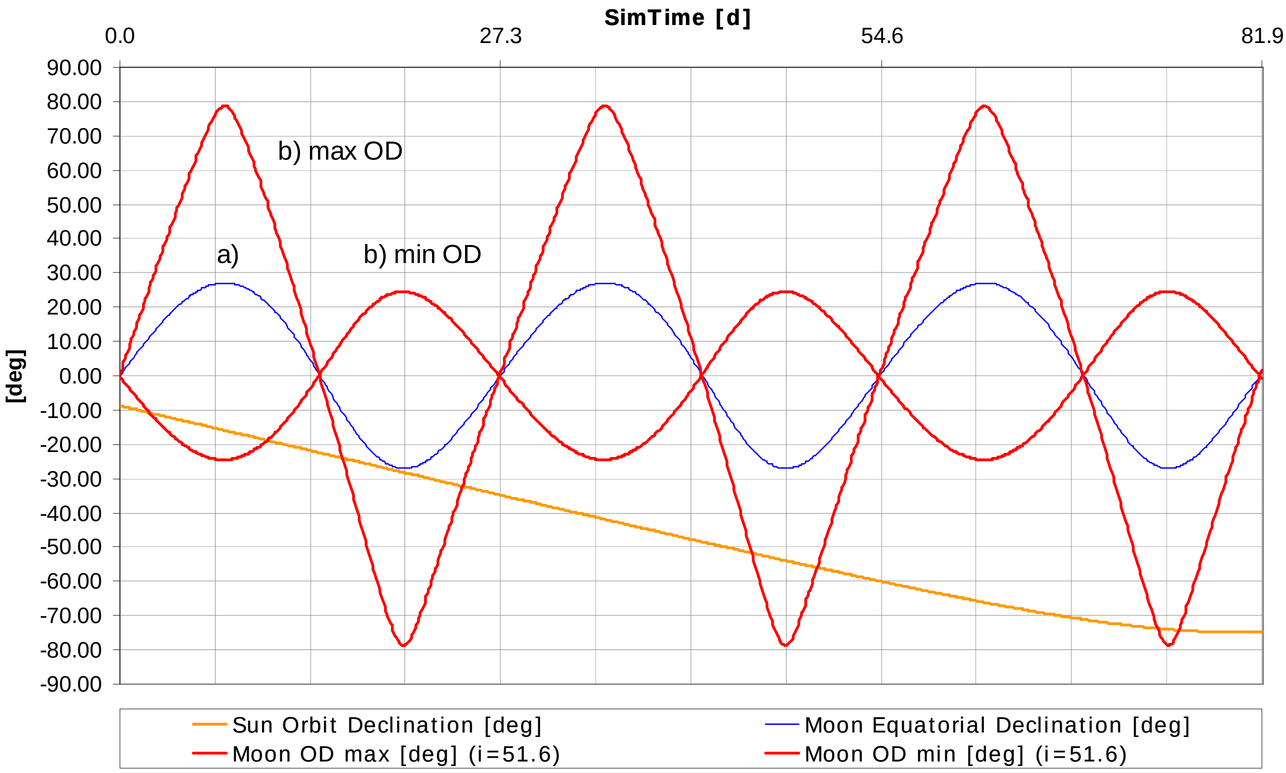


Figure 5.8: Orbital declinations of the Moon to unperturbed circular Earth orbits: a) i =0°; b) i =51.6°

(71) $\hat{\delta}_{\min} = |i_{Moon} - i|$

(72) $\hat{\delta}_{\max} = |i_{Moon} + i|$

(73) $\hat{\delta} = \begin{cases} \hat{\delta}_{\min} : \text{if } \Delta\Omega = k \cdot 360° \\ \hat{\delta}_{\max} : \text{if } \Delta\Omega = 180° + k \cdot 360° \end{cases}$ with $k=0,1,2,\ldots$

with
$\hat{\delta}$: oscillation amplitude of the Moon's orbital declination regarding the spacecraft's orbit
$\hat{\delta}_{\max}$, $\hat{\delta}_{\min}$: max./min. amplitude of the Moon's orbital declination
i_{Moon} : the Moon's orbital inclination
i : the spacecraft's orbital inclination
$\Delta\Omega$: angular distance of the RAAN of the orbits of the Moon and the spacecraft

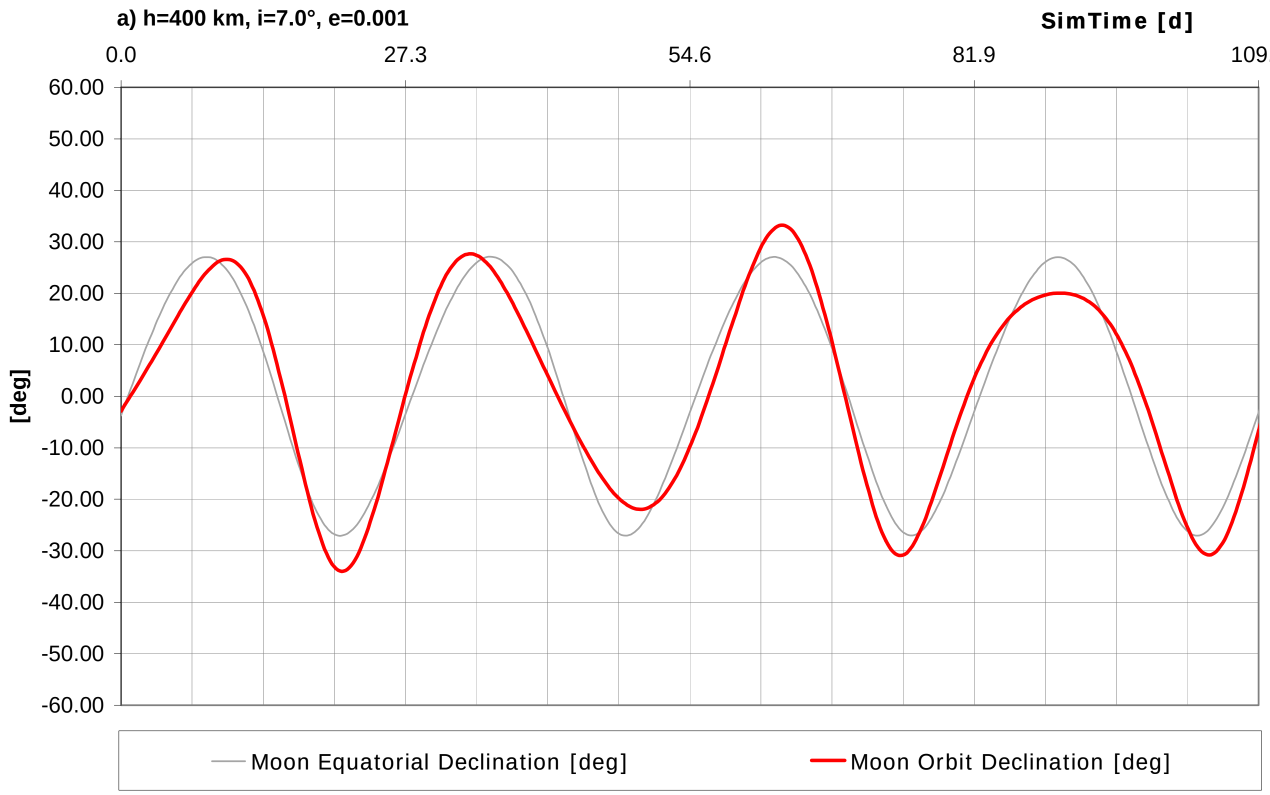
The *ecliptical* inclination of the Moon i^{ec}_{Moon} is 5.15° but due to its RAAN drift its *equatorial* inclination i_{Moon} varies between 18.3° and 28.6°. Taking i_{Moon} =27.2° leads to the extremes of curve b) in Figure 5.8. Note that because the equator of the Moon has a low ecliptic inclination, the effective inclination between both planes specifies approximately the inclination of the spacecraft orbit if entered into a lunar orbit after transfer.

The charts of Figure 5.9 demonstrate the evolution of the orbital declination of the Moon in respect to various perturbed orbits over a period of four lunar revolutions. Case a) shows a 7° inclined LEO, dubbed LEO/A5, featuring nearly equally spaced zero points and therefore are very similar to equatorial (i.e. LEO/E and GEO/E) or unperturbed orbits.

Case b) is LEO oriented in the lunar orbit plane (LOP) and is representative for all orbits with inclinations near the lunar inclination (e.g. LEO/STS). Typically these orbits allow especially slow Moon passes approximately every 50 days, which mission planners can benefit from.

One can also see that orbits with a higher inclination than the Moon introduce additional passages in short distances where declination rates increase with orbit inclination. This is visible in case c) of Figure 5.9 depicting the ISS orbit. Due to the relatively high inclinations, high orbital declinations are obtained with high declination rates at the passes.

The last charts d) and e) of Figure 5.9 depict two cases: one is a GTO-derived orbit and the other is a GEO, both oriented in LOP initially. Although slow, the angular rates of the GTO orbit are sufficient to build up reasonable amplitudes (ultimately reaching the level of case b) while GEO amplitudes stay almost flat, allowing permanent access for coplanar transfers. Nevertheless, GEO will ultimately rotate out of the LOP as well; so in the long run a control strategy is necessary.



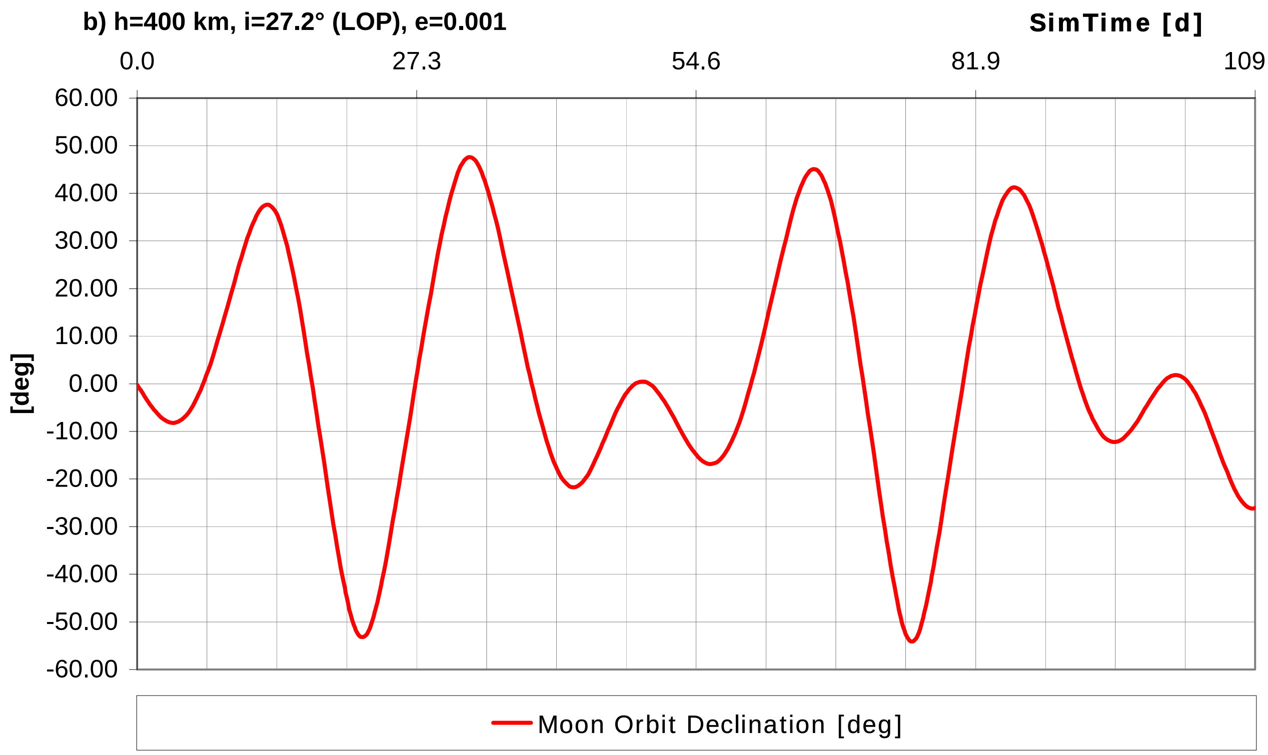
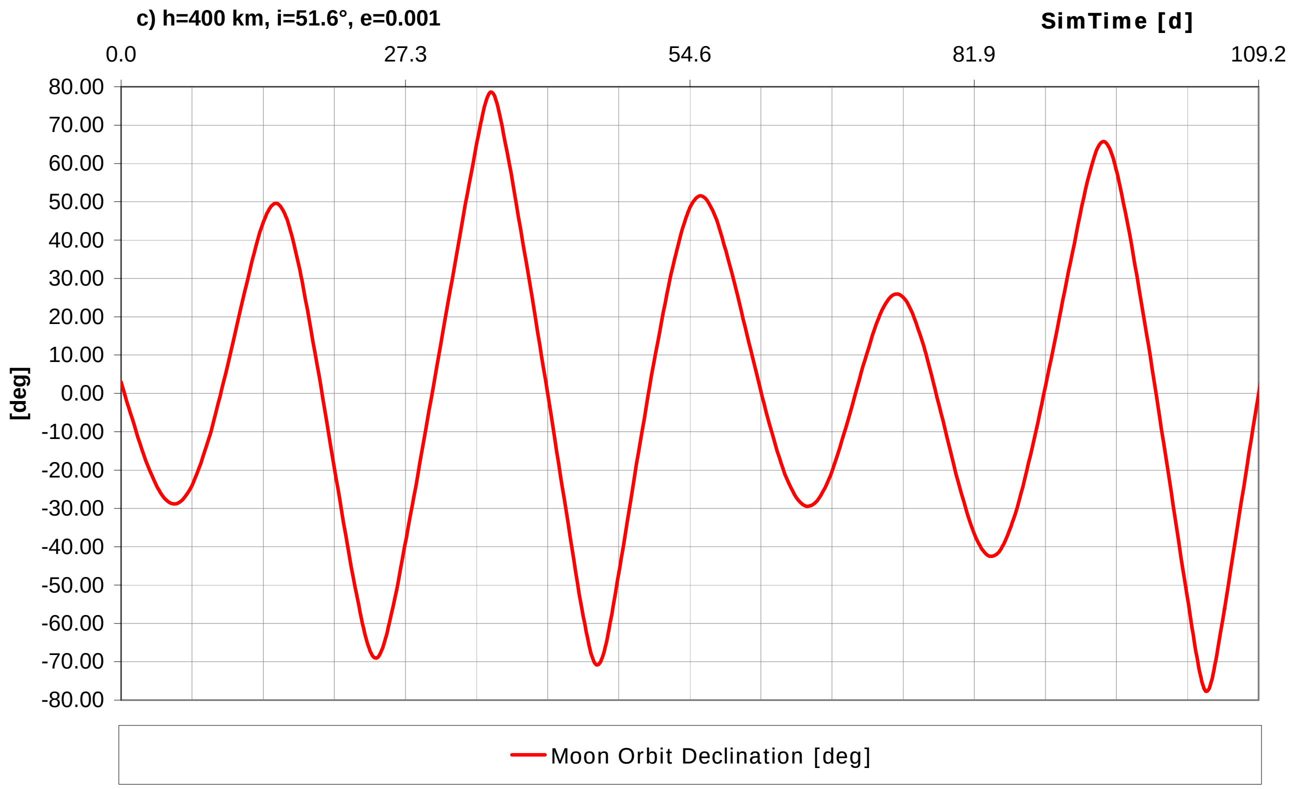


Figure 5.9-a/b: Orbit declination profiles of selected orbits:
a) LEO/A5, b) LEO/LOP



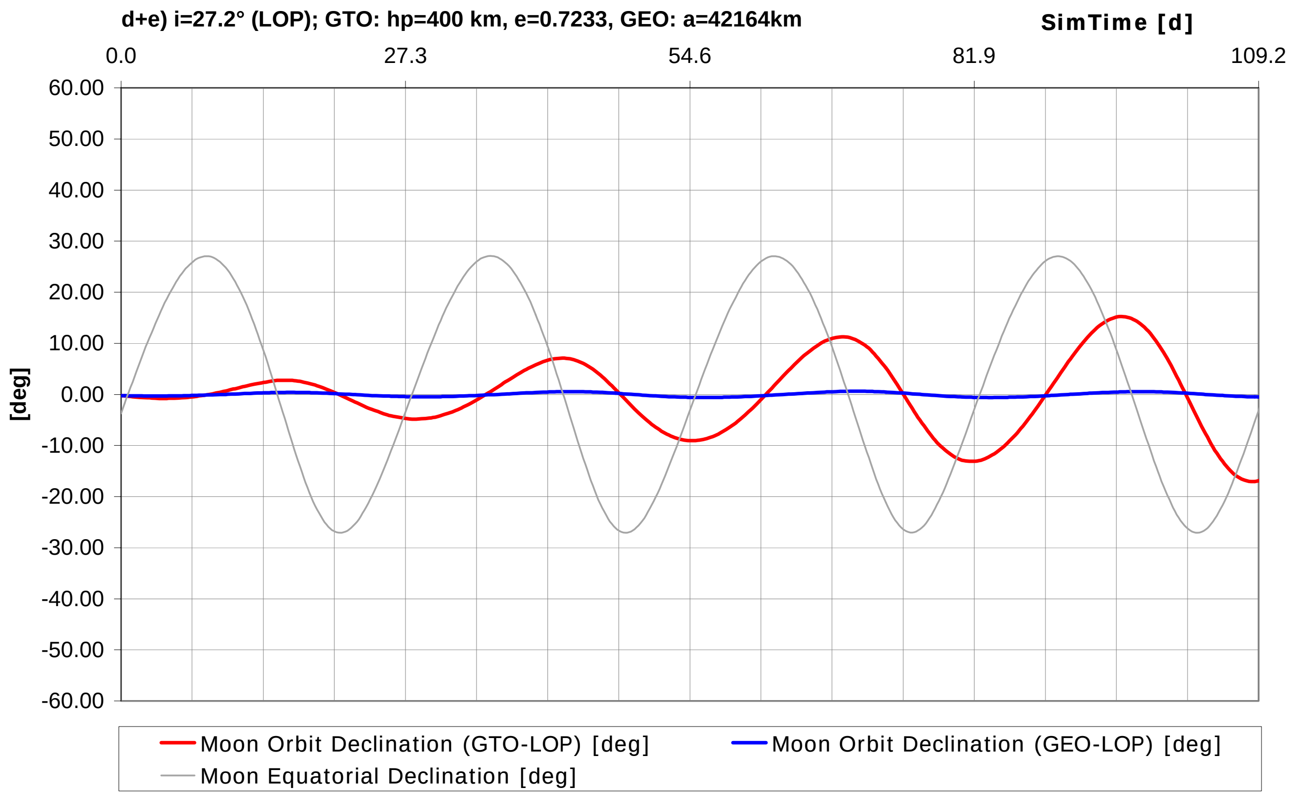


Figure 5.9-c/d+e: Orbit declination profiles of selected orbits:
c) LEO/ISS, d+e) GEO/LOP and GTO/LOP

For the departure and arrival transfer window periods (TWP) one can conclude: low inclined, low altitude and low eccentric orbits lead to the fastest regression of nodes and shortest TWPs. But the orbit declination amplitude is relevant as well. Because an impulsive transfer injection manoeuvre is favourable at a very small section along the orbital track, only a specific selection of orbits is available for every transfer window. Missing the opportunity at optimum one must wait one orbit revolution. Now, the higher the orbit declination amplitudes are, the steeper and faster the Moon passes through the plane of the departure or arrival orbit. This means the transfer windows tighten at high declination amplitudes. Therefore, the range of orbit revolutions to be selected decreases, and the delta-v margin must be increased, respectively.

5.4.4 Delta-v

Delta-v is the major measuring unit within the orbit mechanics and orbital navigation discipline. It describes the velocity increment that a spacecraft propulsion system must provide to change its orbit or trajectory in order to reach one location from another. Table 5.5 summarises the delta-v requirements for transfer missions between selected locations in the Earth-Moon System. Calculated delta-v values are obtained by an analytical patched-conic approach (PCA). In this connection, a transfer mission is simulated as a series of restricted two-body problems and the trajectory is patched by switching the central body at the border of the sphere of influence (SOI) of the Moon. Individual manoeuvres are usually performed at the peri/apocentres and at entering/leaving SOI. Returns to LEO trajectories assume a circularisation Δv of 0.12 km/s, and a return to Earth assumes deorbit to 50 km pericentre altitude.

