Institute of Space Systems
System Analysis Space Segment

Mission Architecture Study
CERMIT

\- *C*rewed *E*uropean exploRation *MI*ssion *T*rail -

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1. Introduction

Currently there are several plans and also designs for spacecraft enabling crewed space flights to Near Earth Asteroids (NEA) as one possible path to the far-future goal of reaching Mars. While Europe generally intends to join international space endeavors there are currently no dedicated plans for an own mission towards a NEA.

Based on a previous design for a spacecraft capable of a crewed asteroid mission [RD – X], DLR's 27th Concurrent Engineering Study investigates the complete mission architecture necessary to conduct such a mission or even a series of it. Labeled the *C*rewed *E*uropean Explo*R*ation *MI*ssion *T*rail (CERMIT) this study shall serve as first suggestion of how such a mission can be undertaken with a European perspective.

The CERMIT study took place from 6th to 10th February 2012 in the Concurrent Engineering Facility of DLR Bremen. All domains and disciplines have been staffed by DLR Bremen employees.

1.1. General Study Background

Near Earth Objects (NEO) are all celestial bodies that have a perihelion of smaller than 1.3 AU and an aphelion of smaller than 5.2 AU. Their compositions share certain similarities with Main-Belt asteroids and therefore it is likely that in fact they have their origin in this region of the solar system as well. Generally it is assumed that they are remains from the early solar system and could therefore provide information about primordial times of it.

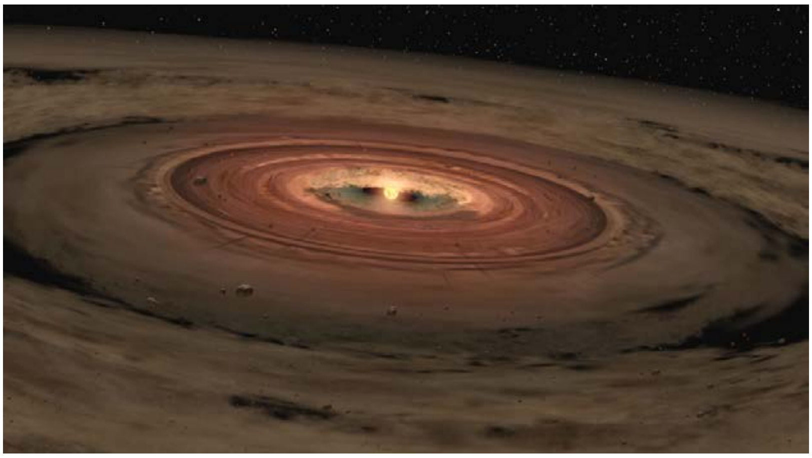

Figure 1-1: Protoplanetary disc of the early solar system [NASA].

Currently 8315 NEOs are known, of which 90 are comets (NEC) and the remaining 8,225 bodies are asteroids (NEA). Due to celestial mechanics, their lifetime is limited to some million years, once trapped in their current orbits, and their usual fate is either collision with the Sun, being expelled from the solar system or collision with a planet, including Earth.

NEAs have sizes between 25 m and 40 km and 1255 are a potential hazard to Earth, due to close approaches and possible collisions, of these 151 have a diameter larger than 1 km and could effectively end civilization and extinct large amounts of life on Earth.

1.1.1. Crewed Asteroid Missions

Asteroids, especially NEAs, are interesting for further investigation out of several reasons, including preparation of even more ambitious missions to e.g. Mars.
They offer information about earlier times of the solar system, can shed light on its formation and development. Generally they can forward planetary research efforts.
Collection of data about their inner structure, composition, strength but also the testing of technology necessary to alter an asteroid's trajectory are vital for scenarios of impact mitigation that are intended to defend Earth from asteroid impact events.
The asteroid environment, the fact that they are heliocentric objects (in difference to Moon, which is gravitationally bound to Earth), and easier access to them than to Mars make NEAs good candidates to test processes for and gain experience with long-duration missions beyond a low Earth orbit (LEO), i.e. without means to re-supply a mission or exchange crew members.
Another possible application of crewed asteroid missions is the testing of InSitu Resource Utilization (ISRU) technology in an actual space environment.
Despite the increased effort to realize a human crewed mission to an asteroid, an endeavour like this has certain advantages over missions solely based on automated probes, e.g.:

- Flexibility/ adaptability/ mobility of the human crew
- EVA allow direct interaction with surface and experiments
- Larger scientific exploit (e.g. more directed selection of samples, identification of worthy targets)
- More complex missions are possible

Currently only the United States have intentions for a crewed asteroid mission, based on their Space Launch System and the Multi-Purpose Crew Vehicle. However no target has been named yet and efforts are currently restricted to technology development. Europe's own plans only state the interest in participation of global space strategies and exploitation of ISS infrastructure. Neither Russia, China nor India have stated any intention to conduct such kind of missions.
[RD 1]

1.1.2. Study Objectives

To investigate the overall mission architecture of a crewed asteroid mission, the following objectives have been formulated and achieved during the CE-study:

- Establishing a viable scenario for a crewed European NEO mission with special regard to:
    - Launch of all spacecraft and mission components into orbit
    - Transfer of the spacecraft on a rendezvous trajectory with the asteroid
    - Re-entry of the returning spacecraft into the Earth atmosphere
    - General mission strategy
- Preliminary design of a launcher (size, mass, staging, tank design, feed system) with, if possible, use of components or technology available in Europe
- Preliminary design of transfer-stage system for accomplishing the maneuvers beyond low Earth orbit
- Capsule optimization
- Creation of a mass budget
- CAD configuration for all mission relevant components

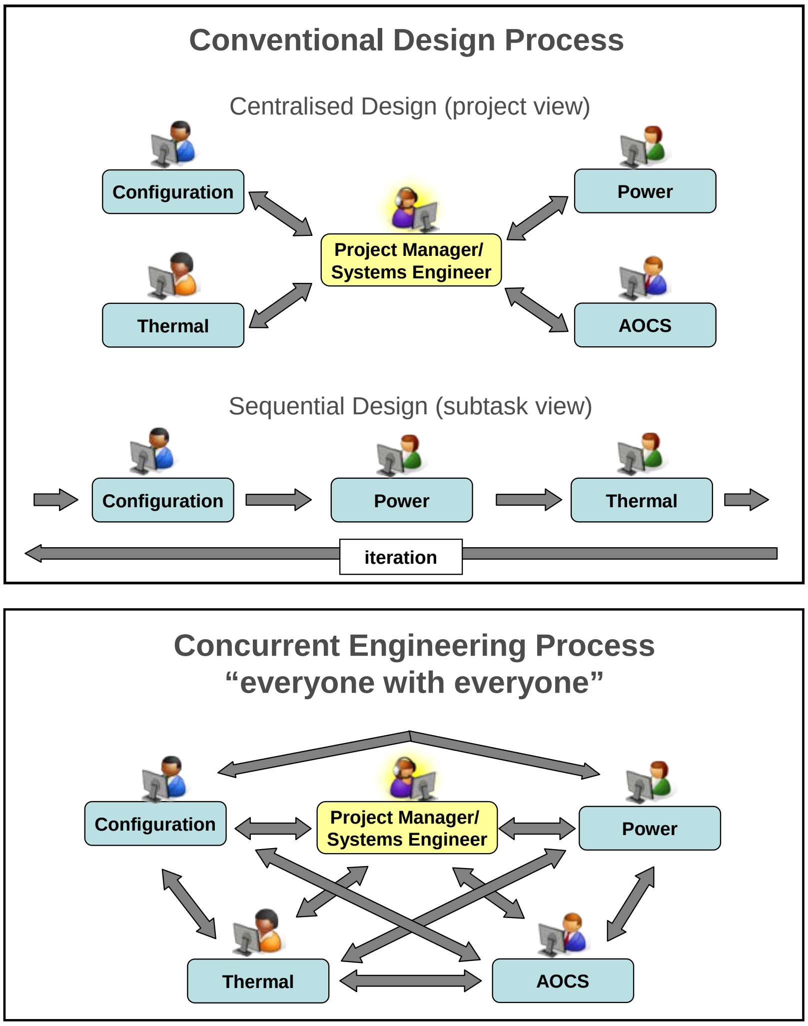

Figure 1-2: The Concurrent Design approach compared to projections of conventional design process.

1.2. Concurrent Engineering Approach

To investigate and define the architecture and technical requirements for a mission along the lines of CERMIT a Concurrent Engineering (CE) Study at DLR Bremen has been conducted. The CE-study comprised the analysis of all necessary mission components, i.e. crew module, habitat, transfer stages and launcher, on a system level.

The applied Concurrent Engineering (CE) process is based on the optimization of the conventional established design process characterized by centralized and sequential engineering (see Figure 1-2 top). Simultaneous presence of all relevant discipline's specialist within one location and the utilization of a common data handling tool enable efficient communication among the set of integrated subsystems (see Figure 1-2 bottom).

The CE-Process is based on simultaneous design and has four phases ("IPSP-Approach"):

1. Initiation Phase (starts weeks/months before using the CE-facility):

- Customer (internal group, scientists, industry) contacts CE-team
- CE-team-customer negotiations: expected results definition, needed disciplines

2. Preparation Phase (starts weeks before using CE-facility):

- Definition of mission objectives (with customer)
- Definition of mission and system requirements (with customer)
- Identification and selection of options (max. 3)
- Initial mission analysis (if applicable, e. g. based on STK)
- Final definition and invitation of expert ensemble, agenda definition

3. Study Phase (1- 3 weeks at CE-Facility in site):

- K/O with presentations of study key elements (goals, requirements)
- Starting with first configuration approach and estimation of budgets (mass, power, volume, modes, …) on subsystem level
- Iterations on subsystem and equipment level in several sessions (2- 4 hours each); trading of several options
- In between offline work: subsystem design in splinter groups
- Final Presentation of all disciplines / subsystems

4. Post Processing Phase:

- Collecting of Results (each S/S provides Input to book captain)

- Evaluation and documentation of results
- Transfer open issues to further project work

The DLR's Concurrent Engineering Facility in Bremen is derived from the Concurrent Design Facility at ESA's ESTEC (European Space Research and Technology Centre), which has already been in operation since 1999. Bremen's DLR-CEF has one main working room where the whole design team can assemble and each discipline is supplied with an own working station for calculations and interaction with a special design tool developed by ESTEC. Three screens, one of them interactive, allow displaying data in front of the complete team. Further working positions are provided in the centre of the working area and are usually reserved for customers, PIs, guests and also the team leader and possibly the systems engineer. Two more splinter rooms provide the design team with separated working spaces where sub-groups can meet, discuss and interact in a more concentrated way.

Figure 1-3: Concurrent Engineering Facility main room (left) and working during CE-study phase (right) at DLR Bremen.

The major advantages of the CE-process are:

- Very high efficiency regarding cost & results of a design activity (Phase 0, A)
- Assembly of the whole design team in one room facilitates direct communication and short data transfer times
- The team members can easily track the design progress, which also increases the project identification
- Ideas and issues can be discussed in groups, which brings in new viewpoints and possible solutions; avoidance and identification of failures and mistakes

1.2.1. Mission Architecture Definition (MAD) Studies

Besides designing actual spacecraft, the concurrent working environment as given by the CEF can also be used to formulate and elaborate whole mission architectures involving more than a single spacecraft.

During these studies no Integrated Design Model or similar software is usually applied as data is not accumulated on an equipment level. The work concentrates more on a system level of various components, i.e. rough mass estimates, and mostly on identifying possible problems or design issues of the mission and also working solutions. Outcome of such a study is the formulation of:

- The overall mission strategy
- Development suggestions
- Pointing out design trades with disadvantages and advantages of all options
- Risks and technology needs
- Requirements for all mission components

Parts of the whole mission design can then be further investigated in ordinary CE studies.

In difference to the spacecraft design iterations in MAD studies, the iterations have to be applied on the overall strategy as well to find a suitable and likely mission scenario. This will in turn affect the mission components (e.g. spacecraft and launcher), whereas changes on them again affect the mission scenario. It is therefore necessary to repeat the iterations to consolidate the mission scenario.

1.3. Document Information

This document summarizes the progress and results of the DLR Concurrent Engineering study about the CERMIT mission, which took place from 6th to 10th February 2012 in the Concurrent Engineering Facility of the DLR Institute of Space Systems in Bremen.
The single domains as investigated during the study are covered in individual chapters, which explain the study progress, elaborate on decisions and trade-offs made during the study and also design optimizations.

2. Mission Background and Overview

2.1. Mission Objectives

The overall objectives for CERMIT are very general and mostly aim at simply fulfilling a crewed mission beyond Earth orbit. Actual scientific goals need to be formulated at a later time, once the feasibility of this kind of mission is established and scientists are more directly involved in the mission planning. The current mission objectives are listed in Table 2-1.

Table 2-1: Mission Objectives for CERMIT.

Objective No.	Description
MI-OJ-0010	Injection of all necessary components for a NEA mission into LEO
MI-OJ-0020	Transfer to and exploration of a suitable NEA target
MI-OJ-0030	Safe return of the crew to Earth
MI-OJ-0040	Prominent role for European participants

2.2. Mission Requirements

In preparation for the CE-study the following mission requirements (Table 2-2) have been defined to allow achieving the current mission objectives:

Table 2-2: Mission Requirements for CERMIT.

Objective No.	Description
MI-DE-0010	The mission duration shall not exceed 180 days
MI-DE-0020	The initial parking LEO shall be circular and have a minimum altitude of 300 km
MI-LA-0010	An existing human-rated launcher shall be used for crew transport into orbit
MI-LA-0020	For support of the European role, the launches shall be conducted from Kourou
MI-LA-0030	The mission launch date shall be in the frame of 2020 to 2040 and have launch window of minimum 1 month

2.3. System Requirements

The mission components, i.e. all involved systems have the following requirements for fulfillment of the mission plan (Table 2-3):

Table 2-3: System Requirements for CERMIT.

Objective No.	Description
ST-PE-0010	Launcher and transfer stages shall be able to support a crew module of of a maximum mass of 40.000 kg as payload to the NEA

ST-PE-0020	The transfer vehicle shall cover all maneuvers (transfer trajectory injection, asteroid arrival, asteroid departure) with a maximum Delta-V of 7.000 m/s
ST-PE-0030	The crew module shall be able to handle 4 crew members for 180 days + launch window
ST-DE-0010	The crew capsule shall be able to conduct a direct re-entry at Earth
ST-DE-0020	Modular and enhanceable technology usage for adapted application in future missions
ST-DE-0030	Where possible only technologies available in Europe shall be used

2.4. Baseline Design

To accomplish the planned mission, three distinct systems are necessary:

1) The transfer stages for maneuvers to and at the asteroid
2) The launch vehicle that transports all mission components into LEO
3) The crew module consisting of capsule and habitat

Depending on the mission-ΔV the actual number of transfer stages is adapted; the same is true for the number of launches needed. The current baseline launcher layout foresees a payload of 200,000 kg into a 300 km orbit, variations with smaller payload mass exist, but would increase the number of launches needed to carry the necessary payload into LEO.
Currently the spacecraft consisting of the crew capsule, habitat module and transfer stages is termed *E*uropean e*X*tensive *P*ersonnel *L*aboratory f*O*r *RE*mote *R*esearch (EXPLORER) and the launcher is be preliminarily labelled *SIRIUS* after the brightest known star.