5.4.5 Summary of Mission Parameters of Parking and Staging Orbits

The aerodynamic drag and the regression of nodes are the most important influences for Earth-bound orbits to be used as location for a support terminal for staging to lunar missions. Concerning drag, which especially influences LEOs, a trade-off is required depending on the duration of the stay: Low initial altitudes would benefit from high possible payloads but suffer from increased station-keeping efforts. For long duration missions, an altitude similar to ISS orbits or above would likely be chosen leading to a reasonable compromise.

Calculated orbit properties and mission parameters for outbound lunar transfers are summarised in Table 5.6. Total delta-v depends primarily on the injection burn and therefore on initial orbital altitude and pre-injection velocity. This is seen by the values for the transfer to LLO. Therefore, GTOs show most favourable delta-v's due to their high perigee velocity, followed by the GEO due to its high altitude. Thus, GTO orbits mark the lowest possible delta-v transfer, even if not coplanar with LOP, and from this point of view alone, GTO would make a good departure location. Indeed, this may benefit small probes launched as secondary payload to GTO anyway (e.g. with Ariane 5). But these orbits will unlikely be utilised for assembly, logistics or even crewed missions: Firstly, the location of perigee with respect to the Moon, which is crucial for a successful injection manoeuvre, is not necessarily in the right direction. Secondly, space station modules to be transported and the transfer vehicle must be launched first to the GTO and this, compared to a launch to LEO, reduces the launcher payload mass more than the GTO-departure saves. The latter is basically true for GEO orbits as well.

Table 5.5: Delta-v requirements between selected locations (Δv in [km/s], to be read from left to right/upwards; values in brackets state the total transfer time in days; i_{Moon}=27.2°; outbound transfers to LLO assume free inclination; if not stated else: LEO: e=0.001, h_p=400km; *[Boden2000a], **[Broucke1979])

From/To	Earth Surface	LEO/E' 300 km	LEO/LOP	LEO/ISS	LL4/5	LL1	LLO 100 km	LSB	DSE
LEO/A5	0.1	0.99	2.70	5.97	4-4.04 (4.6)	3.8–3.84 (3.8)	4.2 (5.4)	5.93	3.2
LEO/E' 300 km	0.07	-	3.7	6.96	4.03 (4.7)	3.82 (3.8)	4.2 (5.3)	5.93*	3.22*
LEO/LOP	0.1	3.7	-	3.27	3.92-4.1 (4.5)	3.71–3.9 (3.8)	4.2 (5.4)	5.93	3.2
LEO/ISS	0.1	6.96	3.27	-	4-4.12 (4.6)	3.8–3.93 (3.8)	4.2 (5.4)	5.93	3.2
LL4/5	0.86* (4.6)	0.98* (4.7)	0.98 (4.7)	1.00 (4.7)	-	0.33* (15/10)	0.98* (15/10)	2.58* (7)**	0.43*
LL1	0.65 (3.9)	0.73 (3.9)	0.75 (3.9)	0.75 (3.9)	0.33*	-	0,57** (1.0)	2.52*	0.14*
LLO 100 km	1.12 (5.5)	1.19 (5.5)	1.22 (5.5)	1.22 (5.5)	0.98*	0.64*	-	1.87*	1.40*
LSB	2.62*	2.74*	2.66	2.66	2.58*	2.52*	1.87*	-	2.80*

Transfers from the various LEO to LLO have almost the same delta-v of 4.04 km/s [Boden2000b] to 4.2 km/s. Here the difference lies in the length and the properties of the transfer windows (see above). For a transfer to other destinations, the inclination of the departure orbit has a noticeable influence on the delta-v. To LL1, departure orbits in or near the LOP benefit from this, but the advantage of about 0.1 km/s is only usable once every 50 to 55 days due to the high RAAN drift rate in LEO. Higher inclinations lead to higher delta-v fluctuations for transfers to LL4/5.

To conclude, all considered circular LEO orbits offer frequent arrival and departure windows for in- and outbound lunar transfers. Transfer window sizes vary, but generally orbits with inclinations similar to the lunar orbit offer slow Moon passes and therefore more injection opportunities per transfer window.

An interesting orbit permanently coplanar with LOP could be achieved at GEO (and HEO) distances. These orbits have a relatively small RAAN drift by nature and regular adjustments to LOP seem feasible with presumably low orbit maintenance delta-v. The GEO may be an interesting option for a staging point if the mission scenario does include utilization objectives involving GEO-operations. Otherwise, a terminal at that distance seems, like in GTO, unlikely due to the payload reduction with launches to GEO.

Though benefits exist of a LEO with a similar inclination as the Moon, judging from these results, there is no major disadvantage for the utilization of the existing ISS infrastructure as a staging terminal.

Table 5.6: Properties of selected Earth orbits (Δv in km/s, total transfer time T in days; calculations start with node alignment; i_{Moon}=27.2°; LLO in 100 km altitude with free inclination; if not stated else: e=0.001 and perigee altitude 400km; GTO: e=0.7233; GEO: e=0.0, a=42164km; *) orbit is initially coplanar with LOP; $^{+}$) perigee location at optimum assumed; **) plane is assumed to be controlled at LOP)

Orbit	Remarks	$\Delta\Omega$ [°/d]	TWP (mean) [d]	Δv (T) LLO	Δv (T) LL1	Δv_{min}-Δv_{max} (T) LL4/5
LEO/ISS	i=51.6°	-5.3	7.7-11.3 (9.9)	4.2 (5.4)	3.81–3.93 (3.8)	4.00–4.12 (4.6)
LEO/SSF	472 km i=31.6°	-6.6	4.7-11.3 (9.1)	4.18 (5.4)	3.71–3.9 (3.8)	3.92-4.08 (4.6)
LEO/STS	i=28.5°	-7.1	1.8-11.3 (8.9)	4.2 (5.4)	3.72-3.9 (3.8)	3.93–4.09 (4.6)
LEO/LOP*	i=27.2° Ω=8.32°	-7.2	1.1-12.1 (8.8)	4.2 (5.4)	3.71–3.9 (3.8)	3.92–4.10 (4.5)
LEO/A5	i=7.0°	-8.02	11.3-16.8 (13.6)	4.2 (5.4)	3.79–3.84 (3.8)	3.99–4.04 (4.6)
LEO/E	i=0.0°	-	12.9/14.4 (13.6)	4.2 (5.3)	3.82 (3.8)	4.00 (4.8)
LEO/E'	300 km, i=0.0°	-	12.8/14.6 (13.6)	4.2 (5.3)	3.82 (3.8)	4.03 (4.7)
GTO/A5^{+}	i=7.0°	-0.40	12.4-14.9 (13.6)	1.8 (5.3)	1.39–1.44 (3.9)	1.57–1.63 (4.8)
GTO/LOP*$^{+}$	i=27.2°, Ω=8.32°	-0.355	12.4-14.2 (13.4)	1.8 (5.3)	1.31 (-1.5) (3.9)	1.52–1.68 (4.8)
GEO/E	i=0.0°	-	12.9/14.4 (13.6)	1.90 (5.7)	1.6 (4.5)	1.86 (5.0)
GEO/LOP	i=27.2°, Ω=8.32°	-0.012	(arbitrary)**	1.90 (5.7)	1.36 (4.5)	1.64 (5.0)

5.5 Lunar Space Station Scenario Example

As an example for the demonstration of the application of the upgraded SSDW, the following fictitious mission statement is given including objectives, constraints and requirements for a lunar space station scenario.

5.5.1 Mission Statement

Background

Situation: in March 2005, with the US Space Shuttle return to flight ahead in a couple of months, the assembly of the International Space Station (ISS) is resumed and is assumed to be well underway with completion around 2010. Exploration plans for the post-ISS-assembly-complete era call for concepts for the next steps of space exploration including human missions to the Moon and Mars. ESA and Roskosmos, jointly ask the virtual "European Manned Spaceflight Industries (EMSI)" company, including Russian industries, to conceptualise a scenario with the following objectives.

Task Description

A study project shall be performed for the creation and operation of a human space infrastructure beyond LEO in a European-Russian led cooperation, with integration of other possibly interested partners (MS1).

The space station architecture shall primarily enable the partners to prepare and support successive steps of lunar surface exploration and initial utilization by providing a gateway and a "safe haven" with permanent access to and from Earth and the surface of the Moon (MS2).

Furthermore, the station shall make use of the special properties of the space environment beyond LEO for research (e.g. radiation) and serve as a test and verification platform for technologies and processes (e.g. enhanced ECLS and human factors) necessary for the exploration of Moon and Mars (MS3).

The station's development, deployment and operation shall allocate minimum costs, risk and time. Thus, it particularly should (MS4):

1. Make use of available technologies and existing infrastructures.
2. Facilitate a minimum configuration design with future growth potentials.
3. Provide accommodation for a permanent crew of 3 with stay times from 6 months up to 1 year and longer, with possible autonomous operation periods in between.
4. Provide a hosting capacity for an additional exploration crew of 3 (maximum temporary crew of 6).

Full operability shall be achieved in the 2015-2018 time frame with a proposed lifetime of at least 10 years (MS5).

5.5.2 Mission Definition

For defining the mission, first the objectives, then the requirements and constraints will be identified. The mission statement above names the forthcoming role of the space station as a gateway and a safe-haven for future activities as the primary objective of this station. The goal here is to create an initial, generic and affordable transfer platform in a short time frame, which other succeeding missions can base on, with special emphasise on lunar exploration, utilization and exploitation. This means that the *preparation and support* of these missions to the Moon and other destinations is the overall goal, not providing the exploration capabilities itself. This matches the further utilization goals addressing research in science and engineering disciplines relevant for succeeding space activities (see section 5.1). Following mission design objectives can be identified:

1. Prepare future cis-lunar, lunar surface and Mars spaceflight activities (*overall goal)*.
2. Create a human outpost, i.e. a pressurised and life supporting space station, serving as transfer platform ('gateway') and safe haven for expeditions on the way to and from the Moon (*primary*) and other locations (e.g. to the Earth-Moon/Earth-Sun Lagrange Points and Mars) (*secondary*).
3. Provide a research platform utilizing the space environment beyond LEO (*secondary*).
4. Provide a test and verification platform for Moon/Mars expeditions (*secondary*).

It should be mentioned that the given political framework for cooperation between Europe and Russia does not mean that there should be no cooperation with the USA, Japan, China or other interested partners. But the mission emphasises the intensifying collaboration between Europe and Russia in space by e.g. implementing the Soyuz launcher at Baikonur or Europe's spaceport at Kourou, French Guyana and their agreement on joint future space activities [NFM2003/10a].

Beginning with only having the mission statement above and the identified objectives, an interaction analysis of the mission and system elements by using an interference matrix [Bertrand1998] is helpful for compiling a preliminary list of requirements and constraints. Table 5.7 lists the output of this process for the design example. Each mission statement (MS1 to MS5) has been used as an entry point and the findings were sorted with respect to the mission element concerned with a particular requirement or constraint.

Table 5.7: Preliminary requirements and constraints

#	Constraint/ Requirement	Subject/ Value	Subject to Trade (y/n, range)	Reason
		Element 1: Mission		
1. R1	Utilization profile	human outpost in cis-lunar space on the way to/from the Moon and other destinations	Moon: n Other destinations: y	MS1+2 O2, O3, O4
2. R2	Utilization profile	make use of the utilization qualities of the space environment beyond LEO with emphasis on engineering research, primarily by exposing instruments and other hardware to the cis-lunar space	y	MS1+3 O4
3.	P/L accommodations	internal (pressurised) and external	y	MS3: engineering (test, verification), human factors
		Element 2: Location/Orbit/Environment		
1. R5	Location/orbit	easily accessible from typically used orbits and launch sites in Eu/Ru	n	MS2: permanent accessibility to/from Earth Rq. 4.2 + 4.3 O2
2. R6	Location/orbit	permanently accessible to/from typical exploration destinations, namely global access to the lunar surface	n	MS2: permanent accessibility to/from the Moon's surface (safe haven) O2
3.	Location/Environment	orbit above Van-Allen belt	n	long life time, crewed
4.	Orbit control	low orbit maintenance desired	y	MS4/high transportation costs to locations far-Earth
5.	Location/orbit	LLO or more likely: LL1 or LL4/5	y	MS1: beyond LEO Rq. 2.1 – 2.4

#	Constraint/ Requirement	Subject/ Value	Subject to Trade (y/n, range)	Reason
		Element 3: Attitude/Flight Mode		
1.	Flight mode	no preference due to mission	y	MS3: engineering+HF
2.	Flight mode	consider configurational flexibility	y	MS4: future growth
		Element 4: Launch/Transportation System		
1.	Use of launch and transfer vehicles	domestic Eu+Ru	assembly+major operations: n additional: y	MS1: Eu+Ru
2. C4	Launch sites	existing: Kourou, Baikonur	y	MS1+4 : available for Eu/Ru
3.	Candidate LVs	Ariane 5, Proton, Soyuz, etc.	y	MS1+4: available in Eu/Ru
4.	Candidate in-space transportation technologies	does not exist; may be derived from Ariane 5 upper stages (ESV,ECA,ECB), Proton Block-D upper stage, ATV, Soyuz TMA, Progress M1, Clipper, ARD?	y	MS1+4: existing/available in Eu/Ru
		Element 5: Crew		
1.	Crew intensity	crew-tended to permanently crewed with stay times (6-12 months) and crew rotation frequency	y	MS4 radiation doses
2.	Crew composition	mainly engineers, scientists, and later also passengers (exploration crews)	y	MS3
3.	Crew composition	balanced Eu/Ru mix + other international partners	Eu/Ru: n no. of other: y	MS1
4.	Crew tasks	science, assembly, maintenance, repair, rescue	y	MS3: research, MS4: growth MS2: safe haven, gateway ops
5. R4	Crew size	variable with nominal 0 and 3, with temporary capacity of add. 2 to 3; max. crew: 6	y	MS4 + crew rotation O2
		Element 6: Space Segment (Space Station)		
1.	Module composition	pressurised compartments, non-press. external platform	presence: n size/no./config: y	MS3+4
2.	Initial orbital capability (IOC)	universal, self-sustaining pressurised module	y	MS4: minimum config Rq.2: autonomous element due to location far-Earth
3.	Crew accommodation	Habitation compartments; consider safety issues	y	Rq. 5.1: stay times, radiation MS3: HF
4. C2	Configuration	Min. config. and complexity	y	MS4, Rq. 8.2

#	Constraint/ Requirement	Subject/ Value	Subject to Trade (y/n, range)	Reason
5. R8	Configuration	multiple docking, with capability for automatic docking; consider docking clearance	>2, y	MS2: gateway ops, i.e. arrival/departure of CTVs, modules and other vehicles
6. R9	Configuration	multiple berthing with nodes; extendible and modular architecture; support devices such as manipulator arms	n	MS5: growth potentials MS4/Rq.6.1: minim config.
7. C3	Module H/W candidates	Eu: Node, ATV-ICC, MPLM, COL, ERA, Cupola, Airlock, etc. Ru: FGB, SM, DC, UDM, SPP, Strela, etc. Others: JEM-EP, TransHab, …?	y	MS1: Eu/Ru Rq. 6.2: autonomous components MS4: existing and available H/W MS1: partners
8.	Launch vehicle constraints for transport in LEO	A5-ESV: m<21 t; 4.57x10.4 m^2 A5-ECB+: m<27 t; s.a. Proton-M: m<21 t; 4.1x11.8 m^2 Soyuz-FG: m<7.4 t; 2.7x7 m^2	n	Rq. 4.2 Rq. 7.5
9.	EVA and external robotics	EVA and remote manipulator capability	n	MS3 Rq. 5.4: tasks
10. R3	Operation modes	automatic, remote and manual control	n	MS4: crew-tended periods
11.	Power/Thermal control	comfortable and flexible to operation modes	y	MS3: research/engineering Rq. 5.5: crew size Rq. 6.1: operation modes
12.	Life Support and Environmental Control	comfortable and flexible to operation modes	y	Rq. 5.5: crew size Rq. 6.1: operation modes
13.	Life Support and Environmental Control	close material streams/cycles as possible with available technology; consider utilization of synergisms	y	MS1: beyond LEO, i.e. far-Earth with high transportation cost and restricted logistic flight frequency Rq. 8.2: cost and time (development)
14.	Propulsion System	Redundant thrusters, tanks and feed lines, flexible operation mode	n	MS1: safety and beyond LEO
15.	Propulsion System	Storable propellants	n	MS1: logistics beyond LEO MS5: lifetime
		Element 7: Ground Segment/Logistics/Infrastructure		
1. R7	Logistics	regular in- and outbound transfer of crew and cargo	n	MS3: crewed + experiments Rq. 5.1: crew intensity

#	Constraint/ Requirement	Subject/ Value	Subject to Trade (y/n, range)	Reason
2.	In-space infrastructure	LEO-support for in-orbit berthing, transfer vehicles	y	MS4: payload restrictions of available launcher
3. C3	Logistics H/W candidates	Eu: ATV or derived Ru: Soyuz TMA, Progress M, Clipper	y	MS1+4
4.	Telecommunication Interface	use commercial systems (GPS, Galileo) where applicable and direct communication with Eu/Ru available infrastructure (TDRS, ground stations)	y	MS1+4, no lunar com. network available or planned
		Element 8: Schedule/Cost/Risk/Management		
1. C1	Management	Eu/Ru; balanced integration and allocation of their industry and resources	n	MS1
2. C5	Cost and risk	minimised	y	MS4: min. config, exist. H/W
3. R10	Schedule	Begin of operation	2015-2018 , y	MS5
4.	Schedule	Lifetime	>10 years, y	MS5
Eu: Europe(an), Ru: Russia(n), MS: mission statement, O: objective, Rq.: requirement, S/S: sub system, H/W: hardware, LV: Launch Vehicle				

Table 5.7 also includes first decisions on the mission concept by using the findings of the analysis in sections 5.3 and 5.4. With the information gathered in Table 5.7, the necessary mission and system elements can be compiled. The architectural concept must include a minimum configuration space station (C2) with pressurised modules (R1) for habitation and life support for 3 crew members with a temporary hosting capacity of up to 6 persons (R3, R4), a laboratory and an external platform (R2) and other necessary systems, like an airlock for EVA as well as docking and berthing hardware, and capabilities, like node(s) and a robotic arm (R8, R9).

Existing Russian/European technologies and hardware must be the basis or origin of all modules and vehicles (C1). These include ISS hardware and vehicles (Columbus, Nodes, MPLM, Airlock, FGB, SM, ERA, Soyuz, ATV, etc.) and planned or experimental systems (Clipper, ARD, etc.), which will presumably be available in time (C4, R10). This excludes more advanced system utilization, solar or nuclear electric propulsion (SEP, NEP) as well as solar-sails, etc. within this initial near-term scenario. This does not rule out that these new systems could reasonably be expected mid-/long-term at a later stage of development. Concerning size and mass, all system elements must meet the requirements to be able to be launched with existing or planned Russian/European launch vehicles such as Soyuz, Proton, Ariane 5 and Onega (C1, C4, R5). In this connection, in-orbit support will be mandatory because of the mass restrictions of the current launch systems. Using the ISS or an US launcher service (by bartering or in-kind contribution) for such tasks could be an option.

5.5.3 Mission Architecting

With the set of preliminary requirements and constraints identified, one can proceed by laying out mission and system concepts. This report will focus on the most relevant mission elements and will provide a summary of the decisions made and the associated justification.

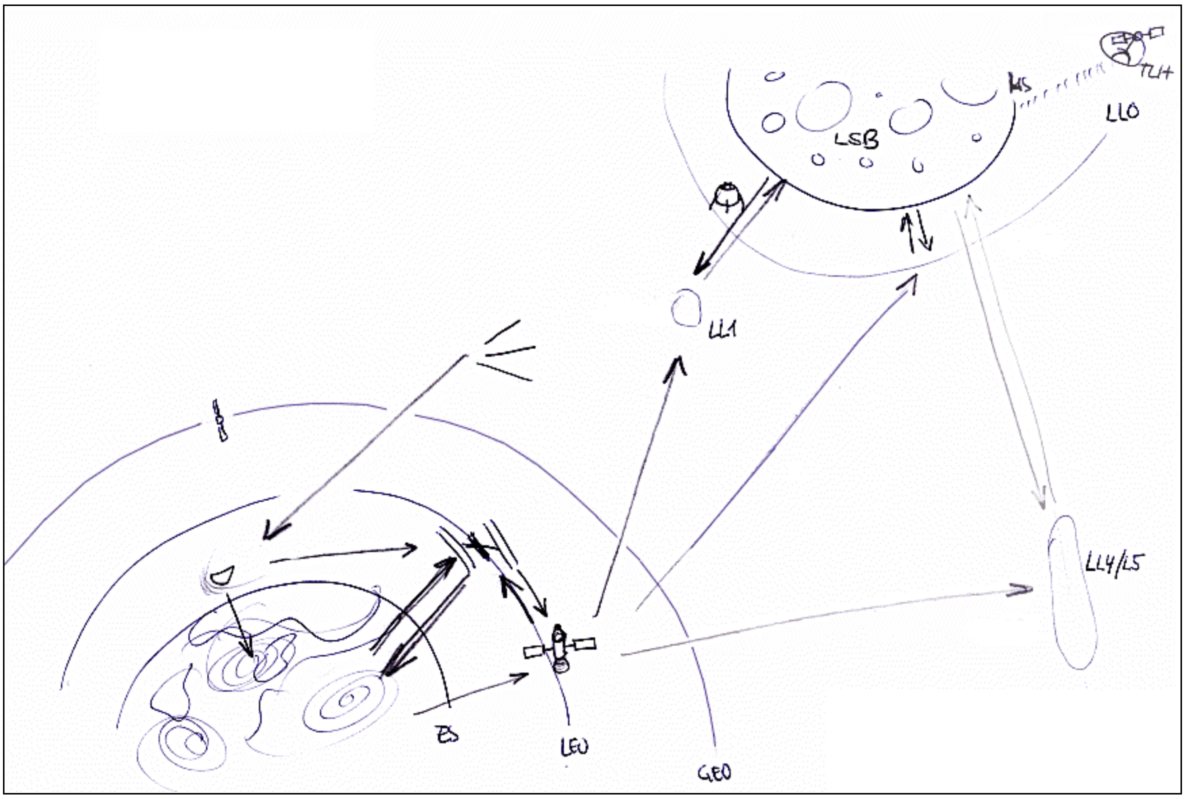


Figure 5.10: Sketch of mission concept alternatives

Alternative concept approaches

The mission statement defines the type of system, which the design process must generate as result, but does not specify the details of how this shall be achieved. The overall mission architecture to be designed is primarily driven by the selection of the location of the Lunar Space Station (LSS), the LEO support infrastructure and the available transportation systems (launch vehicles and in-space systems and their performances). Figure 5.10 shows a first sketch of selected alternative mission options. All have a space station at a “lunar location” serving as a link between the Earth and Moon surfaces in common. Launch vehicles must transport the station’s modules to LEO or deliver them directly to their destination location for assembly. Transfer stages and spacecraft must be available to transport crews and cargos to support the assembly and operation of the station. Re-entry or atmospheric orbital transfer vehicles (AOTV) will conclude the individual missions of astronauts by returning them and their cargo to the surface of the Earth or to a LEO installation. The major differences of the considered alternatives are:

- The location of the space station (LLO, LL1, LL4/5)
- Type of LEO support infrastructure (ISS, other)
- Space transportation infrastructures and their performances to LEO and relevant destinations

The designer is not equally free in selecting between these alternatives. Due to the restrictions imposed by the requirements and constraints of Table 5.7, this section begins by discussing the space transportation issues first.

Space transportation infrastructure

The transportation infrastructure includes primarily launch systems to LEO or direct LTO-injection and in-space transportation systems (transfer vehicles). Presumably available launch systems of the designated project partners in the specified time frame are the vehicles and launch sites compiled in Table 5.8.

Table 5.8: Selected launch vehicles and their properties [Isakowitz1999][CDF23A][Yazdi2002]

Launch System	Ariane 5 ES/V	Ariane 5 ECA	Ariane 5 ECB	Ariane 5 ECB+	Proton M	Soyuz FG/U	Onega
First flight	2002	2005	-	-	2001	2001	-
Payload@LEO/ISS [t]	21	N/A	N/A	27	21	7.8	14.5
Payload [m] diameter x length	4.57 x 10.35				4.2 x 13.1	2.7 x 7	N/A x 10
Launch cost [MEUR]	~200 (incl. ATV)				110	30-50	N/A
Launch site	Kourou				Baikonur	Baikonur/ Kourou	Baikonur/ Kourou
Man rated	No	no	no	-	no	yes	(yes)
Remarks	EPS-V upper stage	ESC-A (H14.4) upper stage	ESC-B (H24.1) upper stage	Fictional (H28) upper stage			

These systems will define, or rather limit, the size and mass of the station elements and the size and performance of the transfer vehicles. Besides systems available and already used today (A5-ES/V, Proton M, Soyuz ST), this list also includes enhanced versions of which developments are planned to date (A5-ECA/ECB, Onega).

Onega is an enhanced version of the Soyuz launch vehicle planned to carry a new Russian crew transportation vehicle, called *Clipper* [Korolev2004]. Another not yet planned or available vehicle listed is an enhanced version of A5-ECB, called A5-ECB+ or A5-27. The modifications made here were envisioned by industrial experts in a study performed at ESA and are assumed to be feasible and available by 2015 [CDF23A]. The enhanced Ariane 5 has an increased LEO payload delivery capacity of approximately 27 tons by increasing the central stage's thrust (+26%) and specific impulse (+3%), replacing the boosters with lighter versions (-28%) and increasing the upper stage's thrust (+11%) and propellant mass (+16%). In addition, one other Russian launch vehicle might be available, the *Angara 5A* rocket, offering a LEO capacity of 27 to 28 t ([Isakowitz1999], [SD2005]) and being planned to be operational from Baikonur until 2012 [NFM2005/02].

Appropriate transfer vehicles for human spaceflight missions to lunar destinations have not existed today since the Apollo programme. Thus, the conceptual design work will include such vehicles based on existing systems and define mission modes for their operation (see section 5.5.4).

LEO support infrastructure

Given an appropriate heavy lift launch vehicle (HLLV), all station elements and transfer vehicles could be launched, similar to the Apollo missions, directly to the station location. This would make a *Saturn* or at least *Energia* class launch vehicle necessary. Since these do not exist today, one must launch the transfer vehicle components separately and use some kind of LEO operation for mating, crewing and preparing the transfer vehicles for lunar transfer injection (LTI) and the cruise phase. For the chosen example, two alternatives exist:

One option is using a freely chosen parking orbit. Then, the resulting scenario is independent from any existing system such as the ISS and is self-sustaining from an operational point of view and could utilise a LEO best suited for the trans-lunar transfer requirements. However, a drawback is the fact that this solution demands extended free-flying and autonomous operation of all elements to be mated in orbit and requires launches in a rather short time frame. This inhibits potential backup and safe haven options as offered by the ISS. Thus, it would make the concept more complex and riskier.

As already mentioned, the second option is integrating the ISS into the concept as gateway/staging platform as a *lunar transfer preparation base* (LTPB), on which the vehicle and its elements could be docked, mated, checked out and crewed. This would include a safe haven available at LEO. Taking into account that using ISS' orbit imposes no principle drawback for lunar transfer injection (see section 5.4) this solution appears attractive. There is one drawback however; the ISS becomes a critical mission element upon which the project builds. Nevertheless, one can assume the second option is well-suited for this example, especially because the mission statement clearly welcomes usage of existing, well-proven infrastructure.

Space station location

As stated in Table 5.7 the space station location must have good access characteristics to and from both the Earth and the Moon. Such a "lunar location" could actually address different orbits, i.e. in Low Lunar Orbit (LLO), at one of the Lunar Lagrange Points (LLP) or others (see section 5.4). Locations such as circular or high eccentric Earth orbits (HEO/HEEO), or trans--lunar halo orbits (e.g. around LL2) offer no specific advantages, therefore, the three considered alternatives are:

A. a lunar orbiting space station at LLO
B. a cis-lunar space station at LL1
C. a libration-point space station at LL4 or LL5

Which of these locations is best suited depends on their properties concerning design parameter with respect to the mission statement. Summarised for this case, a location is hereby as better as it has:

- Low delta-v for in/outbound Earth-transfer

- Low station-keeping delta-v
- Low delta-v to lunar surface locations
- Highest accessibility to the relevant locations
- Low transfer time, especially for crewed flights
- Best visibility and communication links

The accessibility does not refer only to a location being reachable in principle (Moon surface latitudes, LEO and Earth return) but there must also frequently appear large transfer windows. Table 5.9 summarises particular properties of the station location alternatives and show that all three considered locations have their respective pros and cons.

Table 5.9: Properties of potential lunar space station locations (transfers to LLO assume arbitrary inclination and RAAN; section 5.4, [Hoffman1993], [Euler1971], [CDF23A], [Connolly2004] [Broucke1979])

	Property	LLO (100 km)	LL1	LL4/5
Δv	LEO/ISS →	4.2 km/s	3.81 - 3.93 km/s	4.00 - 4.12 km/s
	Lunar TWP	2 h (equatorial)–14 d (polar)	7.7-11.3 d (mean: 9.9 d)	
Δv	Earth landing ←	1.12 km/s	0.65 km/s	0.86 km/s
ΔT_E	Earth transfer time	5.4 d	3.8 d	4.6 d
	Earth TWP	2 h (equatorial)–14 d (polar)	Permanent	
Δv	Lunar landing ←	1.87 km/s	2.52 km/s	2.58 km/s
Δv	Lunar surface →	1.87 km/s	2.52 km/s	2.58 km/s
ΔT_L	Lunar transfer time	hours	1-3 days	11-22 / 7-15 days
Δv	Station-keeping per year	≥80 m/s	0.5 - 36 m/s (Lissajous orbit)	N/A
Lunar surface accessibility		Latitudes ≤ inclination	Global	Global
Lunar surface visibility		Latitudes ≤ ~inclination (high-res, successively)	Approx. half globe (med-res, near side)	Approx. half globe (low-res, leading/trailing at ±60°)
Communication link from a LSB		Periodically, variable	Permanent, still	Permanent, still
Orbital period		118 min	27.2 d	27.2 d
Occultation free period		3 days (twice per lunar month)	Permanent	Permanent

The LLO provides excellent lunar surface observation resolutions and minimum landing delta-v, or rather the lowest amount of propellant necessary for surface excursions, thus, leading to the smallest lunar excursion vehicles (LEV). The maximum eclipse duration is about 0.8 hours with a maximum eclipse-free period of 38 days twice per year [CDF23A]. These eclipse cycles are also supportive for the EPS and TCS subsystem sizing. Otherwise, a LLO has severe drawbacks. It requires the highest effort for station-keeping and Earth-transfer and its orbit maintenance effort is currently difficult to predict (see section 5.3.1). More important, a LLO with a high inclination does not feature a permanent Earth-return capability due to the restrictions to the transfer window period (TWP). If a lunar orbital stage is used, surface stay time of the lander is strictly restricted due to the orbit plane rotation.

The facts of the considered Lagrange points are rather contrary to the LLO. Because they are synchronous to the Moon's motion and rotation, only one of its sides is visible. Due to the constant viewing conditions, photographic cameras can be operated with high exposure times, but due to much longer lunar surface distances, larger optical devices are necessary to achieve comparable ground resolutions. Though, the equilateral Lagrange points LL4 and LL5 are no reasonable choice due to the long lunar transfer times. At LL1, station-keeping delta-v is very low and depends on the size of the orbit around the equilibrium point. For a small 180x500 km wide quasi-elliptic orbit for instance, investigations showed that the station-keeping delta-v can be expected to be below 1 m/s per year ([Euler1971],[Hoffman2005]). But even very large orbits will need only a small amount of propellant for orbit maintenance (e.g. 36 m/s delta-v per year for a 50000 km wide orbit [Akin2001]). Eclipses occur much less frequent than in LLO; from twice per revolution (i.e. about every 14 days) for very small orbits to twice per year for large orbits. The occultations of the Sun, however, are much longer and depend on the size of the orbit as well (3 to 9 days for a large orbit). Detailed designs require a trade-off to obtain a compromise between station-keeping delta-v and power storage capacity. Furthermore, the lunar surface transfer delta-v is larger than at LLO but is of same magnitude.

An important advantage of the LL1 is the far better Earth transfer properties and permanent Earth visibility. This minimises the in/outbound delta-v, leading to the smallest transfer vehicles to and from Earth. Together with the very low station-keeping requirements this makes the LL1 most favourable and is therefore selected for this sample scenario and in the following analysis. The scenario of this design example is named *European Lunar Libration Point Scenario* (ELLIPSE).

5.5.4 Space Transportation Vehicles

The space transportation infrastructure is vital for assembly and operation of a space station. For an ELLIPSE scenario at least three classes of transfer missions with optimised transfer vehicles must be available. Principle classes are assembly, cargo and crewed missions. These classes incorporate various subclasses of missions such as assembly missions of different payload size and cargo logistics missions with or without payload recovery capability.

Table 5.10: Phases of lunar space station transfer missions

#	Phase description	Path
1.	Preparation phase (launch and in-orbit assembly of transfer vehicle with payload)	Outbound
2.	Lunar transfer phase (LTI, LTO-coast, LTO-MCC, target OI)	Outbound
3.	Rendezvous and docking phase (station acquisition and arrival)	Outbound
4.	Docked phase (passive)	-
5.	Station departure phase (undock and leave station vicinity)	Inbound
6.	Earth transfer phase (ETI, ETO-coast, ETO-MCC)	Inbound
7.	Earth atmospheric re-entry and landing (latter if applicable)	Inbound

LTI	*Lunar Transfer Injection*	*OI*	*Orbit Insertion*
LTO	*Lunar Transfer Orbit*	*ETI*	*Earth Transfer Injection*
MCC	*Mid-Course Corrections*	*ETO*	*Earth Transfer Orbit*

Given a space station location, a type of transfer vehicle architecture must be selected that integrates the different mission phases listed in Table 5.10 for each class of transfer missions. Each phase listed in this table is initiated by dedicated events, i.e. manoeuvres or activities as stated and can include further state changes (e.g. stage separation).

Because no dedicated in-space transportation system has existed since Apollo that could be used as human and cargo transfer vehicles enabling such round trips to lunar destinations, conceptual designs of such vehicles are included and the modes of launch, assembly, staging and recovery or disposal of their elements have to be defined. In this connection, it is assumed that expended elements shall be recovered for re-use or disposed by destructive atmospheric re-entry. No elements shall be discarded to lunar surface to prevent large-scale contamination due to regular lunar spaceflight activities.

The assumed delta-v's of this design example are summarised in Table 5.11. All values include margins by taking data from existing systems (Soyuz, ATV), except LTI, LL1-OI and ETI, for which a corresponding margin of 5 % is applied during design calculations in addition to the analytically calculated propellant mass requirement. For LL1-R&D no delta-v values are available, thus, the same value as LEO-R&D is taken that is presumably largely overestimated.

Table 5.11: Assumed delta-v requirements for considered transfer missions

Manoeuvre	Delta-v [m/s]	Comment
ISS-R&D	185	Rendezvous and Docking to ISS
LEO de-orbit	130	Undock from ISS and re-enter Earth atmosphere
LTI	3066	Lunar Transfer Injection to LL1
LL1-OI	870	Orbit Insertion to LL1
ETI	650	Earth Transfer Injection to Earth
MCC	50	Mid-Course Correction (LTO and ETO)
ACM	25	Attitude Control and Free-Flight Manoeuvring (LTO and ETO)
LL1-R&D	185	Rendezvous and Docking with a Lunar Space Station

Definition of Vehicle Types

Table 5.12 lists principle hardware elements that must be available to achieve the mission objectives of station assembly and logistics. Common restrictions include constraints due to the available launch vehicles in terms of dimension and launch mass (see Table 5.8, page 104).

Table 5.12: Elements of the in-space transportation architecture

Element	Purpose	Relevant hardware heritage
LT-Stage	High-thrust autonomous spacecraft or propulsion stage for orbit transfer manoeuvres of high delta-v, especially for Lunar and Earth transfer injections (LTI/ETI) and optionally for mid-course corrections (MCC).	Proton Block-D upper stage Ariane upper stages ATV / Progress AOCS or enhanced upper stage RCS components
Propellant Carrier	Propellant tank module supporting the LT-stage within a mission and that can be jettisoned after depletion of its capacity	Upper stage tanks (e.g. A5, Proton)
Crew transport	Crew transportation spacecraft for a crew of at least 3 astronauts during LEO and LSS missions	Soyuz TMA (Zond, Soyuz-LOK), Clipper, ATV with ARD-derived return capsule (similar to Apollo-CM)
LEO-logistics	Autonomous spacecraft capable of delivering passive payloads (modules, cargo carrier) to the ISS and re-entering Earth's atmosphere	ATV-SC, Progress M-SM
LSS-logistics	Autonomous spacecraft capable of delivering passive payloads (modules, cargo carriers) to the lunar space station, return to Earth and re-enter Earth's atmosphere	ATV-SC, Progress M-SM
Cargo Carrier	Cargo logistics module that contains payloads of various sorts (dry goods, fluids, etc.) during launch to the ISS and in-space transportation to the lunar space station	ATV-ICC, Progress M-OM

Vehicle Layout

For the conceptual design of the transfer vehicles one must follow an iterative and concurrent process including design of mission modes. The launch vehicle database, mass models and performances for LEO/ISS delivery have to be taken into account as well as sizes of tank structures.

The previous section listed principle classes of vehicles and necessary elements. With respect to the mission statement and its constraints, the designer will define these vehicles, which are a) capable of enabling the transfer missions and b) feasible of being built and launched. Many design and analysis iterations including trade-offs of possible configurations must be made to finally find a suitable transportation infrastructure.

Leading design rules meeting the mission statement requirements on cost, risk and schedule are:

- Minimise number of launches for individual missions! Launches are the primary cost driver from operations' point of view.
- Minimise the diversity of elements! Try to create generic systems by using synergisms, re-using vehicle components for different missions and integrating sizing opportunities.
- Make use of existing hardware and minimise necessary modifications! Every modification includes new uncertainties that must be balanced by reasonable margins.
- Use data of actual built and working hardware to estimate the performance of your system!
- Design conservatively, but do not overestimate the masses by too large margins! Space system design typically works at the border of feasibility, and every new system will always be a challenge. In the worst case, margins which are too high make a scenario unfeasible. In the best case, the concept becomes financially prohibitive, e.g. too many launches are required.

Concerning this design example, one begins by specifying the mass and geometric properties of deliverable elements to the ISS. The launch vehicle performances and the LEO manoeuvring systems for rendezvous and docking must be taken into account.

With this information at hand, one proceeds with the core design problem; the definition of the transfer systems, i.e. the dry masses including structure, power, thermal control, attitude control, docking mechanisms, avionics and communication, propulsion; and finally the required amount of propellant leading to propellant volume and tank sizes. Configuration and tank sizing will primarily depend on the mission mode, namely the staging layout and the functional capabilities of the individual elements. During this process a reversed approach from the inbound-leg to the outbound-leg is reasonable because every mission phase (Table 5.10) and the amount of propellant necessary for enabling it must be provided by the system of the preceding phase before. Thus, the return vehicle and descent capsules are designed and then followed by the system that delivers the return vehicle. Because these systems are typically interrelated, this process is highly iterative and also needs creativity in terms of functionality distribution. Software tools can help significantly to reduce the turn-around time during design and analysis by accelerating the processing and evaluation of modifications.

The rest of this section summarises core elements of the designed in-space transportation architecture of the lunar space station example. Whenever possible, existing systems were chosen to provide reliable data on mass and performance. In addition to considered margins, this rather conservative approach guarantees a safe conceptual design with potential for enhancements and optimisations. Depending on the propulsion types, tank residuals (i.e. non-usable propellant in a tank) of 1 to 2% were taken into account. For delta-v calculations a margin of 5% is assumed. Boil-off of cryogenic propellants is neglected for large tanks with short mission duration but taken into account for small tanks or mission durations larger than 30 days by using reference values provided in [NASA-OTV].