2.4.1. Mass budget

The overall mass budget for CERMIT is listed in Table 2-4. For the calculation of the masses and especially the propellant, the subsequent stages have been regarded as payload for previous ones. This means that the mission payload modules (crew capsule, habitat module and all equipment and crew) are the payload for the Asteroid Departure (AD) stage. Together they form the payload for the Asteroid Arrival (AA) stage, this complex then is the payload for the Earth Departure (ED) stages.
With the current configuration a total mass of approx. 360,000 kg has to be transported into LEO to conduct the missions – as comparison, the Saturn V of the Apollo programme had a payload mass of 120,000 kg in the initial version (later versions could transport up to 133,000 kg into LEO) [RD 2].

2.5. To be Studied and Additional Considerations

During the study the transfer stages and the launcher has been investigated while some optimizations for the mission payload (crew module and habitat) have been conducted. There are however certain aspects that still need further investigation, e.g.:

- Ground Segment Infrastructure (including costs)
- Stage optimization for the launcher and the transfer stages
- Planning of a development schedule

Table 2-4: Mass budget for CERMIT, the Asteroid Departure (AD) stage is the payload for the Asteroid Arrival (AA) stage, which again is the payload for the Earth Departure (ED) stage #2, and so on.

AD - Transfer Stage	Iteration #5	10.02.2012	
	W/O Margin	Total	% of Total
Dry mass contributions		kg	
Structure, Engines, Tanks	3343,00	3343,0	8,20
P/L Modules		37404,8	91,80
Total Dry Mass	3343,00	40747,8	kg
Propellant	33015,00	33015,0	81,02
Total wet mass		73762,8	kg

AA - Transfer Stage	Iteration #5	10.02.2012	
	W/O Margin	Total	% of Total
Dry mass contributions		kg	
Structure, Engines, Tanks	5845,00	5845,0	14,34
AD - Transfer Stage (incl. P/L)		73762,8	181,02
Total Dry Mass	5845,00	79607,8	kg
Propellant	78412,00	78412,0	192,43
Total wet mass		158019,8	kg

ED#2 - Transfer Stage	Iteration #5	10.02.2012	
	W/O Margin	Total	% of Total
Dry mass contributions		kg	
Structure, Engines, Tanks	10051,00	10051,0	24,67
AD & AA - Stages (incl. P/L)		158019,8	387,80
Total Dry Mass	10051,00	168070,8	kg
Propellant	89861,00	89861,0	220,53
Total wet mass		257931,8	kg

ED#1 Transfer Stage	Iteration #5	10.02.2012	
	W/O Margin	Total	% of Total
Dry mass contributions		kg	
Structure, Engines, Tanks	10051,00	10051,0	24,67
AD & AA & ED#2 (incl. P/L)		257931,8	633,00
Total Dry Mass	10051,00	267982,8	kg
Propellant	89861,00	89861,0	220,53
Total wet mass		357843,84	kg

Payloads Modules	Iteration #5	10.02.2012	
		40000	kg
		2595	kg
	Without Mar	Total	% of Total
Dry mass contributions		kg	
10_Crew Module	7255,00	6455,6	15,84
11_Habitat_Module	21615,00	21024,4	51,60
Total "Dry" Mass	28870,00	27480,0	kg
System (safety) margin		[kg]	
Crew Module Mass, incl. Sys.Marg.+Consum.+(		9514,6	23,35
Habitat Module Mass, incl. System Margin + Co		27890,3	68,45
Total Dry Mass WITH margin		37404,8	kg
Total Launch mass (Dry Mass + SPO, ir		37404,8	kg

Payloads Modules	37404,8
AD - Transfer Stage	36358,0
AA - Transfer Stage	84257,0
ED#2 - Transfer Stage	99912,0
ED#1 Transfer Stage	99912,0
Total Launch mass	357843,8

3. Mission Analysis

3.1. Requirements and Design Drivers

The mission analysis part of the crewed asteroid mission study is mostly a target selection process, based on known round-trip transfer trajectories and ΔV's. Although the transfer trajectory calculations have not been part of the study, a short introduction shall provide the basic assumptions and processes behind the finally used ΔV values.
The basic mission requirements, also influencing other parts of the systems, are mentioned in Table 2-2 of this report.

3.1.1. Scenario Overview and Assumptions

The primary task of reaching to an NEA and returning to Earth within a given timeframe has been addressed by solving the typical rendezvous problem in combination with some common simplifications for the overall transfer.

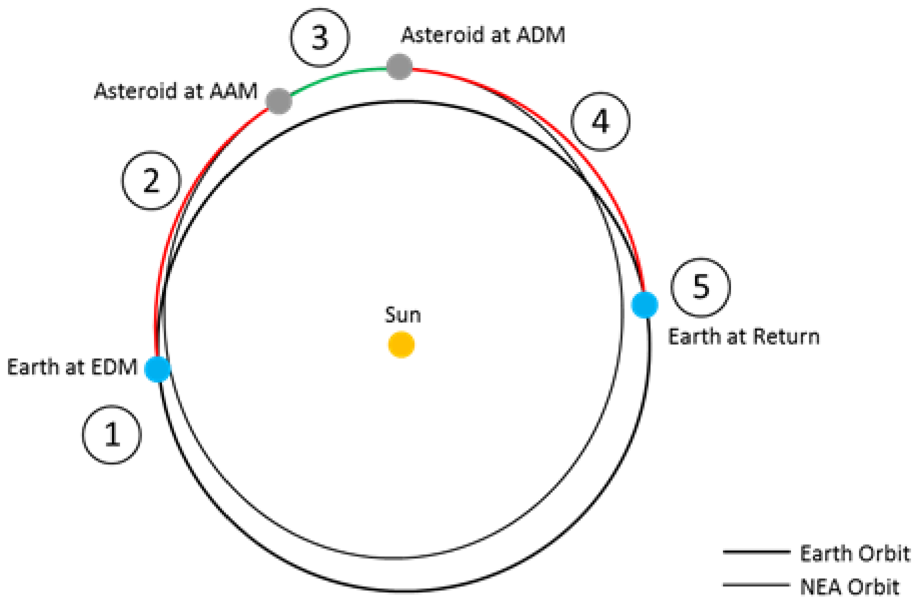

Figure 3-1: Mission profile for a crewed NEA mission with five mission phases.

The first step was the discretization of the entire transfer by applying the patched conics method. The spacecraft experiences a unique gravitational attraction, either by the Sun or by Earth, within each conic section. The Moon and other third-body forces were neglected. The borderline between the conics was defined by the sphere of influence. The attracting focus changes from Earth centred to sun centred, or vice versa, at a distance of 950.000 km from Earth. The gravitational force of NEAs in general is very small, wherefore the phase of close proximity was defined to be a heliocentric trajectory instead of an NEA-centred one. This allowed the utilization of two-body field dynamics and analytical problem solving instead of multi-body calculations with numerical methods for the entire mission. The final mission profile was divided in five phases as it can be seen in Figure 3-1.

Phase 1 is called the stacking phase, where the entire transfer vehicle is build up in LEO. The end of phase 1 happens with the execution of the Earth-departure manoeuvre (EDM), which marks the beginning of the mission duration count. Phase 2 is called the outbound flight, where the spacecraft approaches the asteroid. The close proximity operations, phase 3, start with the execution of the asteroid arrival manoeuvre. During this phase the spacecraft co-orbits the asteroid, which means that the heliocentric orbits of the NEO and the spacecraft are assumed to be equal. The asteroid departure manoeuvre marks the end of the exploration phase and initiates phase 4, the inbound flight. This finally ends with the return to Earth and a hyperbolic re-entry without further manoeuvre.

The three ΔV main manoeuvres for this mission profile, EDM, AAM, ADM and the re-entry velocity are unknowns and need to be determined. Additionally it was required to find the minimum ΔV for the entire transfer, which could only be achieved by varying the in- and outbound times of flight and solving the problem for the optimum pair of conics.

As the transfer problem itself can be described in an analytical way, it required an algorithm to solve for optimum ΔV and especially to provide solutions and launch dates for more than 2500 potential NEA targets. This algorithm in the end provided minimum ΔV for several launch dates around the optimum launch date for potential NEA's that are investigated as mission targets by the target selection process. Further information regarding the algorithm can be found in [RD 1].

3.1.2. Launch Window Requirements

Especially for a human space mission it is essential to provide sufficient launch windows (LW) to handle delays during the launch campaign. An LW of at least one month seems to be reasonable for this kind of mission. Previous studies often neglect the fact that in the past numerous launch delays have occurred when dealing with crewed spacecraft.

An LW is in general described as the timeframe, when a launcher can lift up from Earth's surface to reach its final orbital position. The restrictions are mostly given by the available launcher performance. The herein used definition of an LW is a bit different:

NEA missions will require multiple launches to LEO during the stacking phase. Although each single launch has its own LW to reach the departure orbit, the critical LW is described by the timeframe, wherein the entire spacecraft stack can depart from LEO and still rendezvous with the asteroid. This timeframe is restricted by the transfer vehicle performance and not by the single launcher performance. Of course the single launcher performance enables the transfer vehicle performance as it lifts up the necessary stages and propellant mass.

In summary the departure LW provides a margin for the stacking of the transfer vehicle in LEO. The target NEA will be missed by missing the launch opportunity, which points out the importance of such a requirement. It also justifies the requirement for the selection of backup targets for a given spacecraft architecture. Missions to Moon or Mars on the other hand show recurring launch opportunities within narrower ΔV constraints.

3.1.3. Target Selection Requirements

The importance of the target selection has already been pointed out in the previous chapter. For target selection the following criteria have been applied:

- the minimum total round-trip ΔV shall be less than 7 km/s
- the departure date shall occur between 2020 and 2040
- the diameter of the NEA shall be bigger than 25 m

This minimum set of requirements resulted in a list of 10 targets, shown in Table 3-1.

Table 3-1: NEA target list after the application of the selection criteria.

Name	Launch Year	D [m]	OCC	ΔV_{min} [km/s]	PHA	TOF_{out} [d]	Stay [d]	TOF_{in} [d]
1999 AO_{10}	2025	50-113	6	6.64	No	30-138	10	32-140
1999 CG_9	2033	27-60	6	6.46	No	44-125	10	45-126
2000 SG_{344}	2028/29	33-73	2	4.65	No	43-143	10	27-127
2001 FR_{85}	2039	37-83	3	5.31	No	82-117	10	53-88
2003 LN_6	2025	38-85	5	6.93	No	13-141	10	29-157
2003 SM_{84}	2040	85-189	1	6.80	No	39-135	10	35-131
2007 UW_1	2039	86-192	7	6.99	No	23-134	10	36-147
2009 OS_5	2020	58-129	5	6.80	No	32-135	10	35-138
2009 UY_{19}	2038/39	62-138	1	6.07	No	82-120	10	50-88
2011 DV	2039	219-490	3	5.96	No	75-126	10	44-95

3.2. Options and Trades

3.2.1. Launch Window Definition

As already mentioned, the transfer calculation algorithm has provided all manoeuvre ΔVs and re-entry velocities for the ten preselected targets. The datasets have a resolution of one day. The LW shall last for at least one month (30 days). A plot of the typical ΔV distribution for the exemplary target 1999 AO10 is shown in Figure 3-2.

The task for the final selection has been to find the optimum position of the LW within the available range of launch dates. For scenarios with multiple transfer stages and manoeuver-sharing between several stages, the total ΔV value cannot be directly used as a criterion for the LW. The structural mass of the vehicle changes throughout the mission if staging is used.

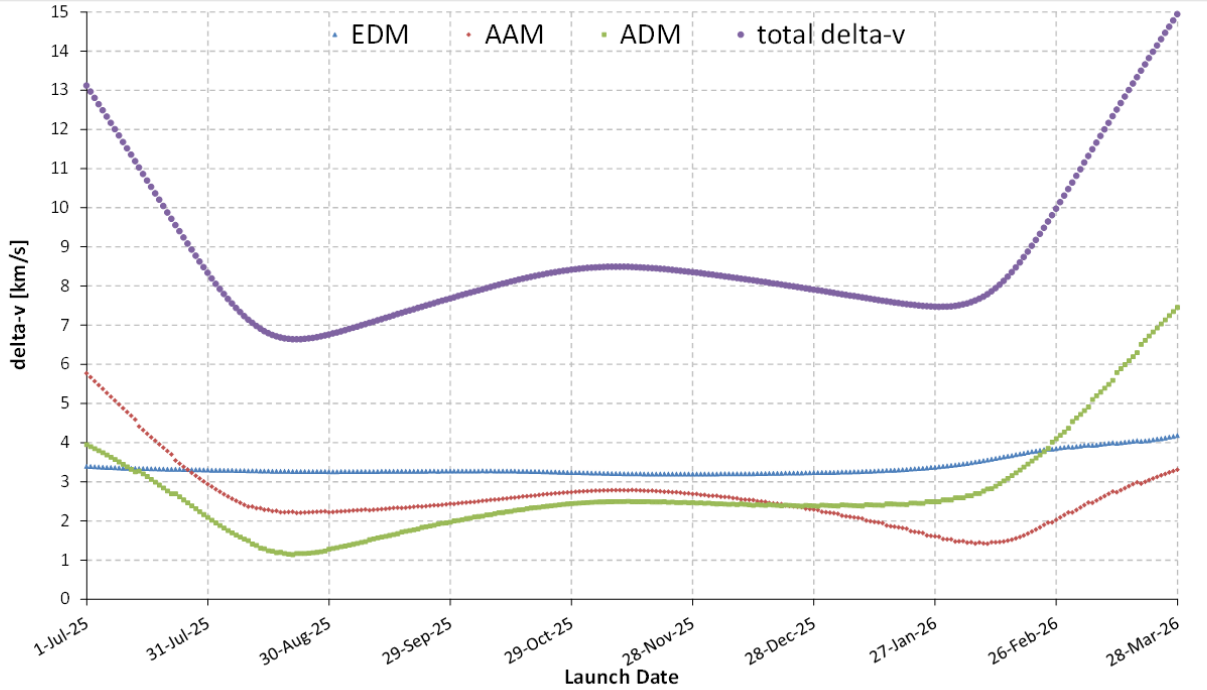

Figure 3-2: Δv distribution of NEA 1999 AO10 with all mission manoeuvres.

Propellant mass that has to be accelerated for later use is more ‘expensive’ for the early stages. This led to the first attempt of LW definition that resulted in the definition of boundary values for EDM, AAM and AAM. This approach was found to be inconsistent with the ΔV distribution and led to non-continuous LW’s.

The second attempt focused more on the continuous LW constraint and therefore 30-day blocks have been analysed to find the best fitting block as LW for the mission. The block has been initially characterized by its mean value, which also showed to be misleading.
The final solution based on the previous one but now the blocks have been built by:

- selecting the launch date with the minimum total ΔV
- building of the LW block by adding 15 days in each direction
- fine-tuning of the LW position by analysing the gradients of the total ΔV-curve at the beginning/ending position of the LW and shifting the “centre” of the LW in a way as to minimize the maximum occurring ΔV (which per the above definition always is at the rim of the LW)

This final approach led to the designated LW’s and has been repeated for each one of the ten initial targets. A review of the resulting ΔV-values led to the decision to select six NEA’s as potential targets for the to-be-defined transfer vehicle architecture. The final results are given in Table 3-2.