ATV-HD

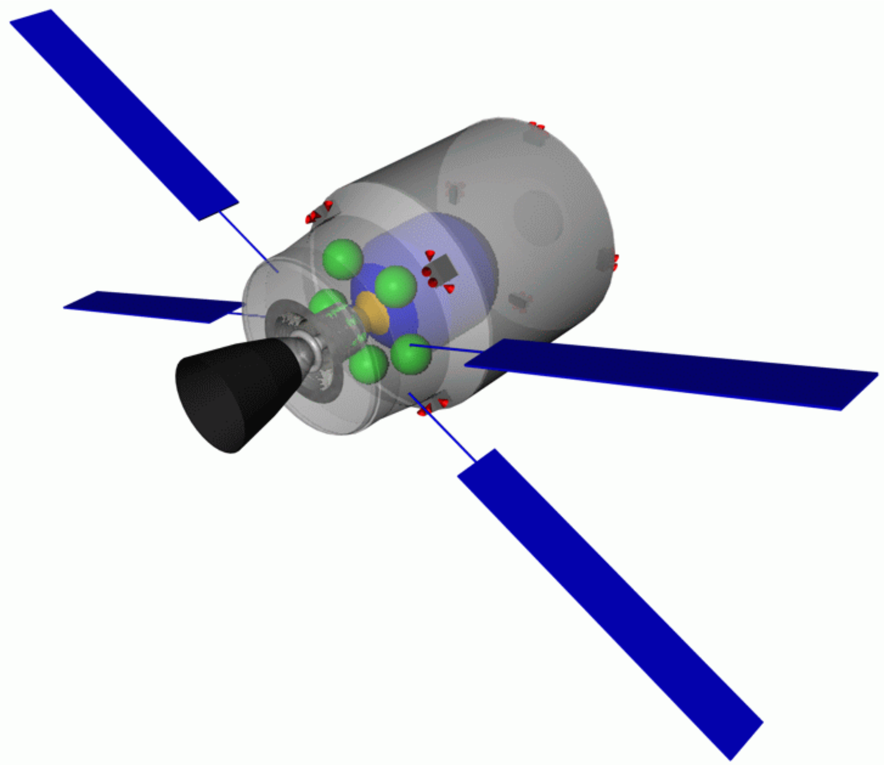
Table 5.13 describes the ATV-HD, a heavy-duty derivative of the European *Automated Transfer Vehicle* (ATV), which is currently being built for ISS logistics and orbit re-boost. Within this ELLIPSE scenario this new vehicle serves as the high-thrust lunar transfer stage (LT-stage). In addition to ATV spacecraft subassembly (ATV-SCSA) with integrated power and avionics section, the rocket engine *Vinci* and a slightly downsized cryogenic tank complex of the Ariane 5 enhanced upper stage ESC-B/H24.1 are used. To save mass the original ATV propulsion system (using hypergolic bi-propellant thrusters) is being replaced by thruster clusters similar to the system used on Ariane 5's *Vehicle Equipment Bay* (VEB), but using an enhanced multiple re-ignitable version of a small cryogenic engine [EADS]. The latter enable the use of the main propellant tank for attitude control as well and therefore reduce tank and support hardware mass considerably.

The illustration shows a potential configuration of this vehicle by attaching Vinci to the ATV-propulsion bay (EPB) and integrating the upper portion into ATV's centre section, which was originally empty. Using this configuration, internal heat rejection shields will certainly be necessary at the aft end of the EPB. Other configurations could integrate the avionics bay in the forward section in front of the tank and possibly installing only the ATV Solar Generator System (SGS) at the aft section.

Not visible in the image is the forward structure with a Russian rendezvous and docking system with a simplified, non-pressurised derivate of the Russian Docking System (RDS). Depending on the mission, this is an active RDS with a passive counterpart. An active system is used for mating the External Tank Modules (ETM) (see below) or the payload to be delivered in advance to the ISS. A passive system is used for crewed or logistic missions in order to allow the Soyuz CTV/Cryo-CTV or Progress LTV/Cryo-LTV spacecraft (see below) to dock to ATV-HD before trans-lunar injection.

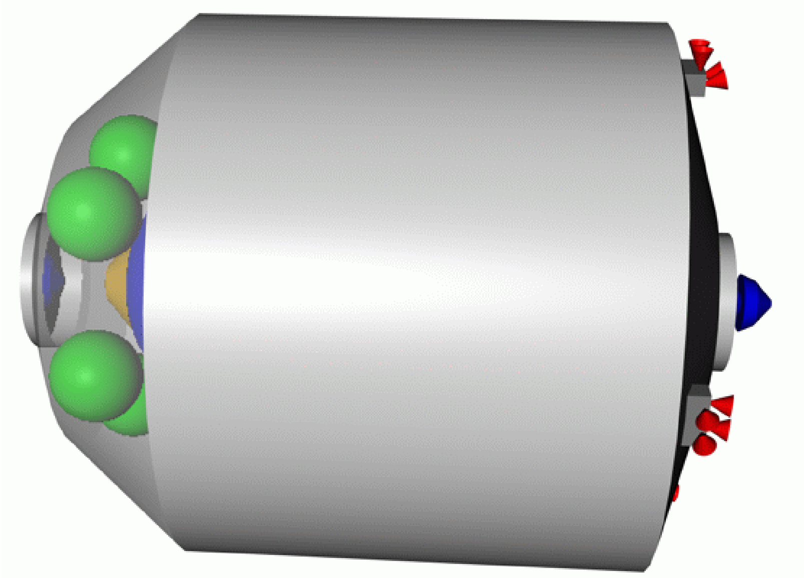
ATV-HD payload delivery capacity is stated in the table below. Here "2L" gives the values for Two-Launch missions, i.e. one launch for the payload and one for the ATV-HD itself. "3L" missions include one additional launch for one ETM (3L-ETM) or another ATV-HD (3L-ATV-HD). Thus, in these cases a two-stage LTI-stage is used, which increase the delivered payloads as stated. Theoretically more than one ETM could be used, which would increase payload from 10 to 16 tons each; the number is, however, restricted in practice because the ETMs must be jettisoned after propellant depletion which includes undock and re-dock operations. For the current scenario only 2L and 3L missions are taken into account. After mission completion the ATV-HD returns into Earth's atmosphere and burns up during direct entry.

Table 5.13: Summary of the "ATV-HD"



Name	Automatic Transfer Vehicle – Heavy Duty		
Purpose	Transfer stage for high thrust orbital manoeuvres (orbit injection and corrections: LTI, MCC LL1-OI, ETI) for transport payloads to lunar destinations		
Main propulsion	Vinci cryogenic rocket engine (180 kN thrust, 467s specific impulse)		
RCS propulsion	32 x 300 N cryogenic thrusters (415s specific impulse) at 8 clusters		
EPS	Photovoltaic solar arrays (α-tracked), 33.6 m², 4.8 kWe Batteries: NiCd 8 x 68 Ah		
TCS	Body-mounted radiators		
Payload types	Station module (autonomous or passive), CTV/LTV, etc.; ETM (optionally)		
Direct payload deliveries [t]	2L mission:	3L-ETM mission:	3L-ATV-HD mission:
	13.5 @ LTO 7.00 @ LL1 6.45 @ LL1-Station	29.7 @ LTO 17.1 @ LL1 16.6 @ LL1-Station	32.0 @ LTO 19.5 @ LL1 17.9 @ LL1-Station
Dry mass [t]	4.9		
Propellant [t]	22.1 cryogenic propellants (LOX/LH2 with mass ratio 5.84:1), He pressurised		
Gross Mass [t]	27.0 @ launch, 25.9 @ LEO departure		
Launch Vehicle	Ariane 5 ECB+		
Lifetime	15 days nominal mission duration		
Remarks	Vehicle is based on Ariane 5 ECB upper stage (H24) and ATV-SCSA		

Table 5.14: Summary of the "ETM"



Name	External Tank Module
Purpose	Cryogenic propellant carrier with 2 docking ports to be used by ATV-HD during LTI
Main components	Enfolded tank complex (main tanks) 5 pressurization tanks (He) 2 non-pressurised RDS ports with equipment (AFT: passive, FWD: active) Harness (piping, vents, sensors, power lines, etc.)
Payload	17.8 t cryogenic propellants (LOX/LH2 with mass ratio 5.84:1), He pressurised
Dry mass [t]	2.1
Net mass [t]	19.9 @ launch and LEO departure
Launch Vehicle	Ariane 5 ECB+ with ATV-L carrier
Lifetime	Designed for 22 days nominal mission duration
Remarks	Delivered to ISS by ATV-L

ETM

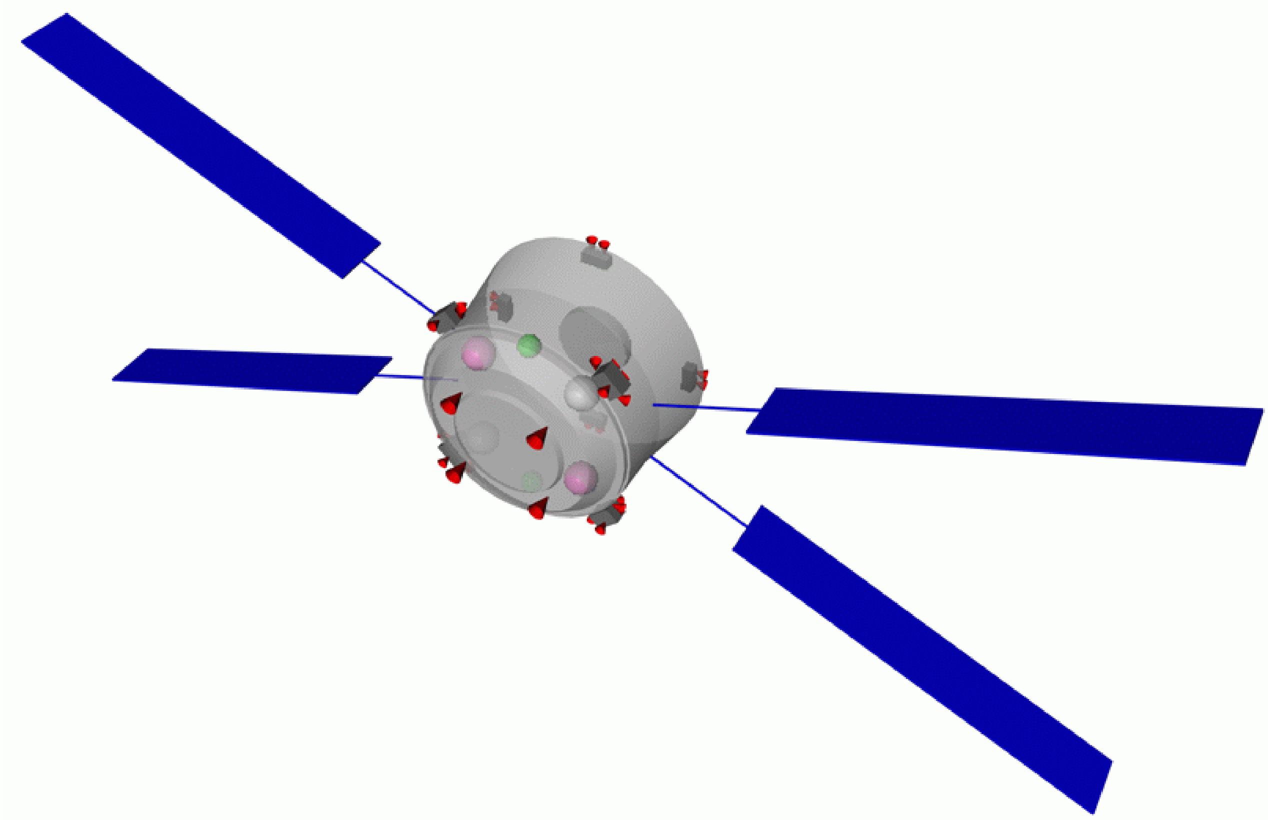
Table 5.14 describes the External Tank Module (ETM) supporting ATV-HD operation during 3L-ETM missions. As mentioned above, this module serves as an additional propellant tank for the LTI manoeuvre with heavy payloads. Basically the propellant tank and supporting hardware are the same as used for the ATV-HD, except for the reduced tank size. Two non-pressurised RDS ports are integrated. The aft docking port (passive) provides mating and transfer of propellant to the main stage. The forward port serves as connector to the payload (and optionally to a second ETM; a 4L mission type which is not considered here).

The ETM front section is furthermore equipped with the forward avionics elements of RDS (*Kurs* antenna, visual target video, etc.), which allows the ATV-L carrier (see below) the delivery of the ETM to the ISS. After propellant depletion, the ETM is being jettisoned from the ATV-HD.

ATV-L

Table 5.15 describes the LEO-logistics vehicle to transport cargo carriers and passive station modules to the ISS. Due to the similarity to the ATV's mission, basically the same hardware, namely the ATV-SCSA, is used.

Table 5.15: Summary of the "ATV-L"



Name	Automatic Transfer Vehicle – Logistic
Purpose	Autonomous LEO spacecraft for delivery of passive elements to the ISS
Main propulsion	4 main engines (490 N thrust each, 324s specific impulse)
RCS propulsion	4 cluster with 5 x 220 N thrusters each (324s specific impulse)
EPS	Photovoltaic solar arrays (α-tracked), 33.6 m^2, 4.8 kWe Batteries: 4 x 40 Ah (NiCd) and 4 x 96 Ah (LiMnO2)
TCS	Body-mounted radiators
Payload types	ETM, station module (passive), cargo carriers, etc.
Payload delivery [t]	19.8 @ ISS (max)
Dry mass [t]	4.1
Propellant [t]	2.2 (NTO/MMH), He pressurised
Net mass [t]	6.3 @ launch
Launch Vehicle	Ariane 5 ECB+
Lifetime	10 days nominal
Remarks	Vehicle is based on the ATV Spacecraft Subassembly

Because no ISS-reboost propellant delivery is necessary, fewer tanks are needed and the propulsion bay size can be reduced. This increases the ATV-L gross cargo mass to ISS. A prerequisite for payloads to be transported by the ATV-L to the ISS (such as the above described ETM) is outfitting the modules with forward RDS elements necessary for docking, which were originally located at the front-cone of the ATV's Integrated Cargo Carrier (ICC). This includes avionics such as the *Kurs* antenna, visual target video, etc. and a simplified, non-pressurised RDS mechanism. The forward propulsion system elements of the ATV (e.g. the thruster clusters), however, are transferred and integrated into the ATV-L central avionics bay. Structural reinforcements due to the larger payload mass are neglected. Like the ATV, this vehicle burns up in Earth's atmosphere after completion of its LEO mission.

Soyuz-CTV and Progress-LTV

The Russian *Soyuz* spacecraft has a long history with its origin dating back to the 7K family developed during the Soviet manned lunar programme in the 1960s [Hall2003]. The principle design was then used for crewed and logistics flights to Russian space stations *Salyut* and *Mir* [Portree1995]. Today, two improved versions, *Soyuz-TMA* and the *Progress-M*, are in operation within the ISS programme.

Because of the compact design and modularity of these vehicles, they offer interesting options for lunar missions, which was their initial purpose with the original design of 7K-LOK. Because the hardware is directly available today, these vehicles are selected within this sample scenario and modified to enable round-trip missions to the lunar space station.

The main advantage of Soyuz compared to other alternatives (especially Clipper) is the minimised entry mass due to the compact Descent Module (DM). This is crucial for enabling a two-launch mission mode (2L) with the launch system taken into account.

Soyuz is used as a crew transfer and rescue vehicle and Progress as a small logistics vehicle, both supported by an ATV-HD as transfer stage. Table 5.16 summarises the characteristics of these vehicles. Based on Soyuz-TMA, or rather Progress-M, the following changes were made to the original design:

- NTO/MMH propellant is used instead of NTO/UDMH, which allows a higher specific impulse I_{sp}=324s instead of 305s and offers compatibility to other used propulsion systems.
- 4 smaller main thrusters instead of the one original thruster due to redundancy and availability reasons
- Enlarged propulsion section, due to the increased propellant mass necessary for LL1-orbit insertion and Earth-transfer injection. The tank integrates with the hull and is covered by thermal blankets.
- Thicker heat shield at the descent module for direct atmospheric entry from lunar distances
- Increased autonomous free flight duration from 4.2 to 14 days by additional amount of consumables of the crew and the ECLSS.

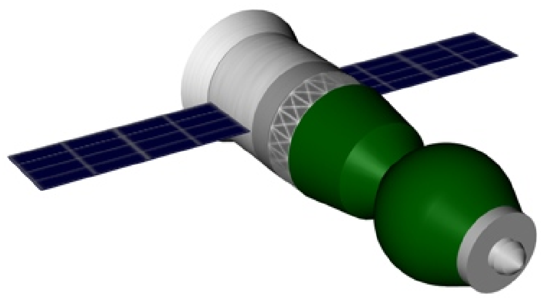
Furthermore, an increased on-orbit storage lifetime is assumed, enabling maximum total mission duration of about 400 days instead of the current 210.

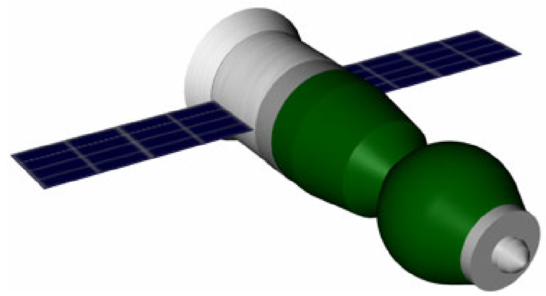
Due to the larger lift-off mass of about 13 tons, no conventional Soyuz-FG launcher can be applied. If available, the first choice would be the planned *Onega* for Clipper, which is based on the Soyuz launcher and would facilitate an adequate Launch Escape System (LES). The other options are Proton or Ariane 5, which are not currently used for crewed flights. Thus, they are rather unlikely due to the man-rating effort.

Cryogenic Soyuz (Cryo-CTV) and Progress (Cryo-LTV)

A more advanced design is feasible by changing the Soyuz infrastructure in order to use cryogenic propellants. Due to the significant higher specific impulse of 415s (using an available thruster design [EADS]) the propellant mass is reduced and additional payload mass becomes available.

Table 5.16: Summary of the Soyuz-based "CTV" and Progress-based "LTV"



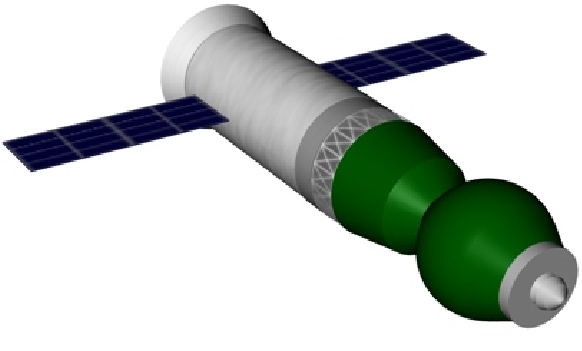


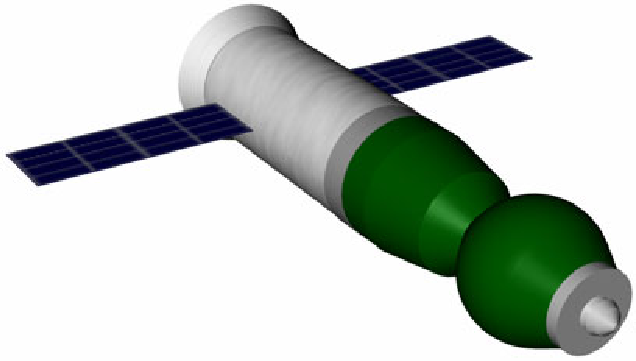
Name	Crew Transfer Vehicle	Lunar Transfer Vehicle
Purpose	Spacecraft for launch/in-space transportation and return of crews and small portions of dry cargo to and from lunar destinations. It is storable in space for emergency crew return.	Spacecraft for launch/in-space transportation and return of a medium-sized portion of fluids, propellants and dry cargo items to and waste disposal from lunar destinations.
Dry mass [t]	6.8	4.5
Propellant [t]	6.1 (NTO/MMH), He pressurised	
Gross Mass [t]	13.1 @ launch, 12.5 @ LEO departure, 8.9 @ LSS	
Launch Vehicle	Onega	
Main propulsion	4 rocket engines (2 kN total thrust, 324s specific impulse)	
RCS propulsion	14 x 137N (main) + 12 x 25N (backup) thrusters (324s specific impulse)	
EPS	Photovoltaic solar arrays, 10 m^2, 1.4 kWe / 0.6 kWe (max/average)	
TCS	Body-mounted radiators, 8 m^2	
Crew	2 or 3 (2M or 3M)	-
Payload types	Crew and dry cargo	Fluids, dry cargo and propellant (NTO/MMH)
Payload, up [kg]	192 (2M) or 120 (3M)	2740 nominal; 3010 maximum (no P/L-down)
Payload, down [kg]	122 (2M) or 50 (3M)	1500 nominal; 5090 maximum (no P/L-up)
Lifetime	14 days independent operation; 370 days mated to station	
Remarks	Vehicles are based on Soyuz-TMA and Progress-M spacecraft with modification to the propulsion system	

Combined with liquid oxygen (LOX), both liquid hydrogen (LH) and liquid methane (LM) are fuel candidates. In any case, major changes to the propulsion system are necessary, such as additional structures and insulation.

Furthermore, designs using cryogenic propellants must address boiling-off of the propellants due to heat leakage through the tank walls. Within this example LH/LOX is chosen, which is the more demanding case in terms of propellant losses and consequently the additional mass reserves. Long-term storage of cryogenic vehicles is, however, not realistic today. It is, hence, reasonable to combine and integrate both Soyuz/Progress propulsion designs into the scenario, the original with storable and the other with cryogenic propellants. Because the first could be parked at the station for a rather long time and only exchanged with low frequency (yearly or longer), it is ideal as a rescue vehicle for the crew.

Table 5.17: Summary of the cryogenic Soyuz "Cryo-CTV" and Progress "Cryo-LTV"





Name	Cryogenic Crew Transfer Vehicle	Cryogenic Lunar Transfer Vehicle
Purpose	Spacecraft for regular launch/in-space transport/return of crew and portions of dry cargo to and from lunar destinations	Spacecraft for regular launch/in-space transport and disposal of a medium-sized portion of fluids, propellants and dry cargo items to and waste disposal from lunar destinations
Dry mass [t]	7.2	4.9
Propellant [t]	5.4 (LOX/LH), He pressurised	
Gross Mass [t]	13.6 @ launch, 12.8 @ LEO departure, 9.4 @ LSS	14.1 @ launch, 13.6 @ LEO departure, 10.3 @ LSS
Launch Vehicle	Onega	
Main propulsion	4 cryogenic engines (1.6 kN total thrust, 415s specific impulse)	
EPS	Photovoltaic solar arrays, 10 m², 1.4 kWe / 0.6 kWe (max/average)	
TCS	Body-mounted radiators, 8 m²	
Crew	3	-
Payload types	Crew and dry cargo	Fluids, dry cargo and propellant (NTO/MMH)
Payload, up [kg]	410 nominal, 570 maximum	3800 nominal; 3920 maximum (no P/L-down)
Payload, down [kg]	600 nominal, 1255 maximum	2150 nominal; 6500 maximum (no P/L-up)
Lifetime	14 days independent operation; 30 days mated to station	
Remarks	Vehicles are based on Soyuz-TMA and Progress-M spacecraft with major modification to the propulsion system	

The advanced cryogenic Soyuz/Progress can then be used for regular expedition flights to the station, with the former crew using the recently arrived spacecraft for Earth return. Thus, the total maximum in-orbit time comes down to about 30 days, for which these calculations are made and which results in an accumulated propellant boil-off of 10% of the return propellant. In reality, this value could even be reduced to approximately 11 days if the stay times on the stations (ISS and LSS) are kept short and the leaving crew directly departs after arrival of the new crew.

5.5.5 Space Station Configuration and Elements

Section 5.5.2 provides a preliminary list of necessary station elements. For configuration design the flight mode in respect to the Earth and Moon is of primary importance for power generation through photovoltaic solar arrays and thermal control. Principally, Earth-Oriented (EO, which is equal here to Earth-Moon-Oriented) or Inertial (IN) flight mode comes into question. Table 5.18 lists their characteristics.

Table 5.18: Characteristics of Lagrange point space station flight modes

Flight Mode	Characteristics	Priority
Earth-Oriented (EO)	✧ Positive aspects:	
	• Favourable for Earth/Moon observation and telecommunications (no antenna tracking required)	A
	• Use of gravity gradient for attitude stabilization (negligible effect)	C
	• More flexibility for positioning of microgravity payloads	A
	• Earth provides reference for crew orientation (EVA and IVA)	B
	• Constant rendezvous and docking circumstances	B
	✧ Negative aspects:	
	• Needs solar array and radiator tracking	A
	• Variable lighting conditions during EVA	C
	• Solar radiation pressure acts as perturbation torque (secular)	A
Inertial (IN)	✧ Positive aspects:	
	• Favourable for astronomy	A
	• Simplified solar collectors and radiators (no panel tracking required)	A
	• Constant lighting conditions during EVA	C
	• Constant thermal conditions; availability of “cold-traps” for cryogenic propellant storage	A
	✧ Negative aspects:	
	• Gravity gradient acts as perturbation torque	C
	• Difficult to keep optimum mass distribution during assembly and orbital segment growth	C
	• Regular re-orientation due to orbit control manoeuvres	A

Table 5.19 summarises the main characteristics of the designed lunar space station. This configuration reflects the requirement of a minimum configuration but nevertheless includes enhanced life support and habitation, environmental research capability and allowing for station growth. After assembly completion, it consists of nine main components, six pressurised modules, two vehicles and one solar power platform (SPP) including photovoltaic arrays, radiators and a truss boom that can be used as an initial exposed platform and which is supported by a robotic arm.

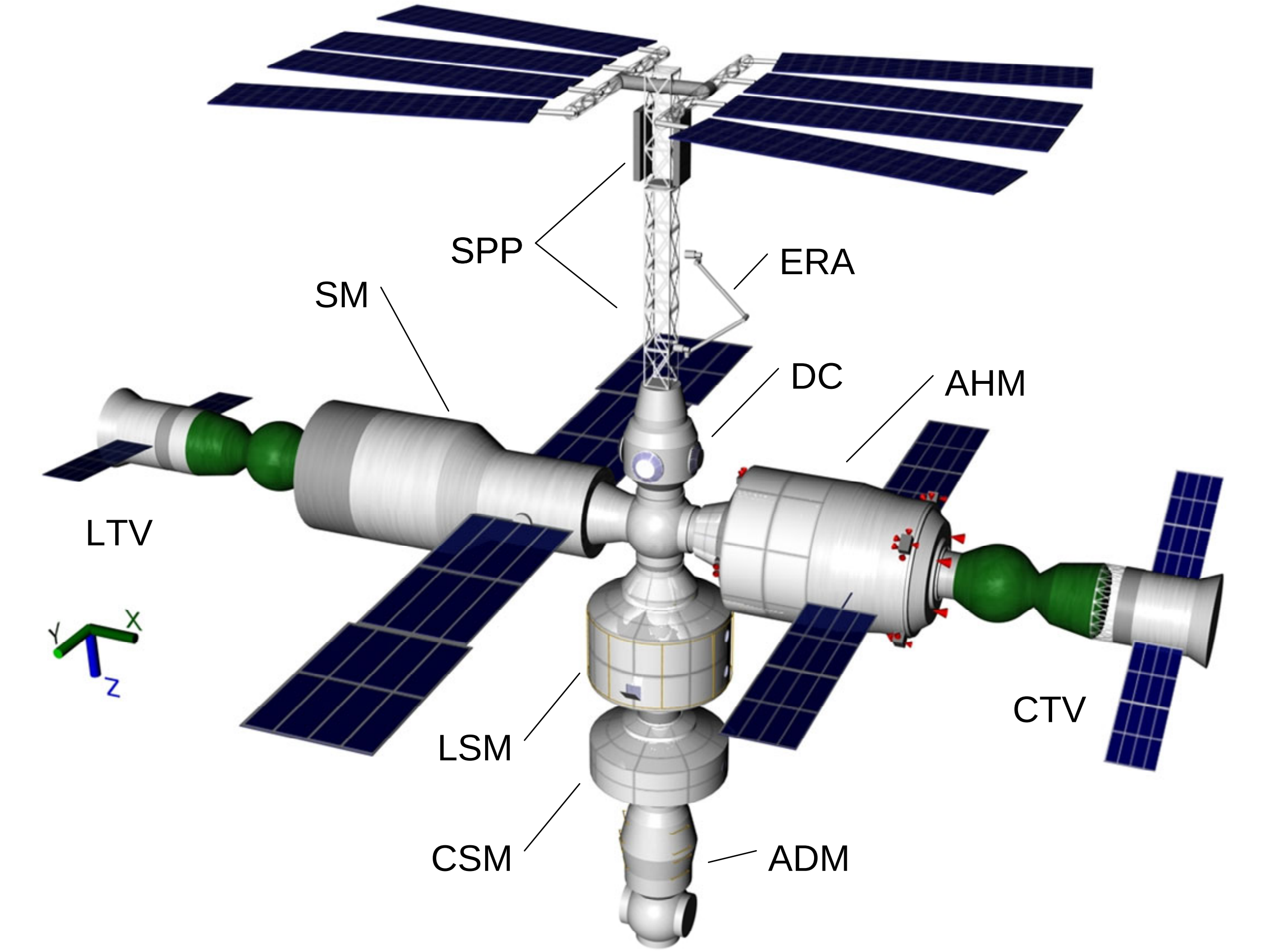
This first lunar space station is named *Eve*, to indicate both, its role as humans’ life-supporting Earth-replacement in cis-lunar space providing habitation and safety, and to mark the beginning of the days of humans’ deep-space exploration.

The station can be permanently crewed after installation of SM, LSM, SPP and the CTV. This assembly state is referred as “core-complete” configuration representing about two-third of the mass at assembly complete. To achieve assembly complete and full operability AHM, CSM and ADM are added.

The European Autonomous Habitation Module (AHM)

The SM’s ECLSS is restricted to basic systems. Thus, more functionality in other modules dedicated to habitation and life support has to be included to limit re-supply needs and to meet the strict redundancy requirements of this far-Earth scenario. Therefore, a second module dubbed AHM with temporary autonomous operation is introduced in the design.

Table 5.19: Summary of the "LSS Eve"



Name	Lunar Space Station *Eve*
Purpose	Provide a crewed space research station at lunar libration point 1, which serves as a basis for erecting a Moon surface exploration staging platform.
Crew	3 permanent, 6 temporary
Operation	Automatic/man-tended during assembly phase Nominal crew rotation every 180 day, later up to every 360 days Logistic flights every 120 days when crewed
Orbit	Controlled Lissajous orbit around lunar libration point 1 (LL1)
Flight mode	Inertial/Sun pointing (Z-ASL) nominal
Mass, t	90.5, including Crew Transfer Vehicle (CTV) and Logistics Transfer Vehicle (LTV)
EPS	Total installed: 33.8 kW_e Photovoltaic solar arrays Subsystems max: 16.6 kW_e Payload max: 17.0 kW_e BOL
TCS	Total: 39.5 kW_{th} SM (BM): 4.5 kW_{th} AHM (BM): 5.0 kW_{th} SPP (trackable): 30.0 kW_{th} (SPP truss)
AOCS	Orbit maintained by periodic propulsive manoeuvres Attitude control by CMGs and RCS thrusters
Main propulsion	2 engines on SM (NTO+MMH, 3 kN, 324s specific impulse) 4 engines on AHM (NTO+MMH, 2 kN, 324s specific impulse)
RCS propulsion	32 thrusters on SM (NTO+MMH, 130 N each, 324s specific impulse) 28 thrusters on AHM (NTO+MMH, 220 N each, 324s specific impulse) 16 thrusters on SPP truss (O2+CH4/CH 4 cold-gas, 100 N, 170s/50s specific impulse)
Lifetime	>10 years

Besides a second Russian module (like a rebuild of the Functional Cargo Block, FGB) European hardware could also be of use here. With developing the *Automatic Transfer Vehicle* (ATV) for use as an ISS logistic vehicle, the *Spacelab* laboratory, ISS *Columbus* and the Italian-built *Node* modules, the technology is available in Europe to build a habitation space station module that can operate autonomously for a certain period of time. The ATV spacecraft subassembly is used as service compartment and will provide propulsion, power supply and thermal control. Due to ATV heritage this habitation module is also called *ATV-Hab*. In this connection, the systems must be certified for long orbital lifetime. Furthermore, the service compartment has a pressurised access tunnel with a passive RDS allowing aft docking of logistics vehicles.

The pressurised cargo carrier is to be replaced by a module derived from the European Columbus module that fulfils the associated safety requirements, especially on radiation and debris shielding. The interior offers space for eight International Standard Payload Racks (ISPR), including three individual crew compartments, each the size of one ISPR. The other racks host life support system components, primarily for the atmosphere management (see ECLSS in section 5.5.6) and personal stowage. The main purpose of this module is to provide an environment of improved habitability (i.e. low-noise surroundings, private zones), including three crew compartments for sleeping, relaxing and recreation of the long-term crew.

Altogether, the module is self-sustaining, thus, it can be un-docked from the station to provide a *Safe Haven* for the crew, if the station had to be abandoned in case of emergency. If station control cannot be established again, the crew can leave LL1 and return to Earth with their CTV docked to the aft port of the AHM.

The Life Support Module (LSM) and the Cargo and Storage Module (CSM)

Based on the Columbus and ISS-Node hardware, an additional pressurised compartment is envisioned to provide extended life support devices. When installed, this Life Support Module (LSM) has water management systems for water recovery and enables a crew of 3 a permanent stay at the station with negligible water re-supply. Built-in functionalities include a galley with meeting and recreation space. Later this module will also host “green” systems, i.e. biological systems offering increased food provision quality and synergistic oxygen regeneration, turning this element to the *GreenHab* of the station. Racks transported in the Cargo and Storage Module (CSM) will deliver these ECLSS components. The CSM is then used as stowage space, e.g. equipment, spare parts and solid waste.

Solar Power Platform (SPP)

As autonomous modules, the SM and the AHM have power and thermal systems installed that cope with the demands of the basic on-board systems. To meet the increased power supply and thermal radiation requirements during nominal (full) operation, an additional dedicated element, the Solar Power Platform (SPP) is used which provides centralised EPS/TCS functions. The module consists of two segments, a pressurised compartment with control units and a truss boom with a trackable solar array assembly and radiator panels attached. It bases on Russian ISS hardware currently being developed.

Because the station will fly in inertial flight mode, panel tracking is not necessary during nominal operation but provides safety margins when sun-pointing attitude control is not maintained.

Furthermore, the SPP truss segment is available as an attachment platform for external payloads and the European Robotic Arm (ERA) developed for the ISS can be used for servicing.

The Airlock and Docking Module (ADM)

Initial Extra Vehicular Activity (EVA) capability is provided by the SM's front section, the docking node. Later, however, this section becomes a central access node permitting access to the other main modules. Because a re-pressurization failure in this module is a serious safety concern, design rules dictate usage of an alternate airlock, thus, a dedicated module with airlock capability along with the EVA-associated stowage space is necessary. This module is the Airlock and Docking Module (ADM). It consists of two segments that can be sealed to each other. The first segment is based on the ISS-Docking Compartment (DC) and is used for EVA preparation and provides space for equipment and gas consumables. The second is basically a SM-docking node and serves as the actual airlock. In addition, the ADM provides multiple docking/berthing ports meeting the requirement for the station's further growth. In case of an emergency the orbital modules of the logistics vehicle have also egress/ingress hatches and can also be used as contingency airlocks.

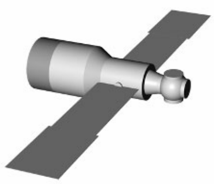
Orbital Stages of Assembly

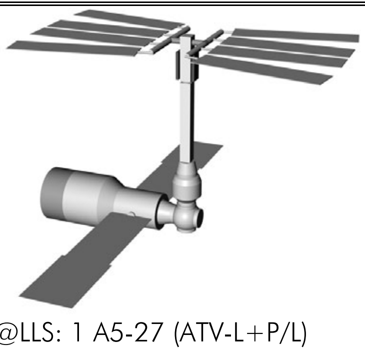
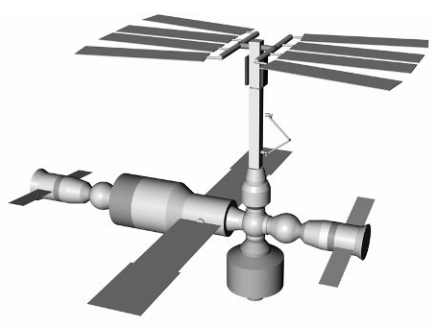
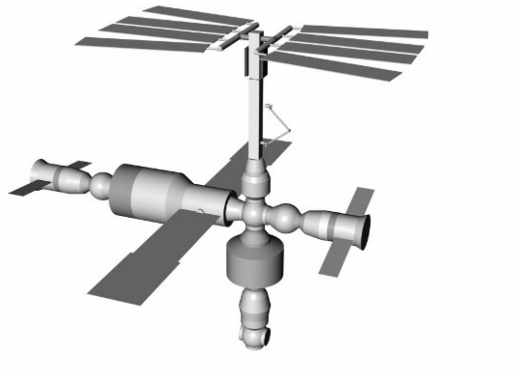
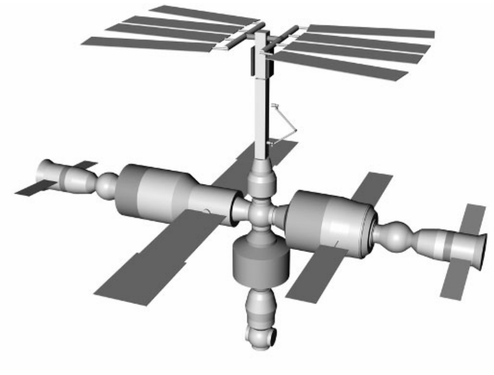
Major constraints concerning the orbital stages of a space station during assembly are:

- Orbital capability including station-keeping, attitude control and telecommunications must be maintained at all times during assembly steps.
- During crewed periods, life support and a crew return option must be available at all times
- Alternative EVA capability must be available.

Furthermore, the demonstration of system maturity has to be considered, e.g. with un-crewed hardware qualification flights. Table 5.20 illustrates the orbital stages and gives an overview of the installed hardware of LSS Eve following the assembly sequence.