3.2.2. Initial Orbit

As the CE study scope asked for a mission strategy, involving launchers and launch windows, the initial low-Earth parking orbits play a significant role. The altitude of these orbits have been assumed as constant but the inclination requirement changed depending on the target NEA and the launch date within a launch window. By orbital mechanics reasons it has been necessary to choose the target inclination by identifying the maximum inclination for a given launch window. This describes the worst case from a launchers perspective and allows to achieve also lower declinations to fulfil the requirement of a coplanar departure manoeuvre.

During the course of the study the parking orbit altitude has been increased from 200 to 300 km to accommodate for a better launch trajectory. The change in altitude does not change the inclination requirement and only slightly changes the EDM value that decreased in the order of a tenth of one m/s.

3.3. Summary

The main outputs of the mission analysis have been the final NEA targets that have been selected by several mission and systems considerations. Totalling a set of six targets in a timeframe of 20 years shall provide a solid base for a crewed exploration strategy. The worst case inclination is 69°, which serves as the design case for the launcher performance estimations. The transfer durations vary between 2 and 4.5 months for the outbound flight, which gives the astronauts the opportunity to prepare the spacecraft and themselves for at least two months for the proximity operations. The inbound flight accordingly varies between one and 3.5 months which also gives the crew at least one month of preparation time for the re-entry. The duration of the proximity operations stay constant with a value of ten days.

Table 3-2: NEA target list after the application of the selection criteria.

Mission Phase	Min	Max
Outbound Flight	65	137
Proximity Operations	10	
Inbound Flight	33	105

Table 3-3: Final targets and their parameters.

Name	Launch Year	D [m]	OCC	ΔV_{min} [km/s]	PHA	TOF_{out} [d]	Stay [d]	TOF_{in} [d]
1999 AO_{10}	2025	50-113	6	6.64	No	30-138	10	32-140
1999 CG_{9}	2033	27-60	6	6.46	No	44-125	10	45-126
2000 SG_{344}	2028/29	33-73	2	4.65	No	43-143	10	27-127
2001 FR_{85}	2039	37-83	3	5.31	No	82-117	10	53-88
2003 LN_{6}	2025	38-85	5	6.93	No	13-141	10	29-157
2003 SM_{84}	2040	85-189	1	6.80	No	39-135	10	35-131
2009 OS_{5}	2020	58-129	5	6.80	No	32-135	10	35-138

3.4. To be Studied/ Additional Considerations

The search for NEA's is not yet completed and will generate more potential targets and refine the orbits of already detected targets in the future. Therefore it is necessary to regularly recalculate the ΔV values for already existing and new NEA's. This can lead to additional target candidates for the mission architecture of this study.

Abort options as a safety feature of NEA-missions have already been calculated in [RD 1] but have not yet been considered in this study. It would be necessary to study these options and provide some initial thoughts about astronaut safety.

As the herein used calculations focused on preliminary mission design, the approach lacks precise orbit propagation. The entire transfer shall be reinvestigated with numerical methods to include multi-body gravitational influences and to uncover potential differences to the simplified approach.

Especially for the re-entry it would be very helpful to provide exact re-entry trajectories. They would allow to investigate the potential touch-down areas of a returning crew capsule. This would affect the decisions regarding the descend and landing subsystems of the capsule.

Last but not least it has been found, that changes in the total mission duration would affect the total ΔV and therefore the system's mass budget. As this also has an impact to all the subsystems and the entire strategy for a NEA mission, this has not been investigated during this study. But preliminary investigations have shown a big potential for certain NEAs, when the mission duration can be extended from the 180 days used in CERMIT [RD 1].

4. Crew Module

The crew module domain during the CERMIT study encompassed two main parts, namely the habitat module for the astronauts, which has been based on the Columbus module of ISS and the crew capsule along with its service module for the latter - the exact nomenclature is sketched in Figure 4-1.

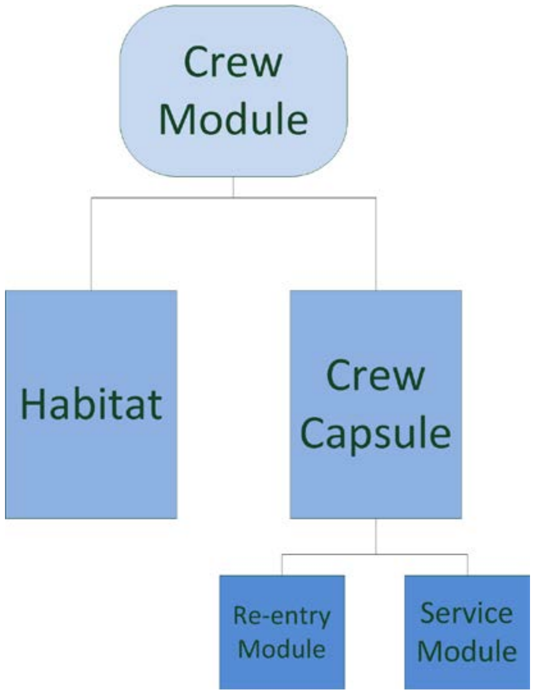

Figure 4-1: Nomenclature used for the Crew Module domain.

The purpose of the crew module is to bring the astronauts to the asteroids and take them back to the Earth. Initially the aim has been to find the possibilities of reusing the European Columbus module as habitat part of the crew module. The number of the astronauts and the duration of the mission affect the habitable volume required. The volume drives the size of the vehicle and the propulsion systems.

4.1. Habitat

4.1.1. Assumptions

To design the habitat it is assumed that 17.7 m^3 of habitable volume (HV) is required for each of the crew astronauts [RD 1]. So for four crew members it is in total 70.90 m^3 which is 67.5% of the pressurized volume (PV), i.e. PV= HV+LSSV = 105 m^3, where LSSV means Life Support System Volume but also includes other equipment for the flight.

To calculate the volumes of the required subsystems and the system as a whole, the following margins have been and assumptions have been made:

- Subsystem (e.g. life support system) margin 30%
- System margin 20%
- Using (only) European technologies
- Possible scaling of Columbus module for mass optimization

4.1.2. Requirements and Design Drivers

The following system requirements have been selected for the habitat design:

- Operation duration: 210 days (180 days mission time, 30 days launch window)
- Crew: 4 astronauts

Design Drivers:

- Diameter remains constant as original Columbus module
- New volume for the life support equipment while the height remains identical to the original Columbus module

Table 4-1: Comparison of Columbus with different pressurized modules.

Parameters	Columbus	Zvezda	Kibo	Destiny	Tiangong2
Mass (kg)	10275	20 295	14800	14,515	20,000
Dimensions D x L (mxm)	4.49mx6.87m	4.35mx13.1m	4.2mx11.2m	4.3 m x 8.5m	4.2x14.4m
Volume (m^3)	108.78	194.69	155.17	123.44	218.96
Pressurized Volume(m^3)	75	89	-	106	-
Mass/Vol. (kg/m^3)	**94.5**	**104.24**	**95.5**	**117.59**	**91.32**
Crew Size	3	6	2, max 4 with limitations	3	3
Launch vehicle compatibility	Space Shuttle	Proton-K (№398-01)	Space Shuttle	Space Shuttle	Chang Zheng
Life time	10	15	10	-	10 years
Docking ports	1	4	-	2	-
Launch	February 2008	July 2000	July 2009	Feb. 2001	2013 (planned)

4.1.3. Options and Trades

The Columbus module was originally planned as part of the ‘Columbus Program’ of ESA to develop an autonomous space station that could be used for a variety of microgravity experiments. It eventually evolved into the European part of ISS since 2008. To find the possibilities of using Columbus as the Crew Module, it has been compared with other pressurized modules from different countries. Table 4-1 contains that data.

After the comparison, the following reasons have been established to reuse Columbus as the Crew module for the mission:

- Its Laboratory was constructed by Alcatel Alenia Space in Italy
- The functional architecture and software were designed by Astrium in Germany

Advantages of using Columbus module:

- Development costs and time are be reduced
- Integration process is easier
- Validated technology (reliable)
- Small adaptation

Another vital option could be using European ATV (Automated Transfer Vehicle) service module which is man rated. It has a pressurized volume of 48 m³, weighs 20700 kg and can carry a payload of 8000 kg to ISS. DLR and EADS Astrium have announced a project to adapt the ATV into a crew transportation system (3 man crew).
The mass/volume ratio of the current ATV service module (122 kg/m³) is larger than that of Columbus module (94.5 kg/m³, s. Table 4-1). Its total pressurized volume (48 m³) is significantly smaller than the total pressurized volume of Columbus module (75 m³). Hence, at this moment Columbus Module is selected for further analysis as the crew habitat module.

4.1.4. Baseline Design

The final design is done on the basis of Core Columbus module. Figure 4-2 shows the dimensions of the original Columbus Module.

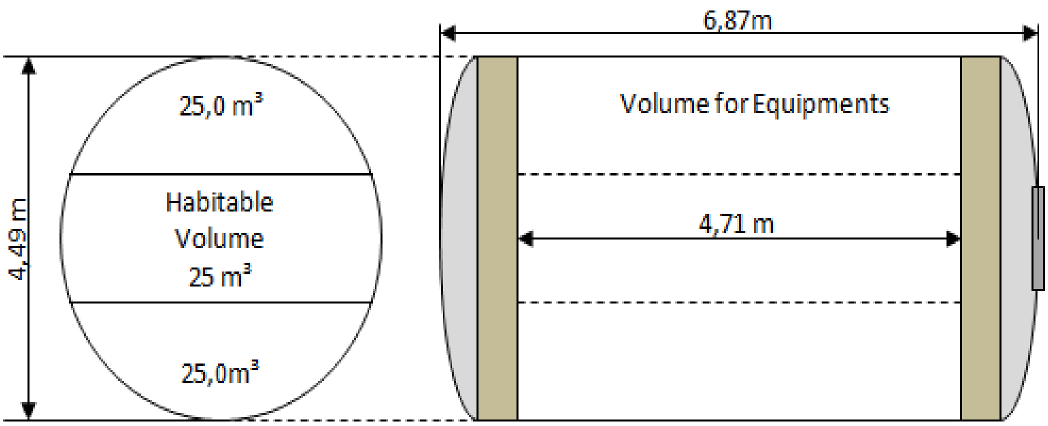

Figure 4-2: Dimensions of Columbus Module.

In the beginning of the study the ‘Crew Module’ group has been provided by a scaled version of the Columbus module for four crew members as a starting point [RD 1].

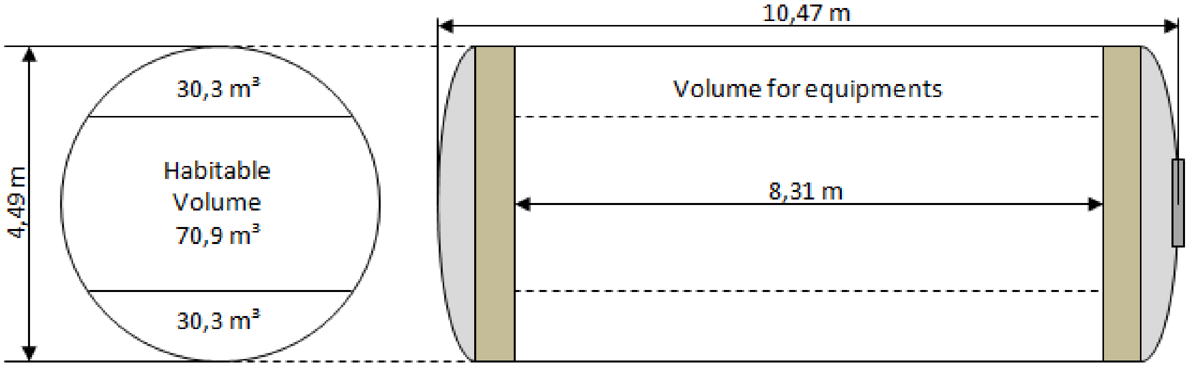

Figure 4-3: Dimensions of Scaled Columbus Module (4 crew).

During the study this Scaled Columbus Module (for 4 astronauts) has been analyzed further to optimize the mass. It has been redesigned and re-scaled by calculating the total subsystem (e.g. life support system) volume required. This model is called the ‘New Columbus module’ and is depicted in Figure 4-4.
The scaling is done by keeping the Habitable volume constant (70.9 m³) and changing the Equipment volume. As a result it is found that the total length can be decreased, which effectively reduces the total mass by around 4,000 kg.

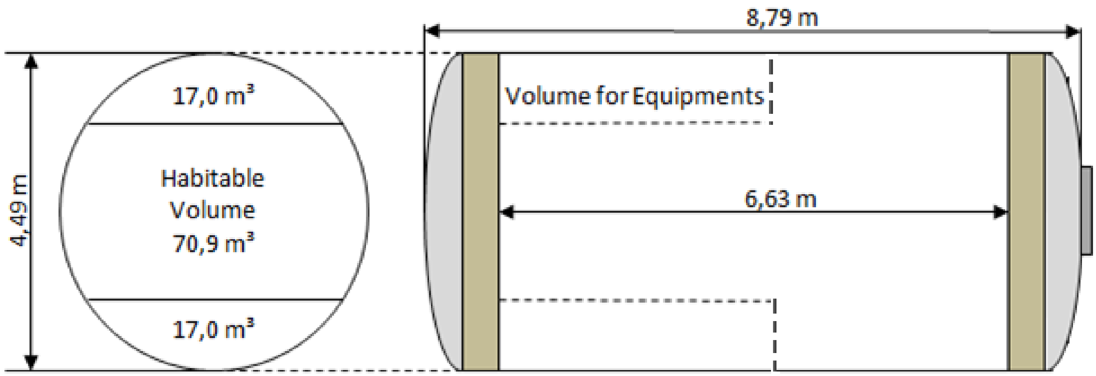

Figure 4-4: Dimensions of the ‘New Columbus Module’.

At this point it is found that the crew needs to stay in the crew module another extra 30 days to meet the launch window. So, the new mission duration is (180+30) days = 210 days. The extra payload for these 30 days is added to the total mass budget. Table 4-2 summarizes the complete module design.
The mass/volume ratio of the original Columbus module is 94.5 kg/m³. For the ‘New Columbus module’ it is 114 kg/m. But the total length is increased for the ‘New Columbus module’. So, theoretically the ratio could be in worst case equal to 94.5 or lower (assumed

new materials and technologies will be available in future). Hence, it can be concluded that the new scaling is done in a conservative manner and theoretically it is possible to reduce the mass by 16.4% or ~2,600 kg.

Table 4-2: Summary of the 'New Columbus Module'.

Parameters	New Columbus Module
Mass (kg)	15,827
Dimensions D x L (mxm)	4.49 x 8.79
Volume (m	140
Pressurized Volume(m³)	75
Mass/Vol (kg/m³)	**114***
Crew size	4
Life time	-
Launch	-

4.1.5. To be studied/ Additional Considerations

After the study on Crew module, it is suggested that detailed analysis should be done on life support systems, structural stability of the new design and any possible further mass optimization. Research can also be done to find the possibilities of using more advanced technologies also from other nations to enhance the capabilities.

4.2. Re-entry Capsule and Service Module

4.2.1. Assumptions

The basic assumption for the Crew Capsule has been that the system should be able to operate for 20 days autonomously e.g. during the initial phase of the mission for docking with the main spacecraft or simply operations during crew launch. To accommodate the crew comfortably enough the habitable volume has been assumed as 4 m³ per astronaut according to NASA standards [RD 1].

Initially the design has been assumed to shelter four astronauts, with the option to reduce the crew in case the design becomes too massive to be manageable by the launcher.

4.2.2. Requirements and Design Drivers

The capsule has to be capable of ascend, re-entry, docking with the spacecraft and allowing EVAs during the mission. The capsule has to provide consumables and system capabilities for a period of 20 days.

The largest possible re-entry velocity based on the given mission analysis has been calculated to 11.58 km/s, which therefore has to be sustained for re-entry by the Thermal Protection System (TPS) of the crew capsule.

4.2.3. Options and Trades

The crew capsule has been assumed to be firmly set during the study and has not been changed. Three options have been derived depending on the crew size beforehand [RD 1]. They are summarized in Table 4-3.

Table 4-3: Mass overview of the Crew Capsule and Service Module depending on crew size [RD 1].

Crew Size	Crew Capulse Mass	Consumables Mass	Service Module Mass	Total
2	4,058 kg	354 kg	1,123 kg	5,535 kg
3	5,052 kg	528 kg	1,616 kg	7,196 kg
4*	5,976 kg	703 kg	2,070 kg	9,097 kg

* has been baseline during the design

Another major design option has been the material selection for the Thermal Protection System (TPS) as applied to shield the spacecraft during re-entry. Three systems have been compared, the TPS of the Advanced Reentry Demonstrator (ARD) of DLR, the material used for Apollo and a European Ceramic TPS material. Whereas the former two both have flight heritage, the latter does not have that, but has the smallest mass and is therefore chosen for the design. The various TPS alternatives are summed up in Table 4-4 and the TPS is depicted in Figure 4-4. The data is derived from [RD 3]

Table 4-4: Mass comparison of the various TPS alternatives.

	Main Shield* [kg]	Area 1* [kg]	Area 2* [kg]	Area 3* [kg]	Total [kg]
ARD	2,146	214.3	137.7	78	2,576
Apollo	1,296	248.3	160	90.3	1,795
Ceramic	1,156	217	174.3	131.5	1,679

* areas from bottom to top in Figure 4-4

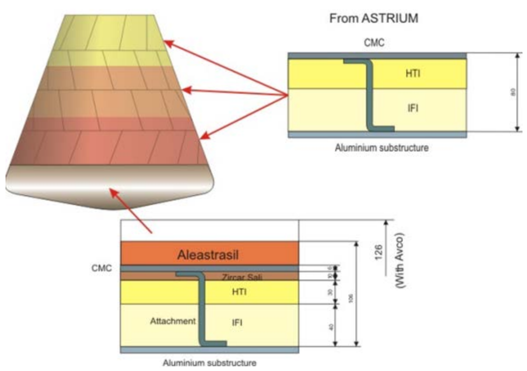

Figure 4-5: Sketch of the TPS.

The calculations have been made without considering relative motion of the atmosphere, no actual trajectory optimization and no re-entry angle calculations from mission analysis. In general the TPS mass increased by 620 kg in comparison to the pre-study value, resulting in a total mass increase of 720 kg and therefore increasing the total mass to 9,515 kg.

It has been initially considered to re-use the ATV Service Module for CERMIT, but this has been ruled out due to the fact that the for CERMIT the Service Module does not have the task of conducting orbital manoeuvers in difference to ATV, therefore the complex propulsion system can be saved.

4.2.4. Baseline Design

The crew capsule has been kept in the 4 crew version throughout the study and its rough layout can be seen in Figure 4-5. The internal layout of the re-entry capsule can be seen in Figure 4-6.

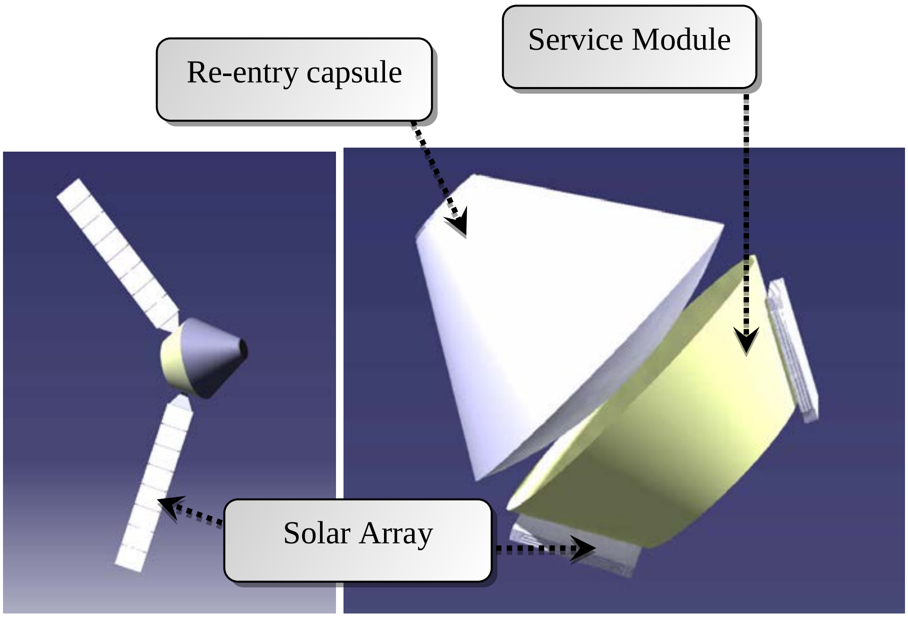

Figure 4-6: The rough crew capsule design with deployed (left) and undeployed (right) solar arrays.

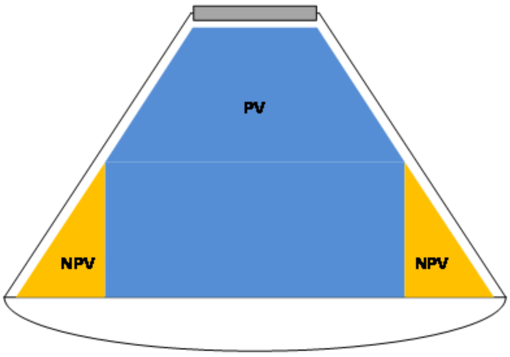

Figure 4-7: The rough internal layout of the re-entry capsule with the pressurized volume (PV) and non-pressurized volume (NPV).

The largest diameter of the re-entry capsule is 5.1 m, it has a height of 3.6 m and the cone angle – based on NASA's Multi-Purpose-Crew-Vehicle – is 32.5°. At its tip it is equipped with a docking port in compliance with the International Docking System Standard (IDSS). In general it has a total pressurized volume of 16 m³.

The service module is a 1.9 m tall and has a bottom diameter of 3.1 m. It carries the support systems, including power supply, for the re-entry capsule. The power is supplied by a solar power generator using two arrays of 8 segments each.

Combined and including the updated TPS data, the overall mass of the Crew Capsule and Service Module is added to a total of 9,515 kg.

4.2.5. Summary

The crew capsule consisting of a re-entry capsule and its service module have been designed for a crew size of 4 persons and have a total mass of 9,515 kg. It is equipped with two solar arrays for power supply and a docking mechanism for attaching it to the habitat. The TPS for re-entry is based on new European technology using ceramic matrix composite tiles for protecting the spacecraft.

4.2.6. To be Studied and Additional Considerations

To design the TPS more thoroughly, realistic models for the re-entry have to be created and used to estimate the heat load on the individual parts of the spacecraft. It should include a detailed atmosphere model and rely on a specific mission analysis providing a re-entry angle based on the return trajectory of the spacecraft.

Also the capsule’s interior has to be designed especially the crew support components that need to fit into the capsule.

5. Launcher

5.1. Design Approach

For the preliminary design of the launcher stages, the following in-house (SART) tools have been used:

- Calculation for Aerodynamic Coefficients (CAC) 2.27 for the determination of the aerodynamics
- Space Transport and System Mass (STSM) 1.31 for the determination of the masses of components, stages and complete launchers
- Raumtransport System (RTS) 1.18 for the iterative ascent trajectory analysis
- Trajectory Optimization and Simulation for Conventional and Advanced Launchers (TOSCA_TS) 1.15 for the 2D ascent trajectory optimization and the determination of the payload performance
- Propellant Management Program (PMP) 0.9.2 for the preliminary design of the liquid propellant tanks

For the preliminary layout and calculations wrt the launcher a process as depicted in Figure 5-1 has been applied during the CERMIT study. As a first step the various first options for subsystems of the launcher have been selected, i.e. boosters and engines. Based on this preliminary calculations of possible configurations (e.g. with different booster types) have been conducted to determine feasibility and allow trades of advantages and disadvantages. Design outputs have been the maximum payload mass, total mass of the launcher and the reachable orbits (altitude and inclination). As a next step the relevance for the mission strategy is put into relation with the various launcher designs and one design (H-900) is picked for further optimization.

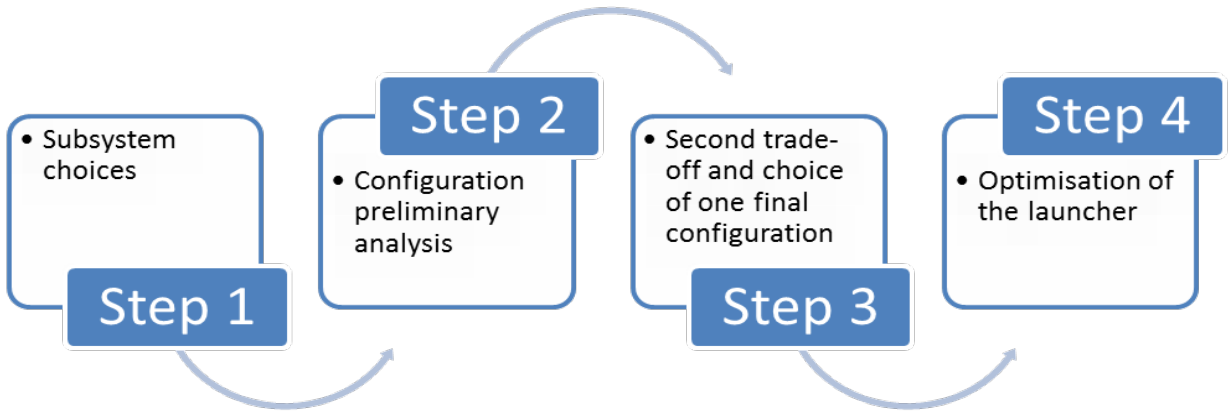

Figure 5-1: General design approach for the launcher as applied.

5.2. Requirements and Design Drivers

In total the launcher has to deliver a payload of 360 tons (2 modules and 4 transfer stages) into a LEO of an altitude of 300 km and with inclinations between 0° to 69.1° (depending on the actual target asteroid). To stay within a realistic scenario the maximum launch number for a single mission has been set to four – possibly also conducted by international partners.
The fairing has to be adaptable to the varying number of transfer stages that need to be transported into LEO (again depending on the actual target asteroid). Selected technologies should have a low development need to reduce cost and risk regarding launcher development. At last but not at least, it is relevant to favour European technologies.
During the study it has been decided that the actual crew will be launched on a separate already human rated launcher, to further reduce the development effort on the heavy lift launcher and the risk for the crew. This is an opportunity for international collaboration if a European launcher is not yet available.
The European launch site (Kourou) would be prominent for the mission. It has a longitude of -52.76° and a latitude of 5.4°.

5.3. Options and Trades

As a basic draft layout, a two stage heavy lift launcher composed of boosters, a cryogenic core stage powered by Vulcain 2 or Vulcain 3 engine and an upper stage comparable to Ariane 5's ECB (including the Vinci engine) is envisioned. As a preliminary label, the heavy lift launcher is named SIRIUS after the brightest known star.

Table 5-1: Booster options for SIRIUS including Étages d'Accélération à Poudre (EAP) of Ariane 5, the Liquid Flyback Booster (LFBB) from the DLR Astra study, and the Ukranian Zenit 1st stage.

	EAP	LFBB	Zenit (1st stage)
Max. thrust [kN]	6372	3 x 1622	8064
Sea level specific impulse [s]	251.4	367.2	309.5
Vacuum specific impulse [s]	274.1	421.7	337.2
Dry mass [kg]	36,535	54,802	38,000
Propellant mass [kg]	237,875	167,500	319,000
Propellant type	HTPB*	LH_2 – LOx	Kerosene-LOx
Note	Fully developed	To be developed	Ukraine production

*Hydroxyl-terminated polybutadiene

The various types of boosters as considered for SIRIUS are listed in Table 5-1 with characteristic parameters. The options include the fully developed and currently used ARIANE 5 booster Étages d'Accélération à Poudre (EAP), the Liquid Flyback Booster (LFBB) as designed during the internal DLR study ASTRA and the Zenit booster, which currently is used as 1st stage for the Ukrainian Zenit launcher and originally was developed for the Soviet Energya-Buran launcher.

The LFBB has the disadvantages of being in an early development and due to its wing configuration a maximum of 5 such boosters could be placed on a 10 m diameter core stage. Out of these reasons LFBB has been ruled out as booster option early in the study.
The Zenit booster uses the Russian (originally Soviet) rocket Engine RD-170 and a kerosene-LOx fuel. It is the most powerful engine ever built and significantly more powerful than solid boosters. In difference the EAP has the advantage of being a European product and also having significant flight heritage (like Zenit).

Table 5-2: Engine options for the core and upper stage of SIRIUS.

	Vulcain 2	Vulcain 3	Vinci
Max. sea leave Thrust [kN]	989	1308/ 1098*	n/a
Sea level specific impulse [s]	315	340.1/ 285.5*	n/a
Max. vacuum thrust [kN]	1359	1529/ 1699*	180
Vacuum specific impulse [s]	433	397.6/ 441.8*	465
Cycle	Gas Generator	Gas Generator	Expander

*for deployed double bell nozzle

The core stage options are named H (for Hydrogen propellant) and with a number indicating the propellant mass in tons (e.g. H600 denoting a propellant mass of 600 tons). Engine options (s. Table 5-2) for the core stage are Vulcain 2 and a proposed Vulcain 3 from DLR’s ASTRA study. It is mostly based on the Vulcain 2 and therefore would have limited development costs. Performance variation, resp. increase, can be gained by varying the number of engines on the core stage. Due to the better performance of Vulcain 3, e.g. an increase of more than 300 kN in thrust, and the reduced amount of development costs, this engine is selected as baseline for the core stage.

Table 5-3: Summary of the various launcher configurations.

Configuration	Zenit-H600-H32	EAP-H800-H32	Zenit-H800-H32	Zenit-H900-H32
Core Stage	H600	H800	H800	H900
Engine Type	Vulcain 3	Vulcain 3	Vulcain 3	Vulcain 3
Number of Engines	3	4	5	5
Total Thrust [kN]	1325.7	1767.2	2209	2209
Upper Stage	H32	H32	H32	H32
Engine Type	Vinci	Vinci	Vinci	Vinci
Number of Engines	1	2	1	2
Total Thrust [kN]	180	360	180	360
Booster Type	Zenit	EAP	Zenit	Zenit
Number	4	8	8	8
Total Thrust [kN]	32256	50976	64512	64512

The design of the upper stage, named analogously like above H32, is based on ARIANE 5 ME, which allows further benefit of already made development investments. Compared to

Vulcain 3, the Vinci engine is optimized for usage in vacuum has a very good specific impulse of 465 s (compared to Vulcain 3's 442 s) and therefore is selected for the upper stage. The complete launcher is selected out of a combination of these elements that is as optimal as possible, its number of boosters, engines and propellant mass. A summary of the different possible configurations is given in Table 5-3 wrt their performance parameters and in Table 5-4 wrt their mass breakdown. It should be noted that only Zenit-H600-H32 has a smaller lift-off mass than SATURN V, which was used for the United States' Apollo-Programme and had a gross lift-off-weight (GLOW) of 2,934.8 tons, whereas ARIANE 5 has a GLOW of about 780 tons (depending on exact version and payload). This shows that these proposed launcher configurations pose a significant milestone for European launcher development.