Table 5.20: Orbital stages of LSS Eve

#	Station Configuration and Mission Type	Characteristics
Stage 1	 3L-ATV-HD@LL1: 1 Proton (P/L) + 2 A5-27 (ATV-HD)	Service Module (SM, 19.1t) • Station control computer (DMS-R) • Telemetry and Telecommunication • AOCS incl. 860 kg propellant (NTO/MMH) • EVA capability (Russian *Orlan* suit) • Basic ECLSS (0.84t) • 2 crew compartments • Galley and casualty ward
		Initial orbital capability of temporary crew of 3 (IOC-T3C) • Mass: 19.1 tons • Volume: 89 m^3 • Electrical power: 9.8 kWe • Radiative power: 4.5 kWth • Net available docking ports (all passive RDS): 4

#	Station Configuration and Mission Type	Characteristics
Stage 2	 3L-ETM@LLS: 1 A5-27 (ATV-L+P/L) + 2 A5+ (ATV-HD+ETM)	Added: Solar Power Platform (SPP 1+2; 16.2t) • Power/thermal control units, *ERA* controls • Truss boom, batteries, trackable solar arrays and radiator panel • *ERA* manipulator system
		• Mass: 35.3t (accumulated) • Volume: 95 m^3 • Installed electrical power: 29 kWe • Installed radiative power: 34.5 kWth • Net available docking ports: 3
Stage 3	 2L@LSS: 1 A5 (ATV-L + P/L) + 1 A5-27 (ATV-HD) 2L@LTO: 1 Onega (LTV) + 1 A5-27 (ATV-HD) 2L@LTO: 1 Onega (CTV) + 1 A5-27 (ATV-HD)	Added: Life Support Module (LSM, 6.3t) • ECLSS for water recovery (1.1t) and food storage • 1 docking port • 2.7 tons of cargo and equipment (incl. *Strela* robotic arm with station reconfiguration capability) Docked: CTV (8.9t) and LTV (8.9t) • 1 crew compartment in LTV-OM • Emergency EVA capability (LTV-OM)
		Core complete: Permanent 3 crew capability (P3C) • Mass: 59.4t (41.6t w/o vehicles) • Volume: 147 m^3 (133 m^3 w/o vehicles) • Installed electrical power: 29 kWe • Installed radiative power: 34.5 kWth • Net available docking ports: 1
Stage 4	 2L@LTO: 1 Onega (LTV-IM + P/L) + 1 A5-27 (ATV-HD)	Added: Airlock and Docking Module (ADM, 5.9t) • 3 docking ports (3 passive RDS) • Airlock (Orlan) and stowage space
		• Mass: 65.3t (47.5t w/o vehicles) • Volume: 158 m^3 (144 m^3 w/o vehicles) • Net available docking ports: 4
Stage 5	 3L-ATV-HD@LL1: 1 A5-27 (P/L) + 2 A5-27 (ATV-HD)	Added: Advanced Habitation Module (AHM, 18.9t) • ECLSS for O2 regeneration (1.8 t) • 3 advanced crew compartments • 1 docking port • AOCS incl. 2060 kg propellant (NTO/MMH)
		Temporary 6 crew capability (T6C) • Mass: 84.2t (66.4t w/o vehicles) • Volume: 172 m^3 (158 m^3 w/o vehicles) • Installed electrical power: 33.8 kWe • Installed radiative power: 39.5 kWth • Net available docking ports: 4

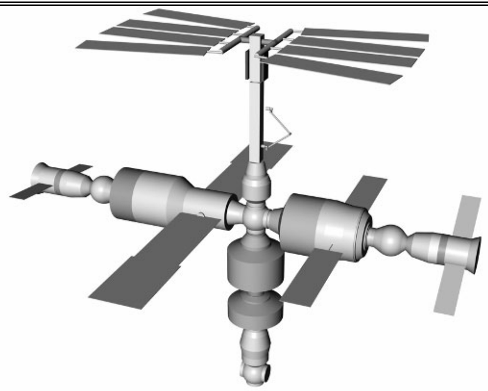
#	Station Configuration and Mission Type	Characteristics
Stage 6	 2L@LSS: 1 A5 (P/L) + 1 A5-27 (ATV-HD)	Added: Cargo and Storage Module (CSM, 6.3t) • Reconfiguration LSM-DM to LSM-CC4-DM • Delivery to LSM: advanced ECLSS for O2 regeneration and food production (800 kg) • 1.9t of cargo and equipment • Storage space (7 m^3)
		Assembly complete of minimum configuration (AC) • Mass: 90.5t (72.7t w/o vehicles) • Volume: 179 m^3 (165 m^3 w/o vehicles) • Installed electrical power: 33.8 kWe • Installed radiative power: 39.5 kWth • Net available docking ports: 4

Figure 5.11 shows the evolution of the gross mass of the station (i.e. the delivered mass to LSS, DMLSS) over the assembly stages and the first re-supply and expedition missions. Furthermore, it shows the number of launches and the specific launch vehicle required for each stage and the evolution of the total mass transported to LEO/ISS. The latter is the initial mass in LEO (IMLEO) required for deploying and initial operation of the station. Key information concerning station assembly and operation are summarised in Table 5.21. Here, the mass ratios μ with respect to the IMLEO are used to indicate the transfer efficiency. The mass ratios are defined as follows:

(74) $$\mu_i = \frac{m_i}{\mathrm{IMLEO}}$$

with m_i being the mass referred to:

- mass delivered to the LSS (index D),
- payload mass to LSS (index P),
- gross mass landed on Earth (index E),
- payload mass returned (index R) and
- waste mass disposed (index W).

It turns out that 19 launches would be required to deploy the station. Three times, namely for the first two stages as for the fifth stage, A5 double-launches are necessary. This means that two consecutive launches must be performed in the time interval of four days to maintain a nominal mission duration of 15 days of the ATV-HD. Assuming a launch rate of five Ariane 5 rockets per year (i.e. a launch every 10 weeks in average) three years would be sufficient for achieving assembly complete.

Assuming two crewed and two logistics missions per year Figure 5.11 shows the first year of operation of the station after assembly complete. Thus, a launch rate of 4 launches is proposed for both the Ariane 5 and the Onega launcher. Table 5.22 lists a summary of information for this phase. The mass ratio of the gross mass delivered to the LSS is 23% with a payload mass fraction of 5%. A mass fraction 2% of the launched mass into LEO would return to Earth.

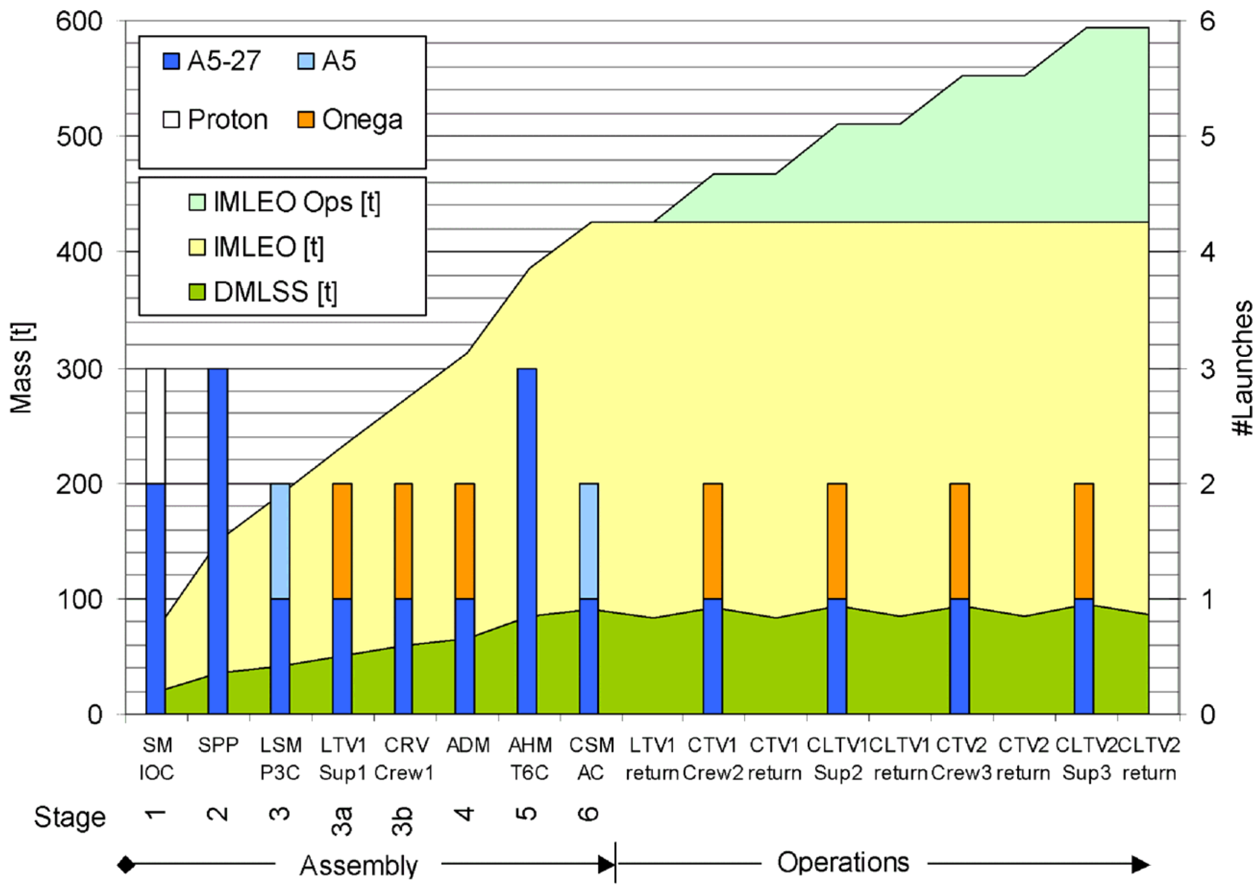


Figure 5.11: Mass evolution of LSS assembly and operation

Table 5.21: LSS assembly statistics

Issue	**Value**	**Remarks**
LSS modules delivered	7	excluding CTV and LTV
Crewed missions	1	one way (CTV remains at station)
Logistics missions	1	LTV
Mass launched to LEO/ISS [t]	426.1	IMLEO due to assembly and initial operations
Mass delivered to LL1/LSS [t]	90.5	μ_D ranges from15% for 2L (ADM) to 26% for 3L-ATV-HD missions (21% average)
Mass returned [t]	3.1	gross landed mass (CTV-DM) incl. P/L, μ_E=7.7%
	0.23	returned payload mass (CTV-P/L down) , μ_R=0.6%
	1.5	disposed waste mass (LTV-P/L down), μ_W=3.7%
Infrastructure elements required	10	ATV-HD
	1	ETM
	4	ATV-L
Launches required	15	Ariane 5 (incl. 13 Ariane 5-ECB+)
	1	Proton
	3	Onega
	19	total launches

Table 5.22: LSS operations statistics after assembly complete

Issue	Value	Remarks
Crewed missions	2	Cryo-CTV
Logistics missions	2	Cryo-LTV
Mass launched to LEO/ISS [t]	168.2	IMLEO due to operations after assembly complete
Mass delivered to LL1/LSS [t]	39.4	μ_D ranges from 23% for Cryo-CTV to 24% for Cryo-LTV missions (23% average)
Payload delivered to LSS [t]	8.42	μ_P=5% (nominal P/L-up of Cryo-CTV and Cryo-LTV)
Mass returned [t]	7.2	gross landed mass (2 Cryo-CTV-DM) incl. P/L, μ_E=8.9%
	1.6	returned payload mass (2 Cryo-CTV-P/L down), μ_R=2%
	4.3	disposed waste mass (2 Cryo-LTV-P/L down), μ_W=5.2%
Infrastructure elements required	4	ATV-HD
Launches required	4	Ariane 5-ECB+
	4	Onega
	8	total launches

5.5.6 Life Support System Layout

All inhabited spacecraft implicitly need an Environmental Control and Life Support System (ECLSS). Figure 5.12 illustrates the different areas of ECLSS and their principle interactions herewith. The ECLSS also has to make sure that crew safety is maintained at all times. For forthcoming spaceflight missions beyond LEO and thus outside Earth's magnetic shield, radiation becomes much more important. Especially the potential exposure to *solar flare radiation* must be taken into account and "storm-shelters" will be mandatory for long-duration missions. Because one possibility is the arrangement of hydrous substances around the crew's sleeping compartments [Zubrin1996], one can cope with this problem by implementing proper internal structural design. This is not discussed here any further.

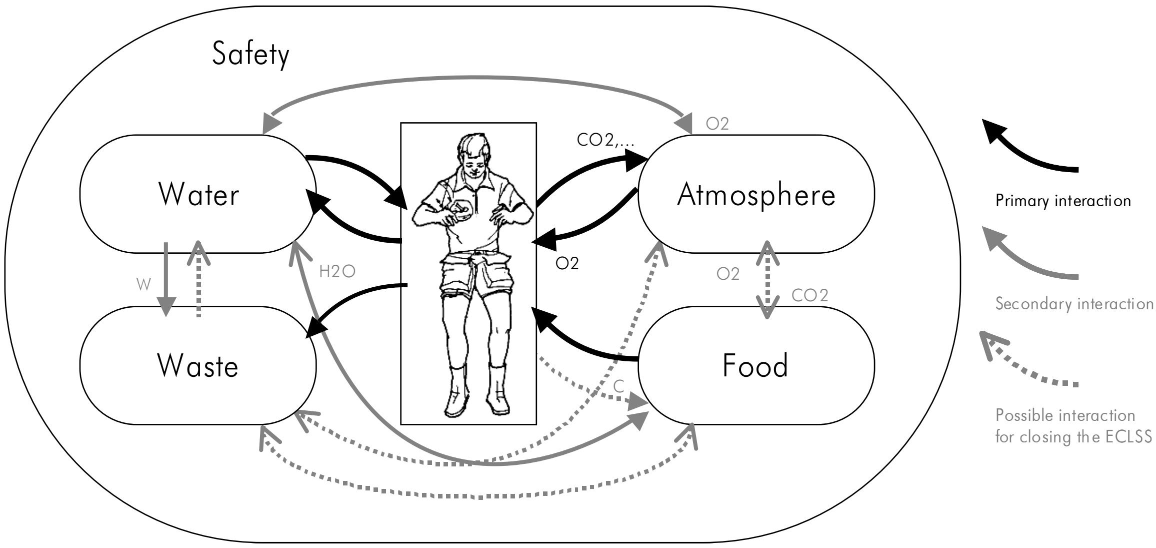


Figure 5.12: Principle interactions within ECLSS

Table 5.23 gives an overview of the re-supply needs of the ISS during the reduced operation period with a crew of two. While the second column states the original values, the third scales to a crew of 3. Thus, a space station with a crew of three with the same ECLSS would need more than 6.3 tons ECLSS re-supply per year.

Table 5.23: ECLSS re-supply mass of ISS per year (values based on Progress logistics flights between February 2003 and August 2004 to ISS-11A with a permanent crew of 2 resulting in approx. 4 flights per year; unit of values with *: kg/crew/day) [NFM2003/03] [NFM2003/10b] [NFM2003/13b] [NFM2004/02] [NFM2004/14]

Transportation mass to ISS [kg]	Total mass [kg] (Crew of 2)	Total mass [kg] Scaled to 3	Mass [kg] per day	Remarks
System consumables	163.2	163.2	0.5	Air and oxygen
Subsystem consumables	1830.4	1830.4	5.1	Thermal and gas conditioning, power and water supply, sanitary and hygienic facilities
Food	892.8	1339.2	1.2*	
Individual ECLSS items	817.6	1226.4	1.1*	Individual medical and hygienic facility incl. underwear
Documentation	113.6	170.4	0.2*	Parcels, tapes, etc.
Water	1064.0	1596.0	1.5*	
Summary	4881.6	6325.6		

Past and current ECLSS in operation like on Mir and ISS only facilitate basic recycling capabilities (e.g. regeneration of small portions of waste water, i.e. condensation to potable water). They are, except minor in-space experiments (e.g. mini-greenhouse *Lada* on ISS [NFM2003/13a]), restricted to physical-chemical systems. The ECLSS of the envisioned space station is seen as an important step regarding development of semi-closed eco-systems for deep space exploration missions. Thus, it must include not only advanced physical-chemical but biological components as well. Such a hybrid design becomes evidently necessary when logistics constraints and re-supply requirements of inhabited stations far-Earth have to be met. Due to the high effort (and costs) of re-supply, the flow of the substances on-board has to be closed as much as possible. The largest mass savings are due to water recycling and carbon dioxide (CO_2) removal and oxygen (O_2) generation [Messerschmid2000]. Compared to an open system, closing the water cycle saves 55% of ECLSS re-supply mass, using regenerative CO_2 removal techniques saves another 15%. Closing the oxygen cycle by generating oxygen by reducing CO_2 further decreases the re-supply mass by about 10%. Applying all these, one can save about 80% of ECLSS related re-supply. Major parts of the remainder are food for the crew, clothing and finally cabin gases to be refilled due to leakage.

Besides the transport up to the station, important transportation needs arise due to removal of waste generated on-board. Waste water is, if not recycled, the largest amount followed by solid wastes including laundry, packaging, faeces and food residuals. Thus, one can save transportation costs by some kind of waste treatment and recycling on-board. Clothing, for example, can be washed and re-used effectively if washing water recycling is available. Theoretically, the organic wastes could be used as nutrients for plants. This will be of particular interest for long-term scenarios and planetary surface bases, but today, common in-space techniques for plant growth and harvesting are relatively complex, too heavy and have not nearly reached operational maturity yet. Thus, for near-term inhabited space installations other options must be considered. Comprehensive surveys of ECLSS technologies are available from

ECKART [Eckart1993], MESSERSCHMID [Messerschmid2000] and GANZER [Ganzer2004], with the latter containing the basis and details of the ECLSS of this design.

ECLSS of this design example

Figure 5.13 illustrates the designed ECLSS for the envisioned LSS. It shows all principal components and the flow of the individual substances between them and thus visualises the interactions within the ECLSS. The main elements are also listed in Table 5.25 with remarks on their characteristics, including the technological readiness level (Table 5.24) as well as their installed location, mass and volume. The tanks are distributed over the modules and are not listed individually here. It shall be remarked that high-priority tanks for contents such as water, waste water and solid waste are included in both autonomous modules, namely SM and AHM. This insures autonomous operation of these modules in case of emergency. Analogously, tank sizing is performed with the constraint of enabling emergency operations of at least 24 days for a crew of six. Most critical is the water management subsystem due to the large amounts associated with it. The total mass of the station's ECLSS at assembly complete is approximately 4.26 tons with a total required volume of 12.6 m^3. Therefore, the ECLSS must be launched in stages and installed in various station elements as stated in the table. Thus, the design must be modular and also operable at intermediate orbital stages.

Table 5.24: Definition of the Technology Readiness Level [TRL]

TRL	TRL definition	TRL	TRL definition
1	Basic principles observed	6	System/subsystem model or prototype demonstration in relevant environment
2	Technology concept and/or application formulated	7	Subsystem prototype in a space environment
3	Analytical and experimental critical function/proof-of-concept	8	System completed and flight qualified through demonstration
4	Component and/or breadboard validation in lab	9	System flight proven through mission operations
5	Component and/or breadboard in relevant environment		

ECLSS refinements during assembly progress

Initial ECLSS capabilities rely on the basic functionalities provided by SM. Because neither water, nor oxygen regeneration is available at this stage, crew stay times are restricted to available resources on-board. SM's electrolyzer *Electron* and traditional oxygen candles using lithium-perchlorate (LiClO4) secure oxygen generation and CO_2 removal is provided by lithium-hydroxide (LiOH) cartridges. Potable water brought up with the SM and the first LTV flight is used by the crew with storing the waste water in tanks of SM.

With delivery and activation of LSM the station water recovery system becomes available, which includes VPCAR, AES and MilliQ and is capable of generating potable water for a full crew of three plus an additional three persons during subsequent crew rotations. Thus, transportation of water becomes unnecessary and re-supply mass reduces significantly.