Table 5-4: Mass breakdown for the various launcher configurations, all masses include a 10% margin.

Configuration	Zenit-H600-H32[1]	EAP-H800-H32[2]	Zenit-H800-H32[3]	Zenit-H900-H32[3]
Booster [kg]	357000	283953	357000	357000
# of boosters	(4)	(8)	(8)	(8)
Prop. mass [kg]	319,000	242,304	319,000	319,000
Dry mass [kg]	38,000	41,649	38,000	38,000
Core stage [kg]	670,249	875,938	883,662	993,812
# of Vulcain	(3)	(4)	(5)	(5)
Prop. mass [kg]	608,745	811,060	811,660	913,118
Dry mass [kg]	61,504	64,778	76,641	80,695
Upper stage [kg]	53,451	54,314	53,451	55,200
# of Vinci	(1)	(2)	(1)	(2)
Prop. mass [kg]	33,000	33,000	33,000	33,020
Dry mass [kg]	20,429	21,292	20,429	22,180
Payload Mass [kg]	106,800	113,000	188,620	200,190
Total GLOW* [kg]	2,258,510	3,314,658	3,987,200	4,105,210

1 perigee: 200 km, apogee: 300 km, inclination: 7° 2 perigee: 208 km, apogee: 300 km, inclination: 20° 3 perigee: 190 km, apogee: 300 km, inclination: 69.1° *Gross Lift-Off-Weight

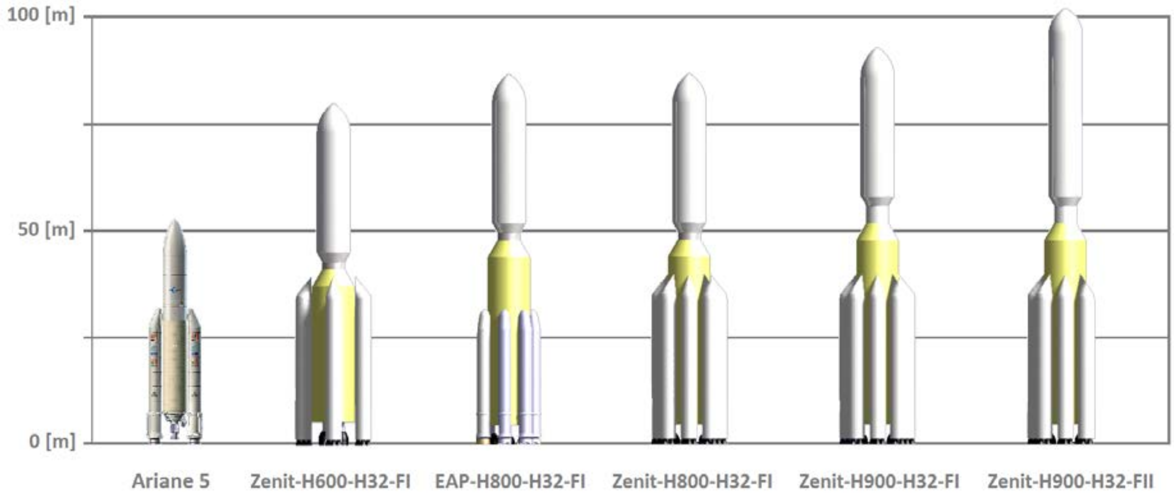

Figure 5-2: On-scale comparison of the various SIRIUS launcher configurations with ARIANE 5.

The mass calculations are based on a first optimisation of the ascend trajectory and consequently they include ascend propellant, reserves, residuals and propellant for the reaction control system. The trajectory calculations began for orbits with an inclination of 6° and incrementally have been increased to the 69.1° which has been the final upper limit determined during the study.
Depending on the actual payload combination of stages, two different fairing types will have to be used for the largest variant of the launcher (s. Section 5.3.1). A graphical comparison for the various launcher configurations is presented in Figure 5-2 along with a scale depiction of ARIANE 5. It is clearly visible that the launcher will be a significant enhancement on current European launcher technology, although the technologies as envisioned for SIRIUS are based on existing technology or are even already in use.
For further reference two of the launcher options, namely Zenit-H600-H32 and Zenit-H900-H32 will be labelled SIRIUS-L (light) and SIRIUS-H (heavy). During the elaboration of the study only SIRIUS-H has been further investigated.

5.3.1. Trade Off

Either of these two launcher variants has to lift a total of 360 tons of payload into orbit in a manageable launch scenario. SIRIUS-L can carry a payload of about 100 tons, SIRIUS-H of 200 tons, i.e. for the former four launches would be needed, for the latter only 2. Regarding logistics, SIRIUS-H appears as a better option as the preparation of four heavy lift launchers in a time frame short enough for a realistic mission time window is challenging.
For a comparable reason launcher options using the EAP are discarded as for one mission, i.e. four launches, 32 of these boosters would be needed in a time frame that drastically exceeds the current production rate (about 12 per year).
Summarizing the various payload stages, the respective masses are:

- AD stage: 36,358 kg
- AA stage: 84,256 kg
- ED1 stage: 99,911kg
- ED2 stage: 99,911kg
- Crew Capsule: 9,515 kg*
- Habitat Module: 27,890 kg

*will be transported by an existing, human-rated launcher, not SIRIUS

For launcher selection a realistic and optimal combination of the payloads is set up. In order to use only two launches, a suitable payload combination would be:

- Launch 1: ED1 + AA = 184,167 kg
- Launch 2: ED2 + AD + Habitat = 164,159 kg

However to reduce the amount of boil-off masses of the cryogenic stages (ED1 and 2) a time frame of 60 days should not be exceeded in orbit. Therefore a less uniform but more realistic approach would be:

- Launch 1: AD + AA + Habitat = 148,504 kg
- Launch 2: ED1 + ED2 = 199,822 kg

Only SIRIUS-H can handle a payload mass of about 200 tons, i.e. this would be the most viable option for a two launch scenario.

5.3.2. Open Issues

Further issues to investigate are listed in the following:

- A SIRIUS-H version with a smaller number of boosters to accommodate 150 tons of payload (Launch 1 option above)
- The difference of the cost between the development of H600, H800 and H900
- The impact on the mission if ED1 and ED2 are launched separately
- Minimization of dwelling time between Launch 1 and Launch 2 to avoid propellant losses (boil-off)

Table 5-5: Mass breakdown structure: Zenit Booster.

Zenit booster	[kg]
Structure Group:	
Mass Structure group:	38000
Mass Structure group: including 0.0 %* margins	38000
Stage Mass empty: (stage coordinates)	38000
Stage Mass empty incl.marg.: (global coordinates)	38000
Orbit/De-orbit propellant:	0
Residual propellant:	0
Reserve propellant:	0
Stage Mass @ burn out:	38000
Difference to MECO Mass from Trajectory Analysis:	1
RCS propell. /inert flow mass:	0
Ascent propellant:	319000
GLOW Stage Mass:	357000

* margins have not been considered for this existing design

Figure 5-3: The Zenit booster.

5.4. Baseline Design

During the course of the study, it has been decided to pursue the SIRIUS-H variant (Zenit-H900-H32) for further analysis. It is depicted in the following summary.

Table 5-6: Mass breakdown structure: H900 core stage.

H900 core stage (5 Vulcain 3 engines)		[kg]
Structure group:		
Mass Structure group:	w/o margins	58886
Mass Structure group: margins	including 10.0 %	64775
Propulsion group:		
Mass Propulsion group:	w/o margins	14473
Mass Propulsion group: margins	including 10.0 %	15920
Stage Mass empty: (stage coordinates)		73359
Stage Mass empty incl. margin: (global coordinates)		80695
Residual propellant:		5400
Reserve propellant:		7493
Stage Mass @ burn out:		93587
Difference to MECO Mass from Trajectory Analysis:		84
RCS propell. /inert flow mass:		225
Ascent propellant:		900000
GLOW Stage Mass:		993812

Figure 5-4: H900 core stage.

5.4.1. Mass Breakdown Structure

Table 5-5, Table 5-6 and Table 5-7 list the mass breakdown for each major component of SIRIUS-H, whereas Figure 5-3, Figure 5-4 and Figure 5-5 provide a visual representation for the individual parts. As given in Table 5-4 the total launch mass for SIRIUS-H are 4105 tons with a payload mass of 200 tons.

5.4.2. Performance

The results of the trajectory analysis for SIRIUS-H are presented in Table 5-8 and Table 5-9 as well as Figure 5-6. As can be seen in the first graph of this figure, the maximum acceleration at the booster's end of burn is nearly 6g, which is larger than for common launchers (e.g. ARIANE 5's maximum acceleration is 4.5g [RD 4]). To reduce this acceleration, the boosters could be throttle the boosters – Zenit can be throttled down to 50%, which would reduce the performance. However due to optimization of the global staging and especially the upper stage, the initial performance is still achievable. Similar considerations are true for the maximum dynamic pressure.

Table 5-7: Mass breakdown structure: H32 upper stage.

H32 upper stage (2 Vinci engines)	[kg]
Structure group:	
Mass Structure group: w/o margins	16642
Mass Structure group: including 10.0 % margins	18307
Subsystem group:	
Mass Subsystem group: w/o margins	1890
Mass Subsystem group: including 10.0 % margins	2079
Propulsion group:	0
Mass Propulsion group: w/o margins	1440
Mass Propulsion group: including 10.0 % margins	1584
Thermal protection group:	
Mass Thermal protection group : w/o margins	194
Mass Thermal protection group : including 10.0 % margins	213
Stage Mass empty: (stage coordinates)	20167
Stage Mass empty incl.marg.: (global coordinates)	22183
Residual propellant:	242
Reserve propellant:	480
Stage Mass @ burn out (fairing separated):	10720
Payload Mass:	200189
RCS propell. /inert flow mass:	100
Ascent propellant:	32200
GLOW Stage Mass:	55205

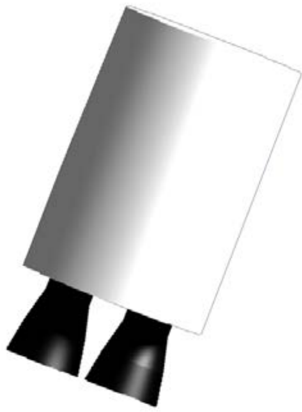

Figure 5-5: H32 upper stage.

The second graph of Figure 5-6 depicts the mass evolution of SIRIUS-H during its ascent. Each change of curve slope corresponds to a stage separation – the durations of each stage's burn are summarized in Table 5-8.

Table 5-8: Timeline for SIRIUS' ascent phase.

Event	Time [s] (T_0=0, Launch Time)
1st stage separation (boosters)	131
2nd stage separation (core stage)	459
3rd stage separation (upper stage)	1259
Fairing separation	228

Table 5-9: Design values during SIRIUS' ascent.

Parameter	Value
Maximum velocity	7.67 km/s (@864 s)
Maximum acceleration	5.74 g (@131 s)
Maximum dynamic pressure	58.6 kPa (@62 s)

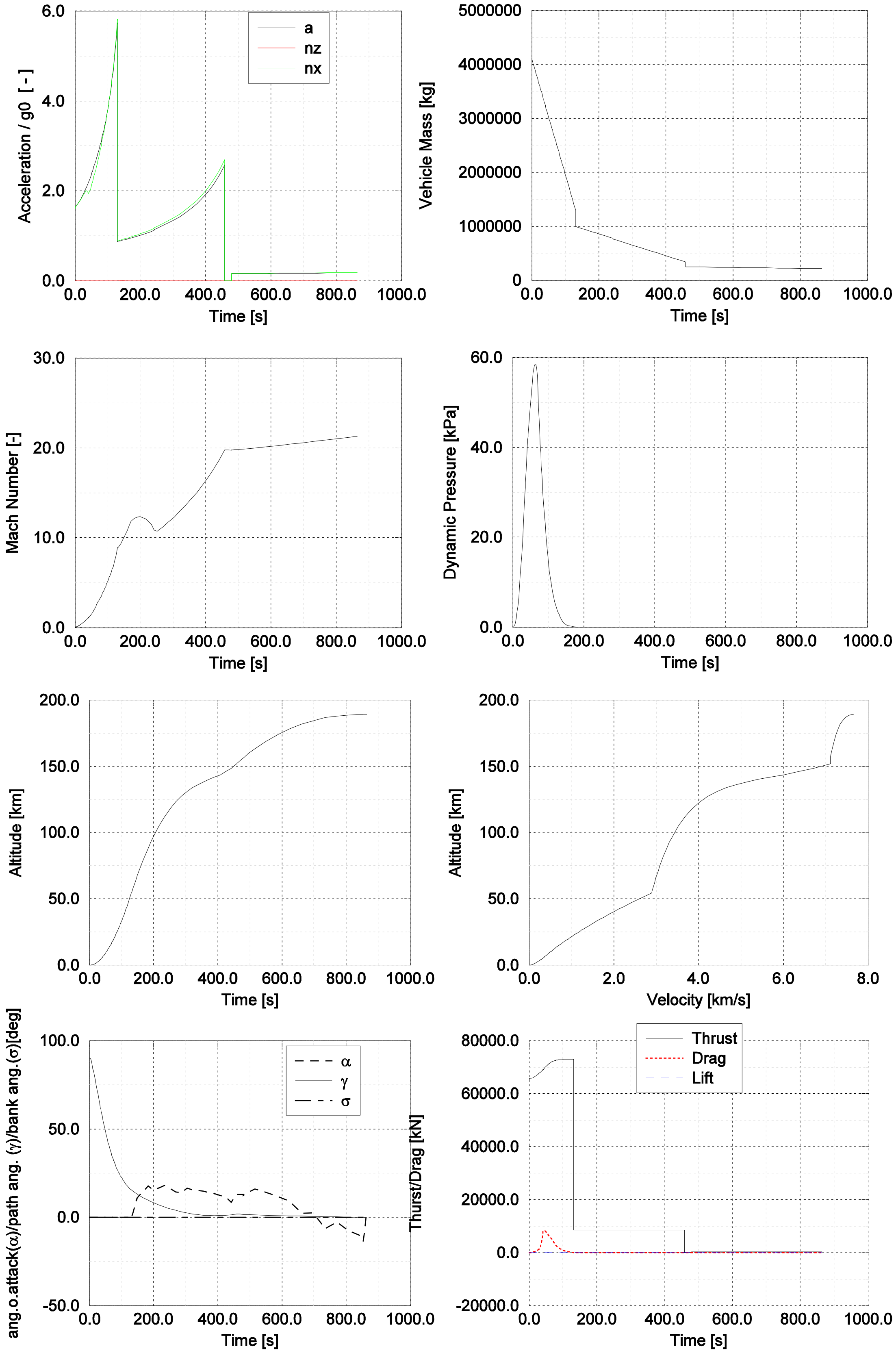

Figure 5-6: Depiction of the SIRIUS-H ascent trajectory into a 190 x 300 km orbit and i = 69.1°.