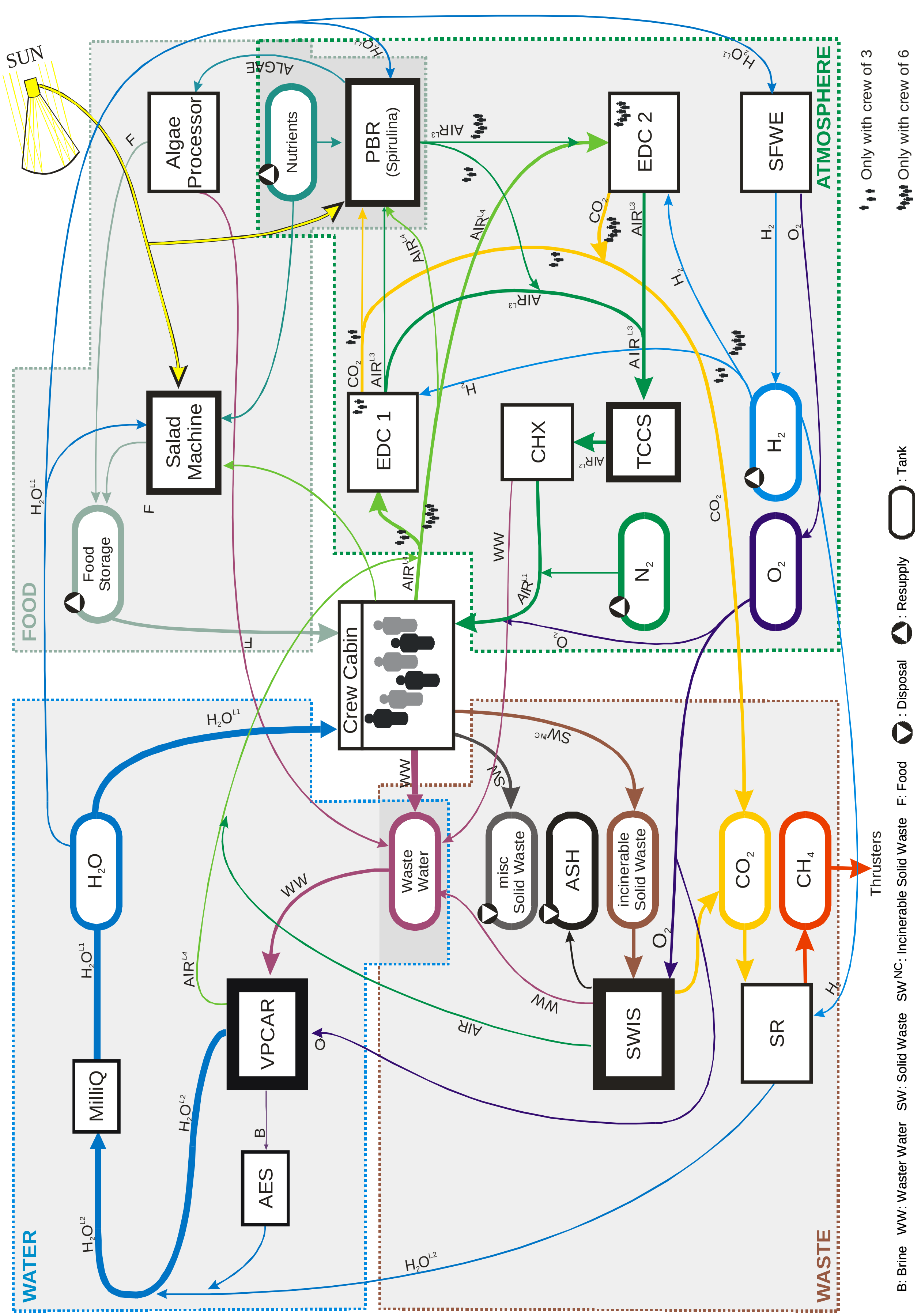


Figure 5.13: ECLSS schematic overview (thickness of frame indicates relative weight)

Table 5.25: Main ECLSS components of LSS Eve (*: components delivered by CSM)

	Element	Task	Module	Mass [kg]	Volume [m³]	TRL
Atmosphere	Static Feed Water Electrolysis (SFWE)	O2 generation; produces H2 (+) light-weight, simple; water as primary (-) cleansing required	AHM	200.00	0.08	8
	Electro-chemical Depolarised CO2 Concentration (EDC)	CO2 removal (+) flexible op's, low power; synergism regarding power (-) complex, cooling required	AHM	89.00	0.14	5
	Trace Contaminant Control System (TCCS)	Trace contaminant removal	SM, AHM	85.00	-	9
	Condensing Heat Exchanger (CHX)	Humidity and temperature control	SM, AHM	-	-	9
	Photo-Bio-Reactor (PBR)	Using algae *Spirulina Platensis*; O2 regeneration; algae production	LSM	160.00	0.30	5
	Lithium Hydroxide (LiOH) cartridges	CO2 removal; used in vehicles and in case of emergency (+) simple, reliable (-) not reusable, re-supply	All	148 up 201 down (crew of 3/ 30 days)	0.5	9
	Oxygen candles (LiClO4)	O2 generation; used in vehicles and in case of emergency (+) simple, reliable (-) not reusable, re-supply	All	243 up 171 down (crew of 3/ 30 days)	0.11	9
Water	Vapour Phase Catalytic Ammonia Removal System (VPCAR)	Ammonia removal using membrane (similar TIMES) (+) very high water quality, high efficiency, light-weight, static op's (-) high energy, membrane replacement	LSM	283.00	-	3 /4
	Air Evaporation System (AES)	Cleansing using evaporation and felt beds (+) high efficiency, simple, reliable (-) high power, bed replacement	LSM	75.00	-	5
	MilliQ	Filtration and UV purification (+) simple, reliable (-) bed replacement	LSM	100.00	-	5
Waste	Sabatier Reactor (SR)	CO2 reduction and H2O and CH4 production (+) high efficiency, reliable; synergism regarding propellant production	AHM	43.00	-	7
	Solid Waste Incineration System (SWIS)	Incineration, production of CO2 and ashes (+) high efficiency, simple, reliable (-) high temperatures (≤ 1000°C), post-treatment of products	LSM*	150.00	-	5
Food	Algae Processor	Algae treatment for food substitution	LSM*	10.00	-	N/A
	Salad Machine	Production of various vegetables	LSM*	200.00	1.90	5 / 6
	Storage	Primary food storage at SM, LSM	All	91.00	0.23	
	Galley		SM, LSM	120.00	-	9
Misc.	Solar Collector	Concentration and distribution system of natural sun-light to PBR and Salad Machine (+) high efficiency	LSM	42.00	-	7
	Fire protection		All	70.00	-	9

	Element	Task	Module	Mass [kg]	Volume [m³]	TRL
	Hygiene facility		SM, AHM	120.00	-	9
	Casualty wage (Sick bay)	Medical treatment of diseases of various kinds	SM	60.00	-	-
	Piping, ventilation		All	672.00	-	9
	Tanks of various contents	Gases (O2, H2, N2, CO2, CH4), fluids (potable water, waste water) and solids (e.g. clothing, algae nutrients and sorts of waste)	SM, AHM, LSM	1690.00	10.00	7-9

Then, the AHM provides advanced physical-chemical devices for regenerative CO_2 removal and O_2 generation including EDC, SFWE and the *Sabatier* reactor. The station's recycling capability is hence complete. Except in the case of leakage, no transport of atmospheric gases is necessary.

Finally, the biological ECLSS add-ons PBR and the *Salad Machine* [SaladMachine], which are to be delivered by CSM, reduce the yearly food re-supply mass by 285 kg (nominal operation, i.e. three crew members, two crew rotations per year) by providing on-board produced edibles including algae-based proteins and vegetables. Furthermore, SWIS heavily reduces the need for transportation of combustible wastes back to Earth. Water as generated effluent enters is fed into the waste water loop and the filtered CO_2-rich exhaust gas undergoes the same treatment as cabin air.

Table 5.26 lists the reduced ECLSS related re-supply from assembly complete on. The amount of system and subsystem consumables is a conservative estimate equal to the ISS, although mass and size are much smaller. Nevertheless, the total re-supply mass of ECLSS related consumables is 1726 kg less than compared to scaled ISS values (compare Table 5.23).

Table 5.26: ECLSS re-supply mass per year of LSS Eve at assembly complete
(Unit of values with *: kg/crew/day; ⁺: values taken from ISS, compare Table 5.23)

Transportation mass [kg]	Total per year	per day	Remarks
System consumables	164.0⁺	0.45	Air and oxygen
Subsystem consumables	1830.0⁺	5.01	Thermal and gas conditioning, power and water supply, sanitary and hygienic facilities
Misc. consumables	54.0	0.15	Hydrogen for CO_2 reduction with SR
Food	1054.2⁺	1.0*	
Misc. Food	101.0	0.1*	Nutrients for PBR algae
Individual ECLSS items	1226.0⁺	1.12*	Medical and hygienic items incl. underwear
Documentation	170.0⁺	0.16*	Parcels, tapes, etc.
Water	0.0	0.0*	
Summary	4599.2		

Table 5.27 lists the disposal masses to be transported from the station and burned up in Earth's atmosphere. Here the worst case is assumed, i.e. the assumption that principally every ECLSS related mass brought up must be returned, except if it is lost on-orbit. Not listed and not-recycled masses are:

- A surplus of 80 kg water per year stored for redundancy (metabolic produced)
- 396 kg methane per year produced by SR; may be used for AOCS
- Around 85 kg per year of personal documentation items is assumed of being returned (50% of brought-up mass).

Table 5.27: ECLSS disposal mass per year of LSS Eve at assembly complete
(Unit of values with *: kg/crew/day; ⁺: values taken from ISS, compare Table 5.23)

Transportation mass [kg]	Total per year	per day	Remarks
Subsystem consumables	1830.0	5.01	See Table 5.26
Solid waste	157.0	0.33*	Non-incinerable and ash,
Individual ECLSS items	1226.0	1.12*	See Table 5.26
Misc. waste	85.0	0.15*	Used documentation items
Summary	3298.0		

5.5.7 Station Operations

Transfer Mission Modes

The term "mission mode" identifies the approach of performing the actual spaceflights with respect to particular mission objectives including the sequence of events. A set of mission modes makes up the mission architecture of a specific scenario. A spaceflight scenario like ELLIPSE makes multiple mission types necessary: heavy and medium-lift assembly missions, cargo and crew logistic missions. This section demonstrates how the missions are performed and gives a description of representative examples. Depending on the design of the vehicles, various options or variations of options during mission design exist:

- Number of stages
- Time of staging and payload separation
- One-way/two-way usage of vehicles

It turns out that splitting crew and cargo transport and minimizing the mass of the return vehicle are decisions that positively influence the overall transfer efficiency. This is expressed by following rules for mission mode design:

- Minimizing the Initial Mass in LEO (IMLEO).
- Minimizing the number of launches required.
- Minimizing the number of different infrastructure elements required.
- Maximizing the Delivered Mass to LSS (DMLSS).

The latter is the main mission purpose and the first three are the main cost driver of a mission. Thus, these issues offer the major evaluation criteria for mission mode selection.

The following three figures depict the optimised mission modes of the ELLIPSE scenario. Beginning at the bottom, the distance from Earth increases with passing LEO/ISS, LL1 and LLO and ultimately reaches the lunar surface, where the mission time increases from left to right.

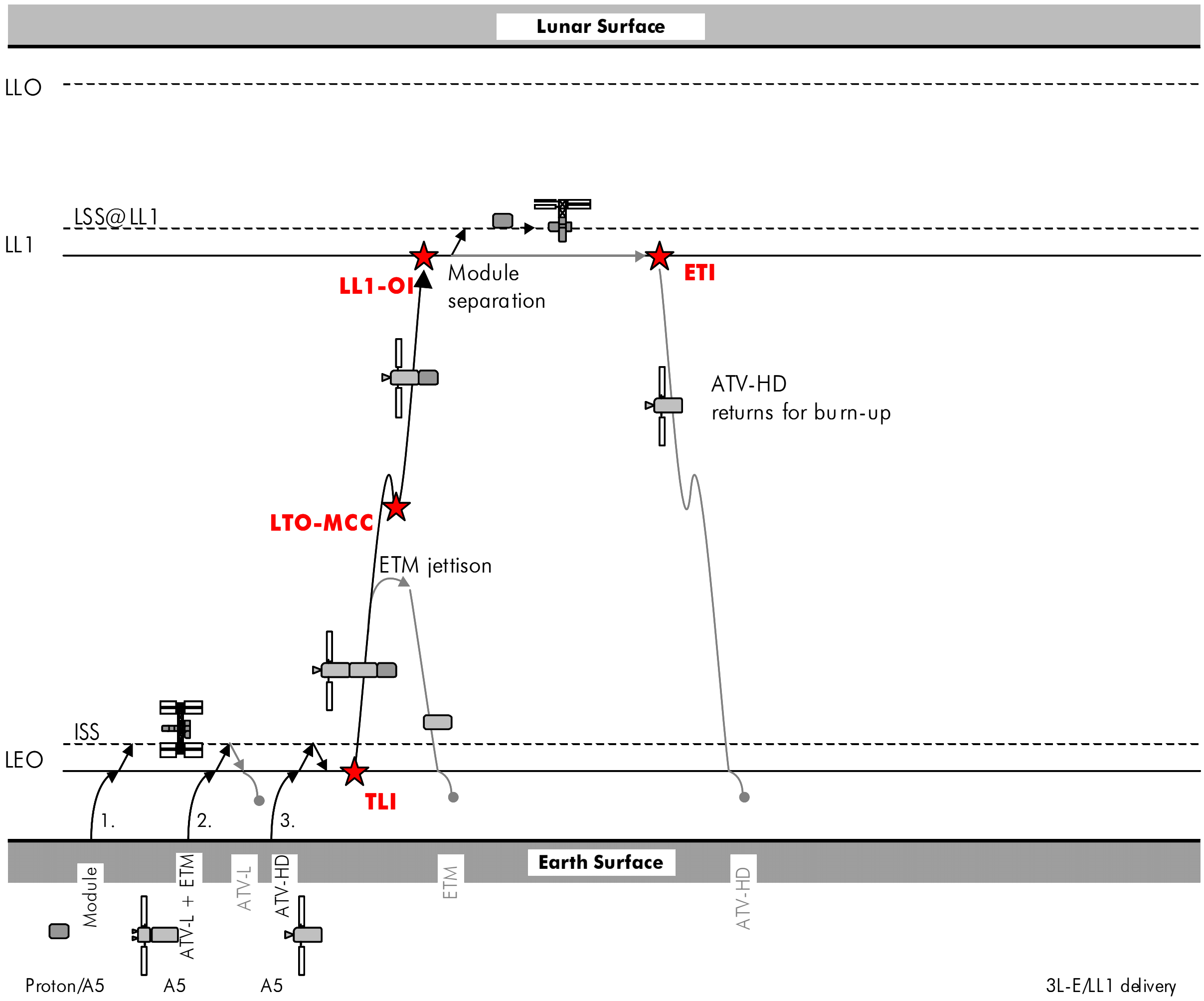


Figure 5.14: Mission mode for heavy payload assembly flights (3L-ETM/ATV-HD, LL1-delivery)

Figure 5.14 depicts the mission mode of an assembly mission delivering heavy payloads to the Lunar Space Station (LSS) at the Lunar Libration point 1 (LL1). This mission type requires three launches (3L) to the ISS; one for the LL1 module and two for delivering the transfer stage and the propellant module. With this sequence using an ATV-HD and an ETM, the maximum payload mass (gross mass of the module) is about 17 tons (3L-ETM) delivered to LL1. Alternatively the ETM can be replaced by a second ATV-HD with which the maximum payload mass to LL1 increases to 19.5 tons (3L-ATV-HD, Table 5.13). The latter is needed for the large SM and AHM modules. However, the other 3L missions will use ETM due to cost savings by manufacturing an ATV-HD instead of the ATV-L necessary for ETM delivery to the ISS.

The two-stage mission sequence shown in Figure 5.14 implies that the LSS payload module has an autonomous flight capability for rendezvous and docking to the ISS and the LSS at LL1, similar to SM and AHM. Otherwise, firstly an additional ATV-L is necessary to deliver the payload module to the ISS and secondly, the module separation would not take place. Alternatively, the ATV-HD can complete the transfer by delivering and docking to the LSS.

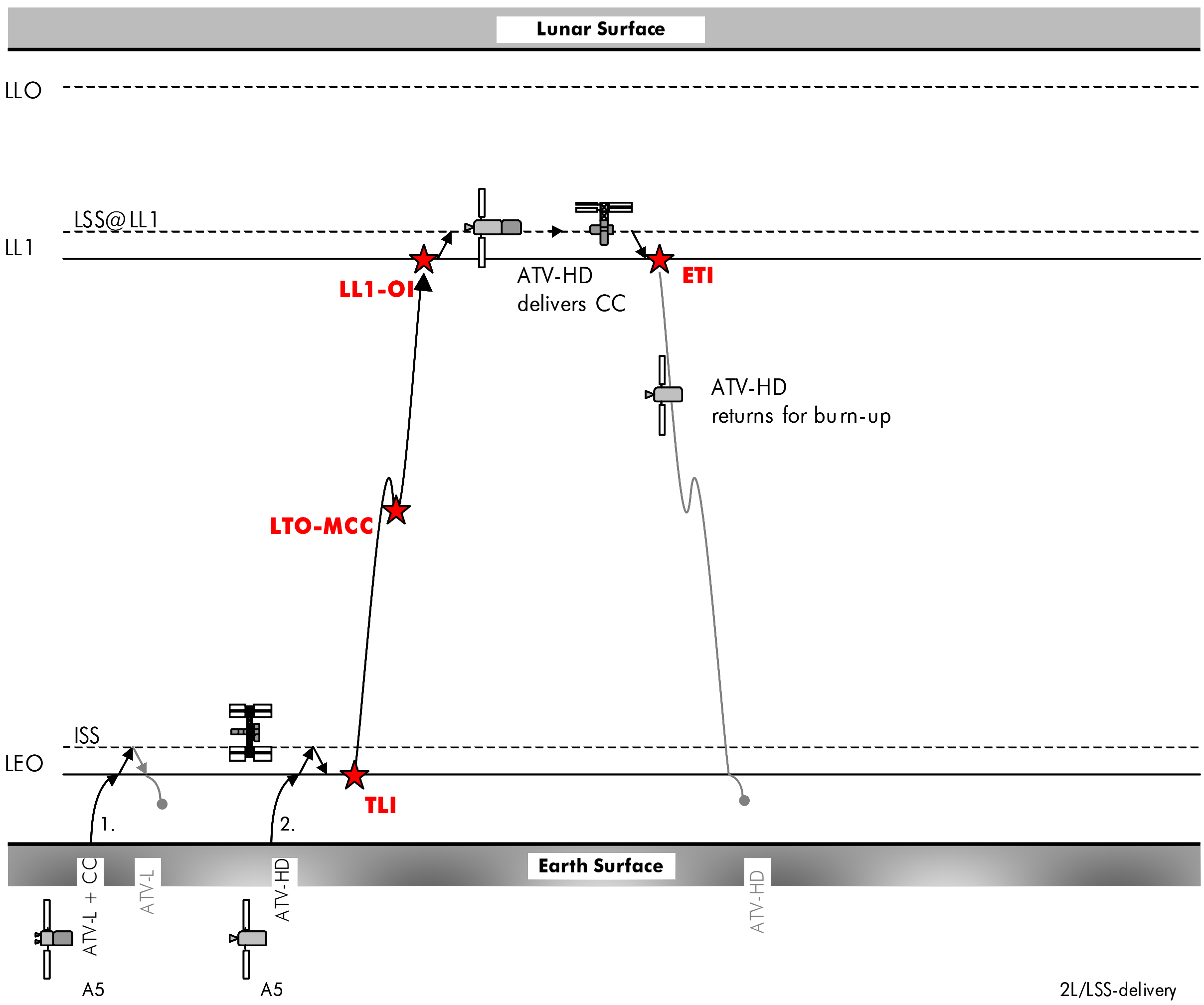


Figure 5.15: Mission mode for light payload assembly flights (2L, LSS-delivery)

Figure 5.15 illustrates the mission sequence used for light payload masses in a two-launch (2L) to the ISS, one-stage mission. The passive payload module (typically a cargo carrier/compartment, CC) is then delivered in advance and docked to the ISS by using an ATV-L vehicle. Then, an ATV-HD carrier is launched and docked to the payload module at the ISS, undocked and then injects itself into the lunar transfer orbit (LTO). After mid-course correction (MCC) and insertion into LL1 orbit, the payload module is delivered to LSS by automatic docking. Afterwards, ATV-HD undocks and injects into the Earth-transfer orbit (ETO) and burns up in the Earth's atmosphere during direct re-entry. The maximum gross mass to LSS delivered this way is 6.4 tons.

Crewed and logistics missions begin rather similarly to the mission mode above, but evolve in a two-stage scenario. As depicted in Figure 5.16, these missions start with the launch of an ATV-HD transfer stage. After a relatively short time, the ship, i.e. the crewed (CTV/Cryo-CTV) or automatic vehicle (LTV/Cryo-LTV; only the crewed scenario is shown), is launched to LEO and docked with the ATV-HD. Following, the two mated vehicles are injected into LTO by using the ATV-HD, which is separated from the ship after eventual mid-course corrections. While the ATV-HD is disposed of by destructive re-entry, the ship continues with its own propulsion system and inserts itself into the LL1-orbit and rendezvous and docks to LSS.