5.5. Tanks

For the preliminary design of the liquid propellant tanks of the launchers core stage, the SART tool PMP has been utilized. For the cryogenic H900 stage, a separated tank configuration has been applied.
This way the boosters can be attached at the stiff interstage structure between both tanks. To be able to load 900 tons of LOx and LH_2 a tank diameter of 10 m has been selected, which gives a total length for both tanks of 40.4 m. The H900 propellant tanks are depicted in Figure 5-7. The LOx is stored in the smaller tank, the LH_2 in the large one.

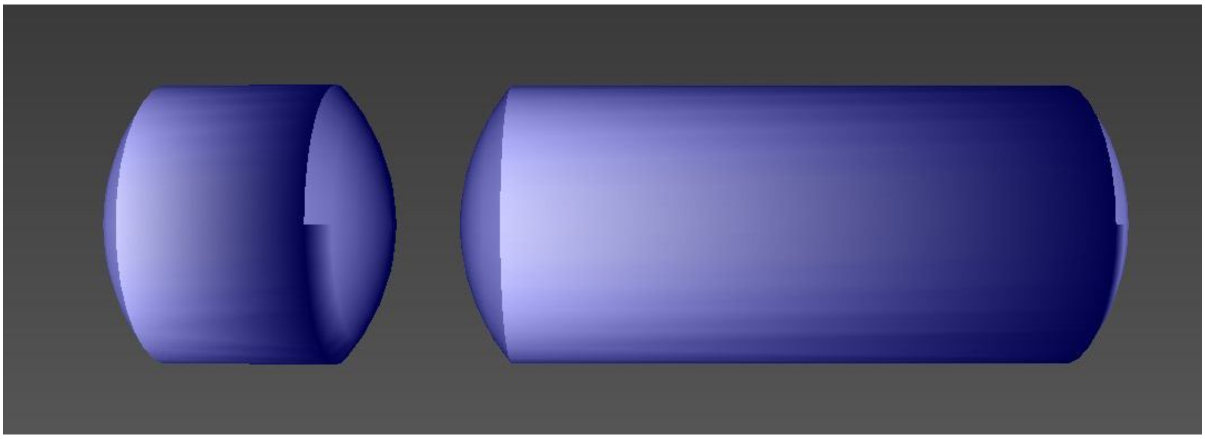

Figure 5-7: LH_2 (right) and LOx (left) propellant tank design for the H900 core stage.

5.6. Additional Considerations

Generally the SIRIUS-H variant is a very drastic increase in capability and technological requirements compared to any previous launcher and especially European technology. While it allows a more relaxed launch campaign, the demands are exceeding even those of the most prominent launcher ever built, Saturn V.
However drafting a launcher with comparable characteristics is not a new occurrence. During the 1960s NASA conducted studies on heavy lift launchers that could carry 210 tons or even 500 tons into orbit, by advancing Saturn V to Saturn C-8 [RD 5].
or designing a completely new launcher, labelled Nova. These concepts were abandoned due to the fact that no plans for missions beyond Moon ever manifested themselves [RD 6].

Besides allowing missions of the kind of CERMIT SIRIUS-like launchers could be used for missions into LEO, especially for building up infrastructure of large sizes, e.g. an ISS successor or even at Moon, possibly the Earth-Moon Lagrange-Points. Considering the very extreme size of SIRIUS-H it appears more likely that SIRIUS-L could be reused.

Although SIRIUS-H has the benefit of a smaller launch number, it is by far exceeding anything that has been built before, whereas SIRIUS-L is closer to current European technology and previous launcher capabilities (Saturn V had a payload mass of about 120 tons). Therefore regarding the technological challenge and risk, it might be a more viable option.

5.7. Summary

Several versions of possible heavy launchers have been designed during the CERMIT study, based on existing launcher technology and know-how in Europe. Two basic configurations, one with a payload mass into orbit of about 100 tons and one with 200 tons, labelled SIRIUS-L and SIRIUS-H respectively, have been investigated further. Due to the reduced number of launches of SIRIUS-H in comparison to SIRIUS-L, i.e. two versus four, the former has been selected in the study as a baseline.

6. Transfer Stages

6.1. Requirements and Design Drivers

For the mission scenario as described earlier, three maneuvers are necessary, namely the:

- Earth departure maneuver (EDM),
- the asteroid arrival maneuver (AAM) and
- the asteroid departure maneuver (ADM).

A group of six asteroids has been selected as example pool of mission targets to keep the mission flexible and realistic. For each asteroid a minimum launch window of one month has been assumed around the minimum mission ΔV and the maximum ΔV for each window became the design case for the given target, whereas the worst case of these six targets became the design case for the stages (i.e. less demanding missions can be achieved with less than the four stages, because one stage can be used for more than one maneuver).
Besides payload mass and ΔV and important design driver is the dwelling time in space for each stage, which mostly influences the choice of propellant. The combination of LOx and LH_2 allow large values for the specific impulse (about 450 s), but these components have to be stored at temperatures of 70 K resp. 20 K, which is a significant drawback. First, heat transfer between the two tanks leads to boil-off of the LH_2 and freezing of the LOx. Second, solar radiation further heats the tanks, increasing the boil-off of LH_2. In general even with Multi-layer-insulation (MLI) it is impossible to store these cryogenic propellants for long durations in space.
Consequently the asteroid maneuvers cannot be executed by cryogenic propellant – a suitable alternative is monomethylhydrazine (MMH) and mixed oxides of nitrogen (MON), which is storable and hypergolic (i.e. it can be stored for several years and is easy to ignite by simply putting the two propellant components into contact).
To reduce the development effort the same transfer stages should be used for all possible target asteroids and launch dates. To provide the necessary ΔV the propellant loading is adapted accordingly.

For the transfer stage engine only European technology has been considered, in accordance with requirement ST-DE-0030. For staging using LH_2 and LOx the Vinci engine has been selected, due to its large specific impulse and moderate thrust. For storable propellants the Aestus 2 engine has been chosen, which does not exist yet, but the combustion chamber has already been tested and provided an improved performance over Aestus. The properties of both engines are summarized in Table 6-1.

Table 6-1: Engine options for the transfer stages.

	Vinci	Aestus 2
Thrust (vacuum) [kN]	180	55
Specific impulse (vacuum) [s]	465	336
Cycle	Expander	Gas Generator
Propellant	LH_2/ LOx	MMH/MON

6.2. Options and Trades

Early in the design process it has been decided that the Earth departure maneuver should be conducted with two identical cryogenic stages (to reduce development costs). For the worst and thus design case (launch from LEO towards 1999 AO_{10} on 11^{th} August 2025) about 180,000 kg of propellant are required for this manoeuver. While a single stage could in less demanding cases lead to a reduced launch mass due to an improved structural index, it would be more difficult to keep up the modular approach and adjust the number of transfer stages for a wider range of missions. However for some targets and launch dates only one stage is needed for Earth departure and another one for the asteroid manoeuvers.

For the EDS two configurations have been investigated – parallel and sequential mounting. The former solution would mean first of all either an increased fairing diameter or increased effort for docking. It would also be difficult to conduct stage separation if one stage has been emptied but the engine has still to run on the other one. Therefore a sequential configuration has been selected, where each stage has an own engine.

Two additional stages have been designed for the asteroid manoeuvers. For simplicity and to keep the vehicle mass at the asteroid low, one stage has been designed to exclusively conduct the AAM and a second one to conduct the ADM. This way stage separation during a manoeuver is prevented. The separation of the first asteroid transfer stage occurs after arrival at the asteroid. Consequently a delay in the ignition of one engine has just a very limited impact on the mission. This would not be the case if it would happen in the middle of a manoeuver.

For some cases the ΔV required for the asteroid manoeuvers are low enough to use only one of the two asteroid transfer stages, which then executes both manoeuvers.

6.3. Baseline Design

The stages have been sized in the reverse order of their use, as they have to be designed depending on their payload and ΔV (which have been subject to a margin of 5%). The relation between the stage dry mass and the propellant mass has been determined based on existing stages. For storable propellant, the data has been limited to ARIANE 4 and Proton stages. For cryogenic (LOx/LH_2) stages more extensive data has been used.

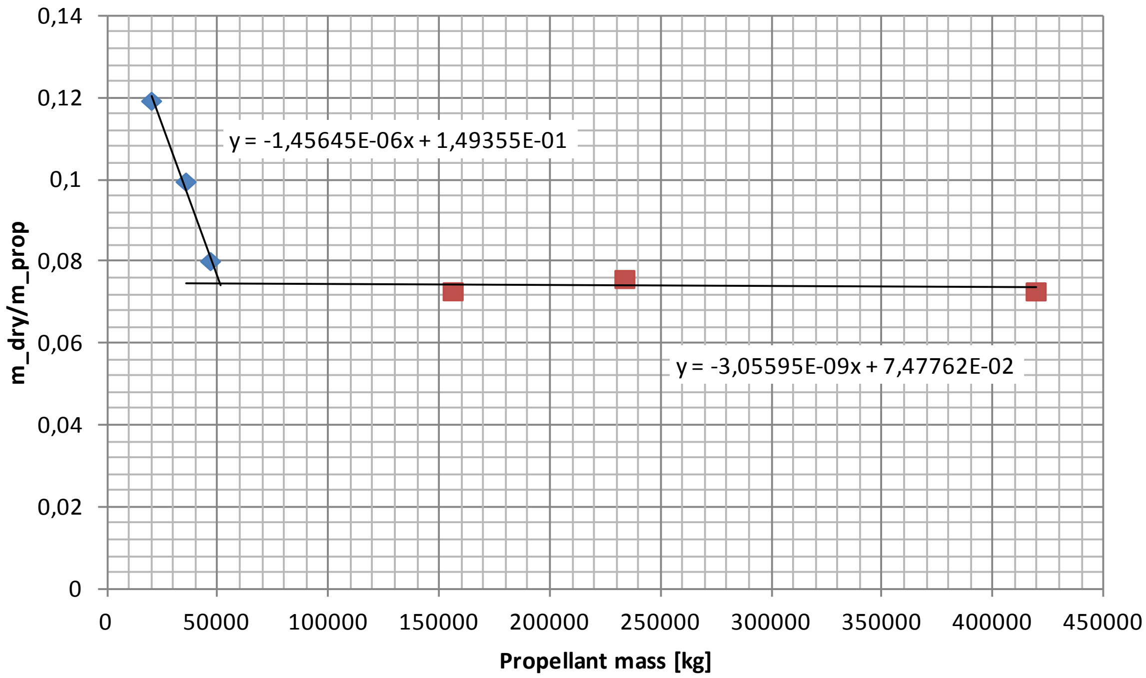

Figure 6-1: Evolution of the dry mass to propellant mass ratio over propellant mass for MMH/MON.

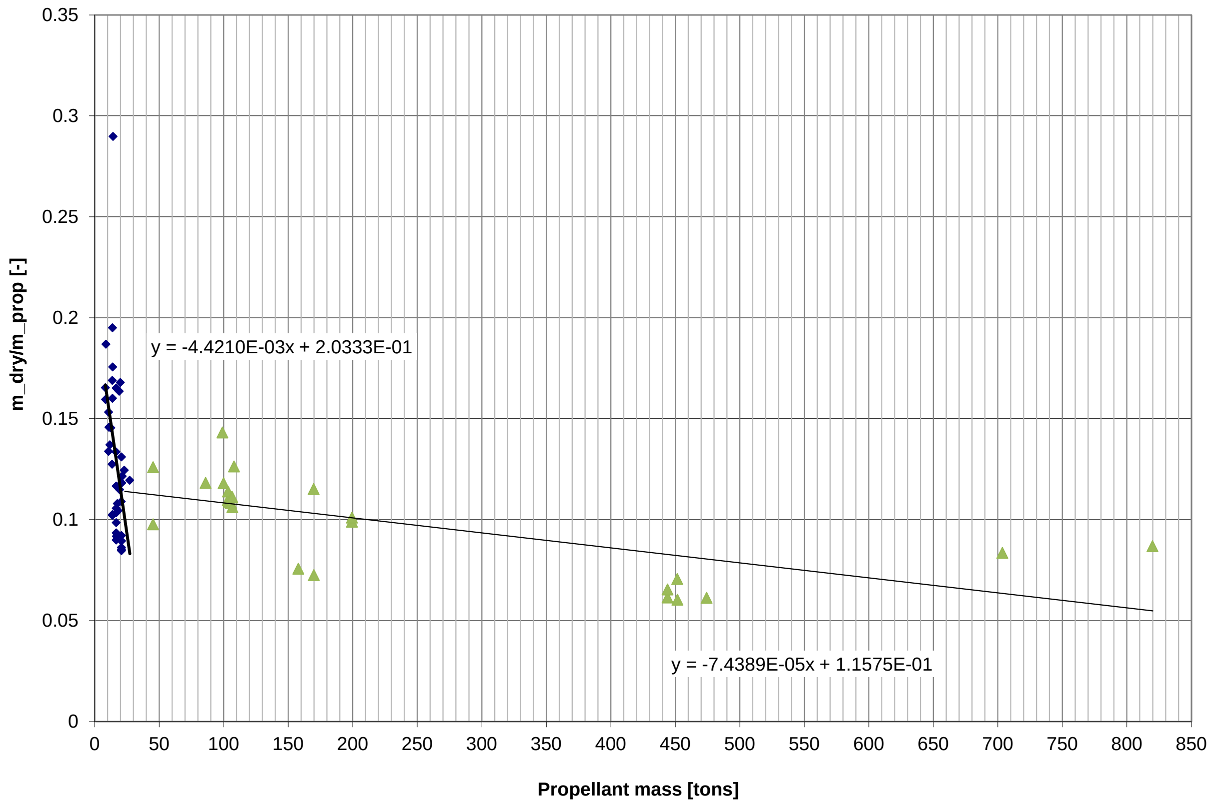

Figure 6-2: Evolution of the dry mass to propellant mass ratio over propellant mass for LH_2/LOx.

For each of these two propellant combinations, this relation has been modeled with two linear trend lines (see Figure 6-1 and Figure 6-2). More complex trend lines are too time consuming for the frame of this study. In this sizing both the dry mass and the propellant mass are unknowns and interdependent. No margin has been taken into account on the dry mass, as adding a margin on the propellant leads to a margin on the dry mass trough this relation between the two variables.
The first graph Figure 6-1 represents the relationship for the structure index (dry mass over propellant mass) and the propellant mass for MMH/ MON. For propellant masses of 50000 kg or below, the structural index follows the equation $y = -1.45 \cdot 10^{-6} \cdot x + 10.15$, for those masses over 50000 kg it is $y = -3.05 \cdot 10^{-9} \cdot x + 0.075$.
Figure 6-2 shows the same relation for LH_2/LOx. For a propellant mass of 30,000 kg or less, the equation $y = -4.42 \cdot 10^{-3} \cdot x + 0.2$ is applied, for larger masses $y = -7.5 \cdot 10^{-5} \cdot x + 0.12$.

6.3.1. Asteroid Departure Stage

The worst and thus design case for the AD occurs for a launch towards 1999 CG_9 on 3rd September 2033 with a ΔV requirement of 1.86 km/s (w/o margin). The payload for this stage consists only of the crew module (crew capsule and habitat) with a total mass of 37,405 kg. In combination of with the parameters for the Aestus 2 engine, this results in the following size of the AD stage:

- Propellant mass: 33,015 kg
- Dry mass: 3,343 kg.

For other, less demanding asteroid targets or launch dates the propellant mass can be reduced.

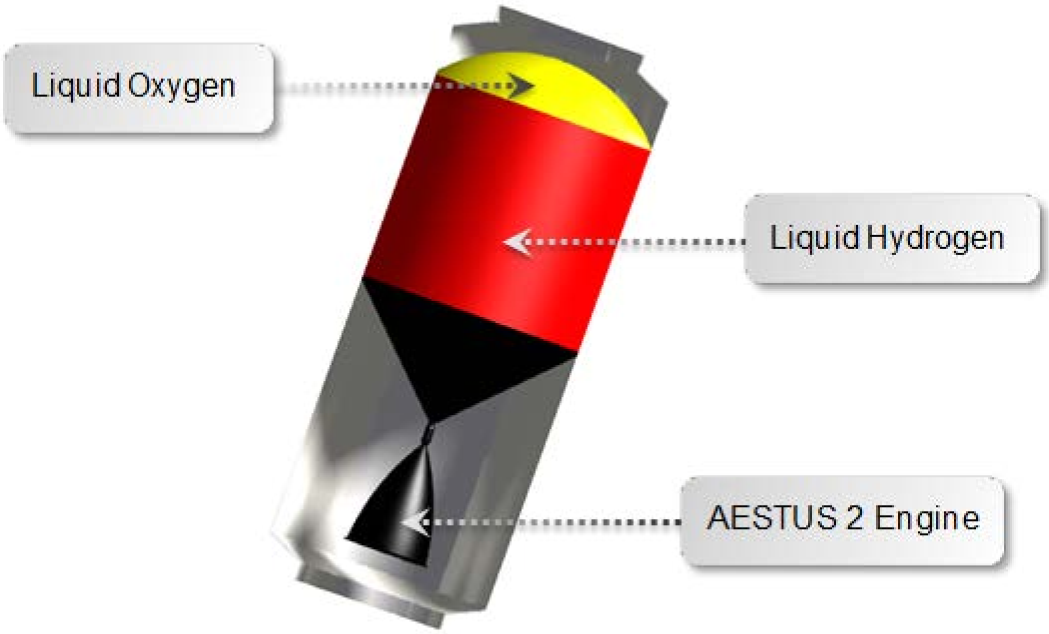

Figure 6-3: AD stage with AESTUS 2 engine.

An overview over the layout of the AD stage is given in Figure 6-3.

6.3.2. Asteroid Arrival Stage

For the sizing of the AA stage, the worst case has been the launch towards 1999 AO_{10} on 11th August 2025. In this case the ΔV requirement for the AA maneuver is 2.37 km/s (w/o margin). The payload for this stage comprises the crew module (crew capsule and habitat) and the AD stage, resulting in a total mass of 63,870 kg (for this case the propellant mass is 23,120 kg). This gives the size of the AA stage as follows:

- Propellant mass: 78,410 kg
- Dry mass: 5,845 kg

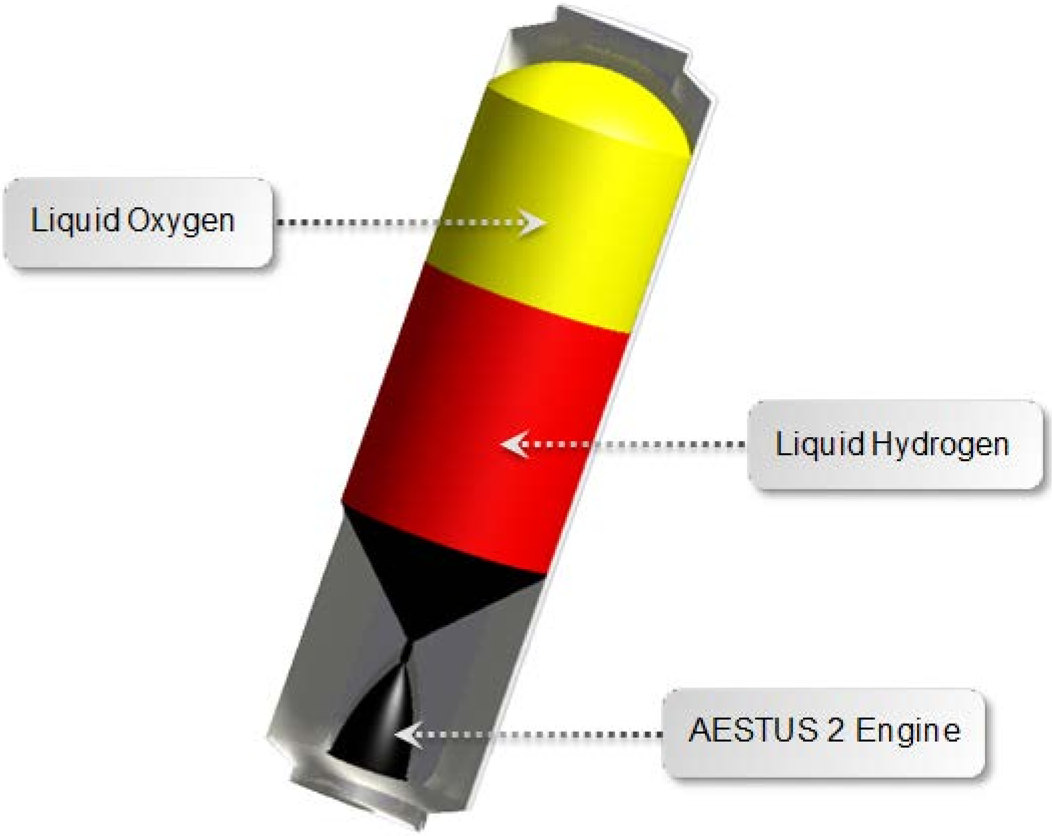

Figure 6-4: AA stage with AESTUS 2 engine.

Figure 6-4 provides a depiction of the AA stage as designed during the CERMIT study.

6.3.3. Earth Departure Stage

The design case for the ED stages is also the travel towards 1999 AO_{10} on 11th August 2025. The ΔV for this case is 3.25 km/s (w/o margins). For these two identical stages the Vinci engine is used and the maneuver is divided in a way that both stages are identical, i.e. the first stage covers about 40% of the total ΔV, the second one the remaining 60% (with the same propellant mass). Consequently the sizes of these two stages are:

- Propellant mass: 89,860 kg
- Dry mass: 10,050 kg.

Figure 6-5 shows the layout of the ED stages (identical) with the Vinci engine.

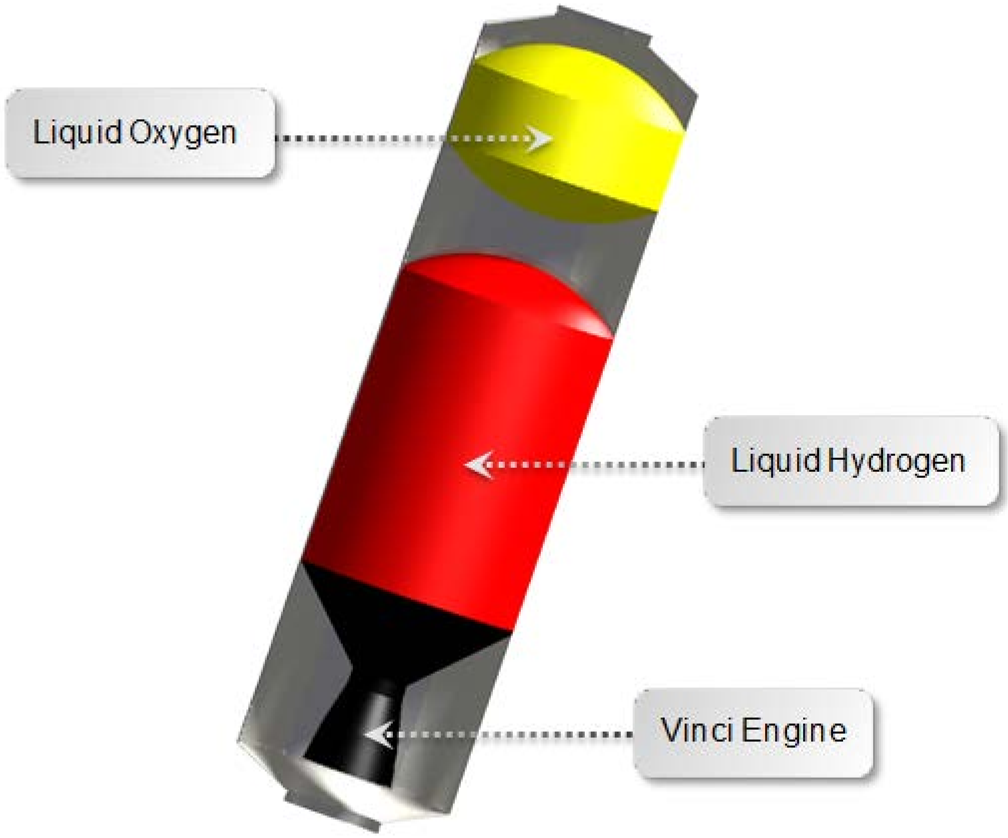

Figure 6-5: ED stage with Vinci engine.

6.3.4. Summary

The calculations above are summarized in Table 6-2 to provide a complete overview over the design.

Table 6-2: Baseline design of the transfer stages.

	AD stage	AA stage	ED 2 stage	ED 1 stage
Structure mass [kg]	3,343	5,845	10,051	10,051
Propellant mass [kg]	33,015	78,412	89,861	89,861
Total mass [kg]	36,358	84,256	99,911	99,911
MR	2.09	2.09	5.8	5.8
Engine	Aestus 2	Aestus 2	Vinci	Vinci
Mass flow [kg/s]	16.68	16.68	39.47	39.47
Specific impulse (vacuum) [s]	336	336	465	465
Propellant	MMH/MON	MMH/MON	LH_2/LOx	LH_2/LOx

6.4. Tank Description

6.4.1. Asteroid Departure Stage

The MMH/MON propellant tanks for the AD stage are depicted in Figure 6-6, where the smaller compartment stores the MMH and the larger one the MON. For the above mentioned design case (propellant mass of 33,015 kg) a common bulkhead with a tank diameter of 3.6 m is selected. The common bulkhead has been chosen to reduce the mass and stage height in comparison to separated tanks.

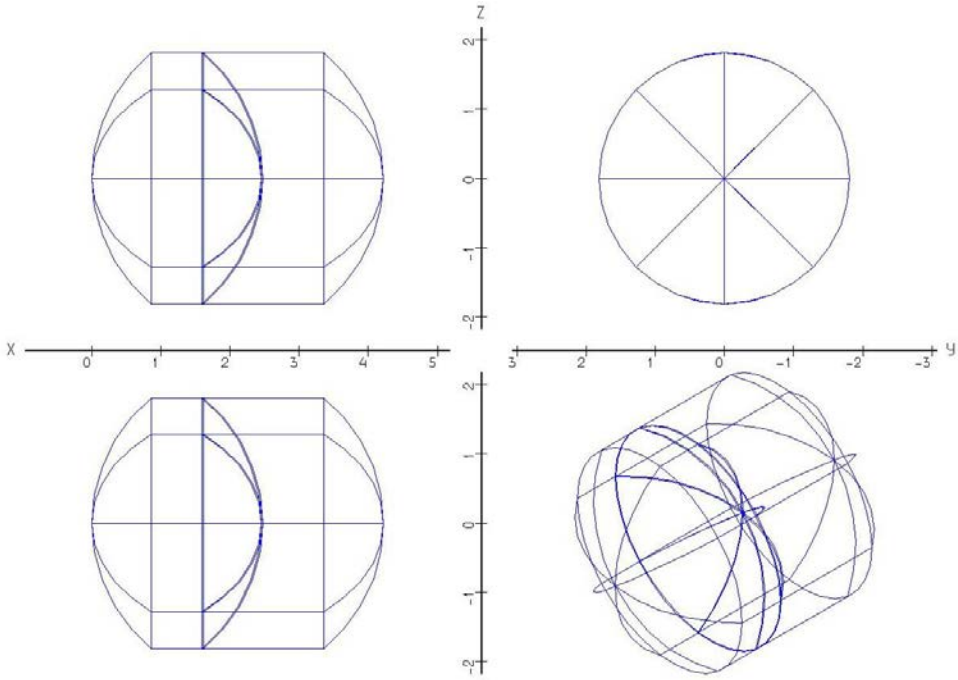

Figure 6-6: MMH/MON tank design for the AD stage.

6.4.2. Asteroid Arrival Stage

For the AA stage an identical diameter as for the AD stage (3.6 m) has been selected to simplify the positioning under the fairing. This diameter results in a tank length of 8.9 m for the required propellant mass of 78,410 kg MMH/MON. For the same reasons as above, a common bulkhead design has been selected. In Figure 6-7 the left tank compartment stores the MMH, the right one the MON.

6.4.3. Earth Departure Stage

For the ED stage 89,860 kg of propellant are required, including boil-off mass for a stay of 30 days in orbit on top of the 1 month launch window. The stage tank design is depicted in Figure 6-8, where the smaller tank holds the LOx, the other one the LH_2. The diameter for both tanks is 5.4 m and the total length is 14.7 m. The separated tank design is chosen to

reduce the amount of insulation (and thus mass) between the two tanks to prevent heat exchange between the LOx and LH_2 part (70 K and 20 K respectively).
For minimizing the propellant boil-off mass during the stay in orbit, 50 layers of MLI with an estimated mass of 1 kg per square meter are assumed for both propellant tanks [RD 7].

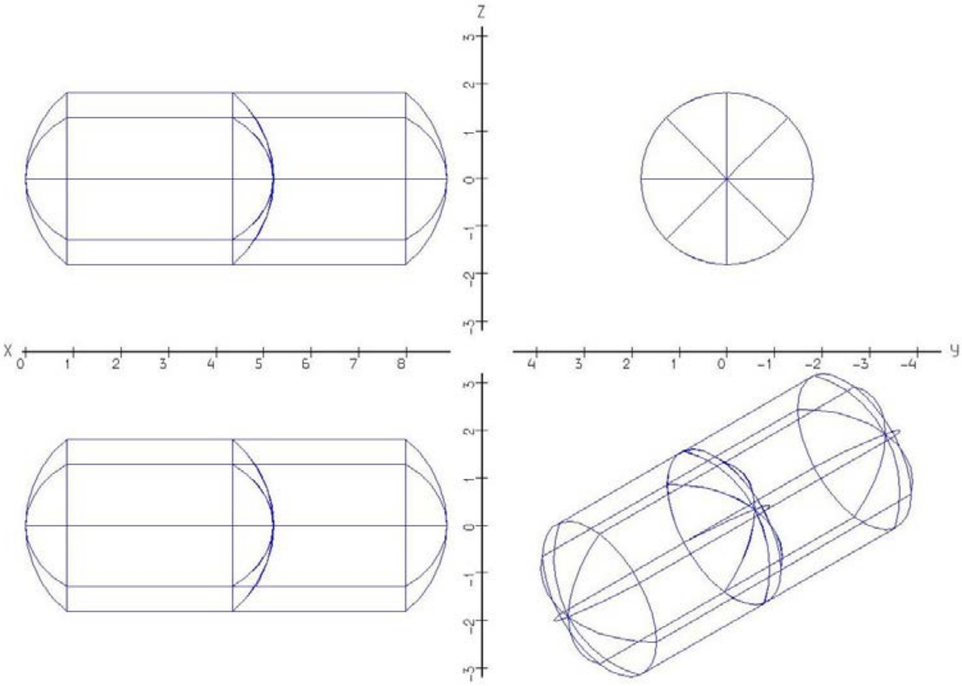

Figure 6-7: MMH/MON tank design for the AA stage.