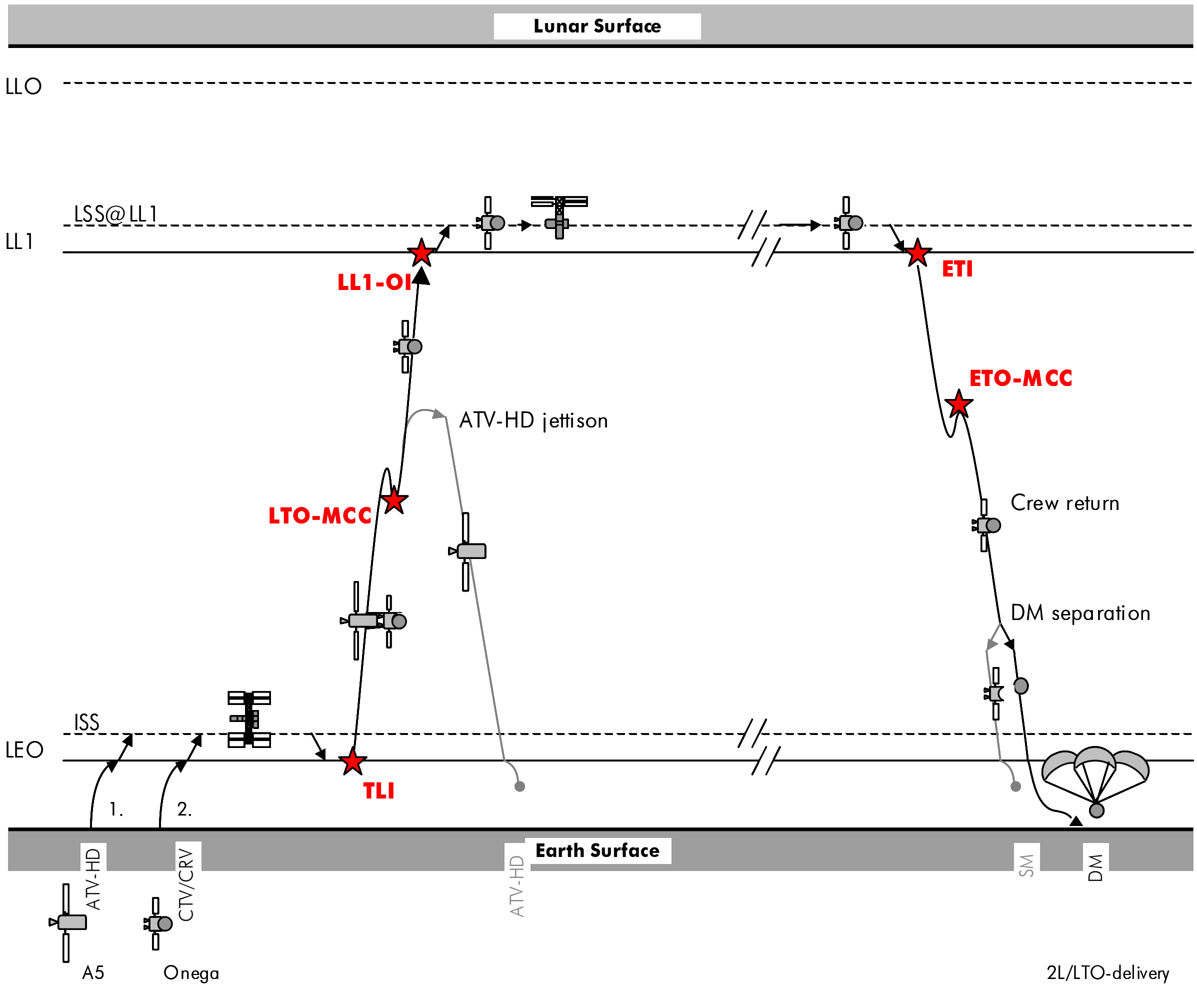


Figure 5.16: Mission mode for crewed and logistics flights (2L, LTO-delivery)

Crew return or waste disposal is then performed by the same vehicle, which departs from the station and performs ETI for Earth re-entry. In the case of crewed flights, the descent module (DM) separates before and lands after aero-braking on Earth's surface. The maximum LTO-delivery mass of the ATV-HD, i.e. the maximum gross mass of the CTV/Cryo-CTV or LTV, is 13.6 tons. Using a Cryo-LTV, the nominal net LSS-payload is 3.8 tons up and 2.15 tons down (see Table 5.16 and Table 5.17).

Re-supply and Logistics

Table 5.28 summarises the re-supply mass to the station and compares the values with the ISS re-supply needs. ISS values here refer to the time period with grounded US Shuttle fleet and restricted station operability and a reduced permanent crew size of two persons (February 2003 to May 2005). The comparison reveals that during this time period, the ISS logistics mass can be assumed to be 30% higher. With the nominal cargo-up capacity of the logistics vehicles LTV, only three regular flights are necessary instead of four (or rather five when scaled up to a crew of 3) in the case of the ISS. The total number of launches, however, is higher for LSS because two launches are necessary for each mission (2L). Taking advanced Cryo-LTV vehicles, the number of missions decreases to two each carrying 3800 kg up to the station. Then, also the total number of launches is four, thus, equal to ISS bound logistics launches or rather fewer than ISS requirements with a crew of three.

Analogously Table 5.29 summarises the mass transported from the LSS for disposal. This verifies that three logistic missions are sufficient to support LSS operation with conventional LTV type using storable propellants. Using Cryo-LTVs, as above, two missions are sufficient, each carrying 2150 kg of waste away from the station.

To conclude, logistics operation of the LSS is feasible, even with conventional currently available technology. Enhancing the existing vehicles with cryogenic technology would require only two logistic flights per year and therefore even undercut ISS-based requirements.

Table 5.28: Yearly re-supply mass LSS versus ISS (values with $^{+}$ scaled from ISS; ISS values based on Progress logistics flights between February 2003 and August 2004 to ISS-11A with a permanent crew of 2 resulting in approx. 4 flights per year [NFM2003/03] [NFM2003/10b] [NFM2003/13b] [NFM2004/02] [NFM2004/14]

Transportation mass [kg]	Total ISS (crew of 2)	Total ISS (scaled to 3)	Total LSS	Remarks
ECLSS items	4881.6	6325.6	4599.2	See Table 5.23 and Table 5.26
System equipment	1535.2	1535.2	1500.0^{+}	On-board systems upgrade/maintenance
Propellant	2181.6	2181.6	1140.0	Refuel
Re-boost	1000.0	1000.0	0.0	Orbit control, drag compensation
Science instruments	284.8	284.8	300.0^{+}	
Summary	9883.2	11327.2	7540.0	
Vehicle cargo capacity	2500.0	2500.0	2740 3800	LTV/Cryo-LTV nominal capacity (up)
No. of vehicles	4	5	3 2	
Average cargo mass	2470.8	2265.4	2200 3770	

Table 5.29: Yearly disposal mass LSS (values with $^{+}$ scaled from ISS; see Table 5.28)

Transportation mass [kg]	Total LSS	Remarks
ECLSS items	3298.0	See Table 5.27
System equipment	700.0^{+}	Un-used on-board systems and equipment
Science instruments	100.0^{+}	Un-used science instruments
Summary	4098.0	
Vehicle cargo capacity	1500 2800	LTV/Cryo-LTV nominal capacity to LSS (down)
No. of vehicles	3 2	
Average cargo mass	1433 2150	

5.5.8 Station Growth and Utilization Options

With assembly complete of the designed minimum-configuration lunar space station described above, the actual design task of the mission statement is performed. However, the minimum-configuration space station was to be investigated as an intermediate step towards subsequent space exploration and exploitation activities (sections 5.1), namely a) the station's role as gateway and safe haven for lunar surface exploration missions, b) the role as a research platform utilizing the specific space environment beyond LEO and as an advanced test and verification platform (section 5.5.2). In this section, a brief outlook on these activities is presented and sample station growth options are outlined.

a) Lunar Exploration Missions based on the LSS

The implementation of the designed space infrastructure above, including the LSS and its modules, the transfer and other vehicles, and ISS as LEO support, opens up a new opportunity for lunar surface exploration missions, i.e. a fully reusable spacecraft that is based at the LSS at the Lunar Lagrange point 1 (LL1). The resulting space station of this sample scenario can subsequently serve as an efficient gateway to the Moon, providing safe and low-cost global access to the lunar surface.

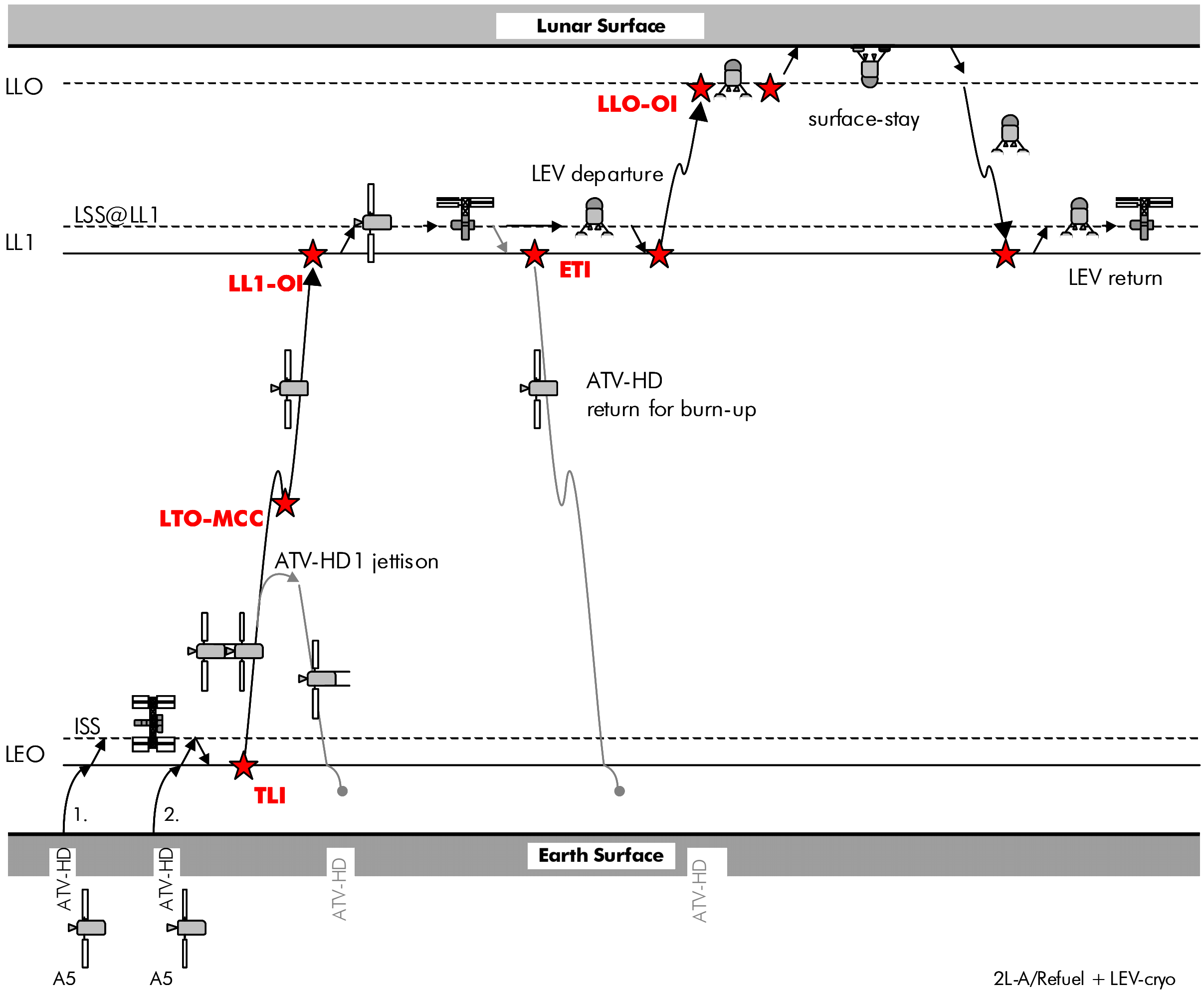


Figure 5.17: Crewed missions to the Moon's surface using LSS-gateway

Parametric assessments based on past designs and developments, such as the US *Apollo* and the *LK/Block E* projected during the lunar landing programme of the former Soviet Union, reveal that such a re-usable *Lunar Exploration Vehicle* (LEV) is presumably feasible. This ferry based on the LSS at LL1 could repeatedly bring a crew of two to anywhere on the surface of the Moon and bring them back to the station. Merely the propellant necessary for these excursions would have to be delivered to the station. The associated mission mode is illustrated in Figure 5.17. Here two of the three permanent station crewmembers use the LEV to land on the Moon's surface and return back to the station. Thus, only two launches from Earth are necessary to carry out one lunar excursion, enabling missions to various sites in a regular fashion. Thus, most of the surface activities requested could be performed:

- Deployment of scientific instruments
- Collection of samples
- Maintenance of surface installations
- In-situ investigations on resource utilisation and exploitation
- First lunar base survey and scouting

Therefore, the LL1-gateway architecture obviously offers significant advantages compared to other current envisioned scenarios (e.g. [Hovland2004], [CDF23A]), which typically use three or more heavy-lift launches plus one crewed launch for one surface mission. The benefits here are not only of financial kind but can also be seen from the operational and safety point of view. However, missions for delivery of heavy payloads *to lunar surface* will be better performed by direct missions not using the LL1-gateway.

Besides development, construction and delivery of the LEV, steps towards implementation of such missions will also include extension of the station in terms of additional modules in order to support lunar surface operation. At least one, more likely two additional modules will be necessary, i.e. a stowage module for hosting crew surface activity equipment (suits, drills, etc.) and sample containment. There will be also a need for a laboratory for on-board sample analysis and classification (compare with next section) and equipment to allow for LEV refuel operations, for which the ADM could be upgraded.

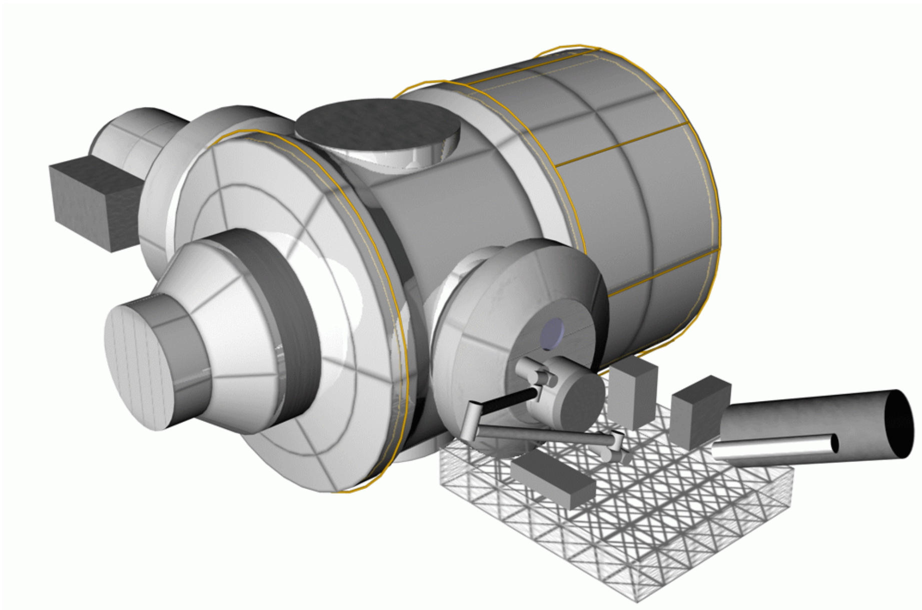


Figure 5.18: Multi-purpose Lab-Node with exposed platform and airlock attached

b) Space research and engineering platform

Research of the cis-lunar space environment as well as engineering tests and verification activities on-board LSS could be of great value for preparation of future lunar base erection and Mars exploration programmes. In addition to the minimum-configuration, these activities will make additional hardware and facilities necessary. This could include a range of various instruments and equipment. At least one dedicated laboratory module will be necessary. A minimum solution allowing for synergy in terms of flexibility could be a Laboratory Node Module (LNM) or *Lab-Node* for short (Figure 5.18). This module, derived from the Italian built ISS-Node

2, could host microgravity experiments, ovens, and other research equipment. As shown in the figure, further hardware could be attached to this module, such as an additional external platform for exposing samples to cis-lunar space and operating specific observation instruments externally with tailored interaction possibilities and better suited than those on the SPP-truss (which is primarily suited for long-term experiments and samples directly exposed to the Sun's irradiation). Furthermore, a dedicated airlock module could complete the station's equipment.

Of course, other potential utilization scenarios with their associated hardware exist, such as testing inflatable structures (such as presented in [TransHab] and [Bigelow]) beyond LEO for instance. Many experiments are thinkable, e.g. laser communication over very long distances, etc. Time will show how manifold humans will utilise newly available opportunities.

5.5.9 Re-use of ISS elements

With the presented concept of a lunar space station, there is of course the possibility of using original station elements and equipments, which are already in orbit at the ISS. This option is in particular interesting if one takes into account that the ISS might not be needed in its complete configuration at the time when the LSS assembly starts.

An obvious example is the Service Module. The Zvezda module could be upgraded with modern technology where necessary and could be transferred using the transfer vehicle designs. Other analogously re-usable modules include the SPP, DC, and eventually utilisation modules, including Destiny, JEM, Columbus and/or Russian modules to be installed meanwhile. Using this approach the number of launches and IMLEO can be decreased. Construction time and cost would be significantly reduced.

5.5.10 Concept Conclusion

The design task formulated with the sample mission statement and performed with the resulting lunar space station scenario is one application example, demonstrating the conceptual design, modelling and simulation tools developed during the underlying work. While most approaches to lunar exploration architecture choose mission scenarios departing from Earth or LEO and landing on the lunar surface, the results of this investigation reveal that a Lunar Space Station at the LL1 can offer various benefits in the current situation of human spaceflight because it:

- Generates experience with human spaceflight beyond LEO.
- Enables demonstration and verification of key technologies in (semi) deep space.
- Enables exploration of the Moon globally and systematically.
- Is feasible and affordable mid-term by using existing or tailored technology and hardware.
- Fosters new models for international cooperation, i.e. robust ESA-Russian element embedded to a global lunar surface exploration programme led by the USA.

Thus, space stations beyond LEO do not seem to be too farfetched; and they are definitely not *science fiction* but rather the logical next step on the way of humans into space. Like a stepping-stone, they could enable immediate steps of sustained development of outer space with technologies available today. When thinking of human space exploration and systematic exploration of the Moon, it is therefore absolutely reasonable to take mission scenarios into account that involve crewed cis-lunar space infrastructures.

6 Conclusion and Outlook

The research work documented with this dissertation deals with the conceptual design of space missions beyond low Earth orbit. Its primary objective was extending the methodology and tools developed in the framework of the *Space Station Design Workshop* (SSDW). With an advanced methodology and tools for generating, analysing and evaluating space mission and space system concepts in terms of feasibility, technological requirements, utilization, operations and cost, the presented results addressed nearly all fundamental components of systems engineering.

As a result, this research work integrated mission design aspects into the conceptual space station design process, took into account the "human factors" during the design process by addressing teamwork and its organization, and focused then on the technical means for modelling, simulation and analysis of the space segment of cis-lunar and interplanetary spaceflight missions. The major new elements are the COMET modelling software and the improved IRIS++ spacecraft simulation programme. With these extensions, the SSDW offers today a presumably unique environment for conceptual space mission design and analysis in terms of a harmonised integration of a proven interdisciplinary methodology and generic software infrastructure.

To demonstrate the concept, the developed approach was applied to a design example that addressed an acute topic of human space exploration in our times, namely the task of architecting a sustained and long-term programme to the Moon and beyond. The results show that a conventionally less explored scenario involving crewed space installations in cis-lunar space could play an interesting role within such programmes. The space station located at LL1 would allow permanent and global access to the surface of the Moon with presumable re-usable vehicles, combined with relatively low operational effort. Furthermore, this lunar space station scenario seems feasible and affordable with currently available technology mostly based on existing hardware.

The next development steps of the SSDW are proposed to extend the concept towards enabling sustaining conceptual design of planetary surface infrastructures on the Moon and Mars. This means, design of *space stations* would not only be aiming at the orbital segment of exploration systems, but at more elements of human spaceflight missions, namely surface stations and associated subsystems. As shown by this dissertation, mission aspects, especially the transfer-leg, are of primary importance when dealing with such scenarios involving far-Earth destinations. This area includes definition of LEO support infrastructure, design and sizing of transfer, ascent and descent vehicles and development of mission modes for in/outbound transfer missions for assembly and logistics. Like within this investigation, the subsequent steps will mainly be concerned with the system simulation software, which should reasonably be extended with mission planning and optimisation capabilities and a graphical user interface for convenient transfer mission design, including navigation, trajectory control and analysis modules.

With every step of improvement, conceptual design of human spaceflight missions and systems will remain for many years to come one of the most fascinating and demanding fields for research and education.

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