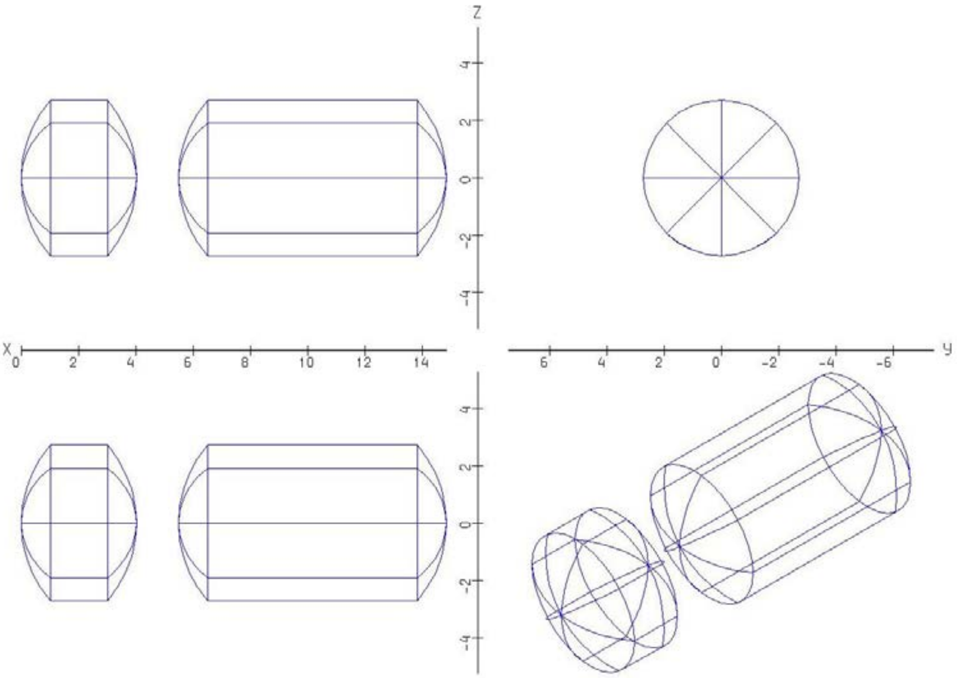

Figure 6-8: LH_2/ LOx tank design for the ED stages.

7. Configuration

For the visualization of the study results CAD models have been created by the study team using CATIA.

7.1. Requirements and Design Drivers

The design drivers and requirements for the configuration have been the input from all other domains, e.g. the usage of which booster type, number of rocket stages and dimensions for the stages.

Table 7-1: Baseline design of the transfer stages.

Launcher				
Name	Zenit-H600-H32	EAP-H800-H32	Zenit-H800-H32	Zenit-H900-H32
Length [m]	79.13; n/a*	86.66, 95.89	86.66, 95.89	92.54; 101.77
Diameter [m]	17.2	16.4	17.2	17.2
Booster				
Name	Zenit	EAP	Zenit	Zenit
Amount	4	8	8	8
Length [m]	39.6	31.51	39.6	39.6
Diameter [m]	3.9	3.2	3.9	3.9
First Stage (Core Stage)				
Length [m]	31.8	39.3	39.3	43.05
Diameter [m]	10	10	10	10
Engine	Vulcan 3	Vulcan 3	Vulcan 3	Vulcan 3
No. of Engines	3	4	4	5
$Mass_{prop}$ [ton]	600	800	800	900
Second Stage (Upper Stage)				
Length [m]	7.9	7.9	7.9	7.9
Diameter [m]	5.4	5.4	5.4	5.4
Engine	Vinci	Vinci	Vinci	Vinci
No. of Engines	1	1	1	2
$Mass_{prop}$ [ton]	32	32	32	32
Fairing I				
Payload	AAS, ADS, HM	AAS, ADS, HM	AAS, ADS, HM	AAS, ADS, HM
Length [m]	35	35	35	35
Diameter [m]	8	8	8	8
$Mass_{prop-AAS}$ [ton]	78.4	78.4	78.4	78.4
$Mass_{prop-ADS}$ [ton]	33	33	33	33
Fairing II				
Payload	EDS 1, EDS 2	EDS 1, EDS 2	EDS 1, EDS 2	EDS 1, EDS 2
Length [m]	44	44	44	44
Diameter [m]	8	8	8	8

*Fairing II not applicable

An overview about the design criteria is given in Table 7-1. There have been several design iterations and variations for the launcher (s. Figure 5-2), eventually the Zenit-H900-H32 (SIRIUS-H) has been selected for further investigation by the design team.

7.2. Baseline Design

SIRIUS-H has 8 Zenit boosters, a core stage with 5 Vulcan 3 engines (s. both Figure 7-1) and an upper stage with two Vinci engines (s Figure 7-2). The AAS and ADS have both one AESTUS 2 engine (s. Figure 7-3). EDS1 and EDS2 are equipped with a Vinci engine (s. Figure 7-4).

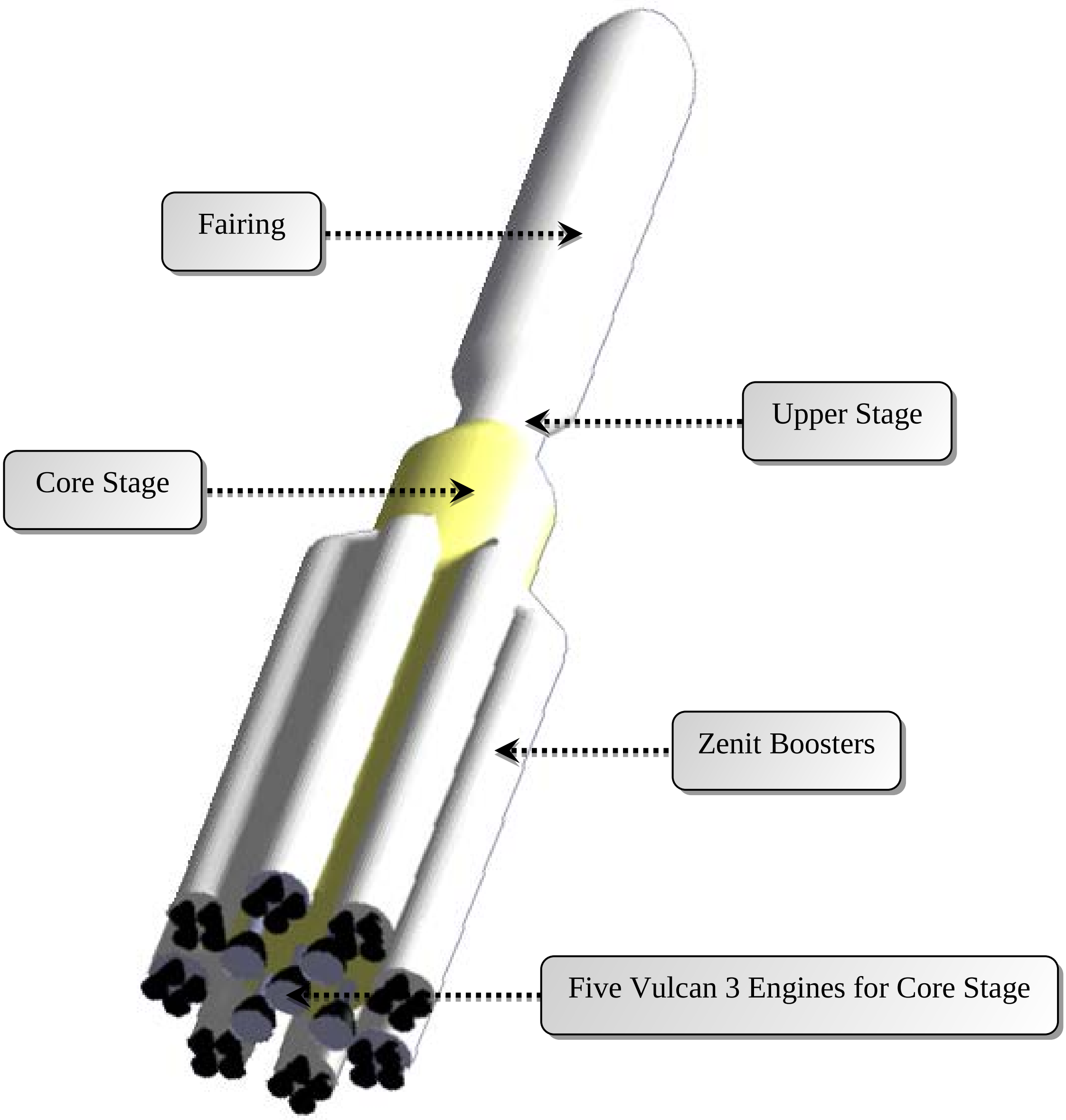

Figure 7-1: Bottom view of SIRIUS-H with Fairing II.

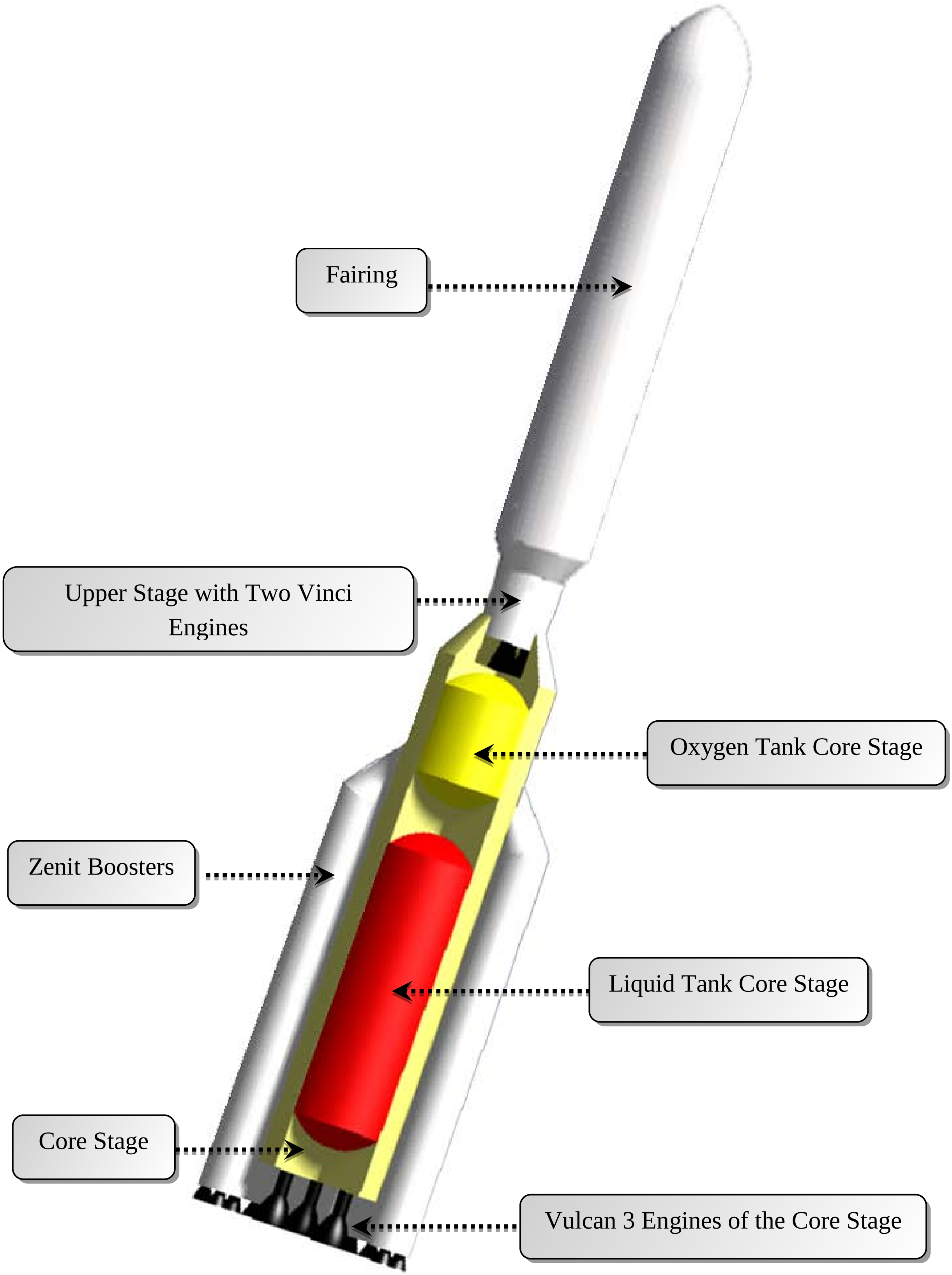

Figure 7-2: Main elements of SIRIUS-H with Fairing II.

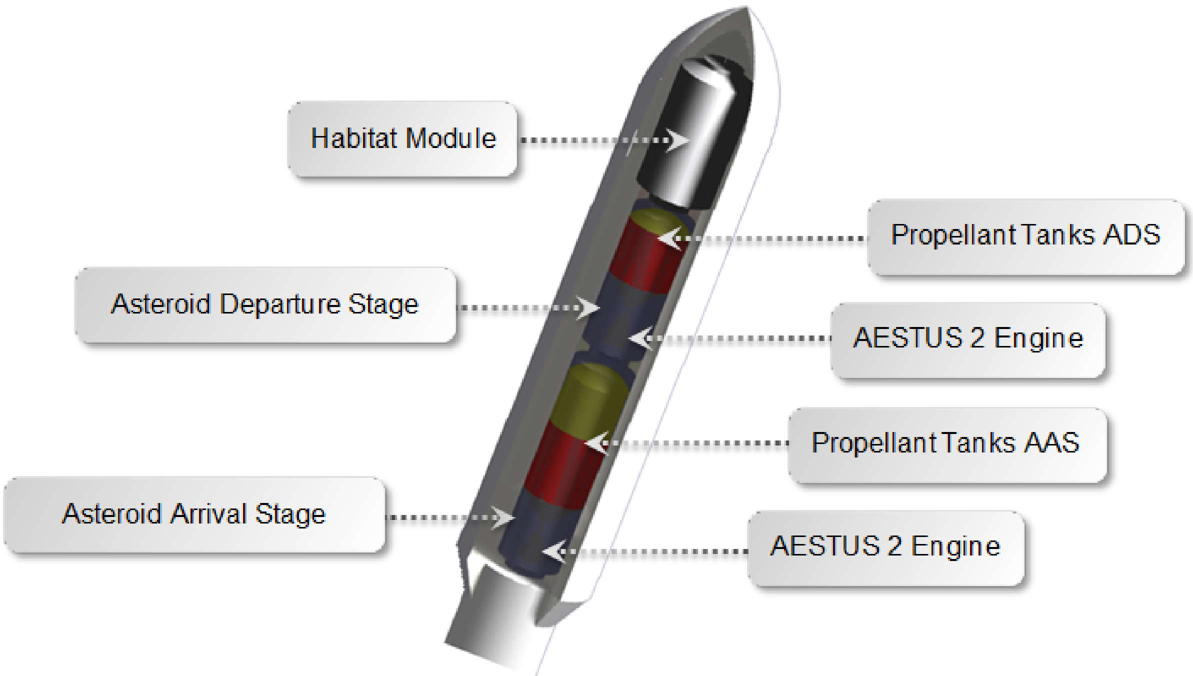

Figure 7-3: Cross-section view of Fairing I, containing AAS, ADS and the habitat.

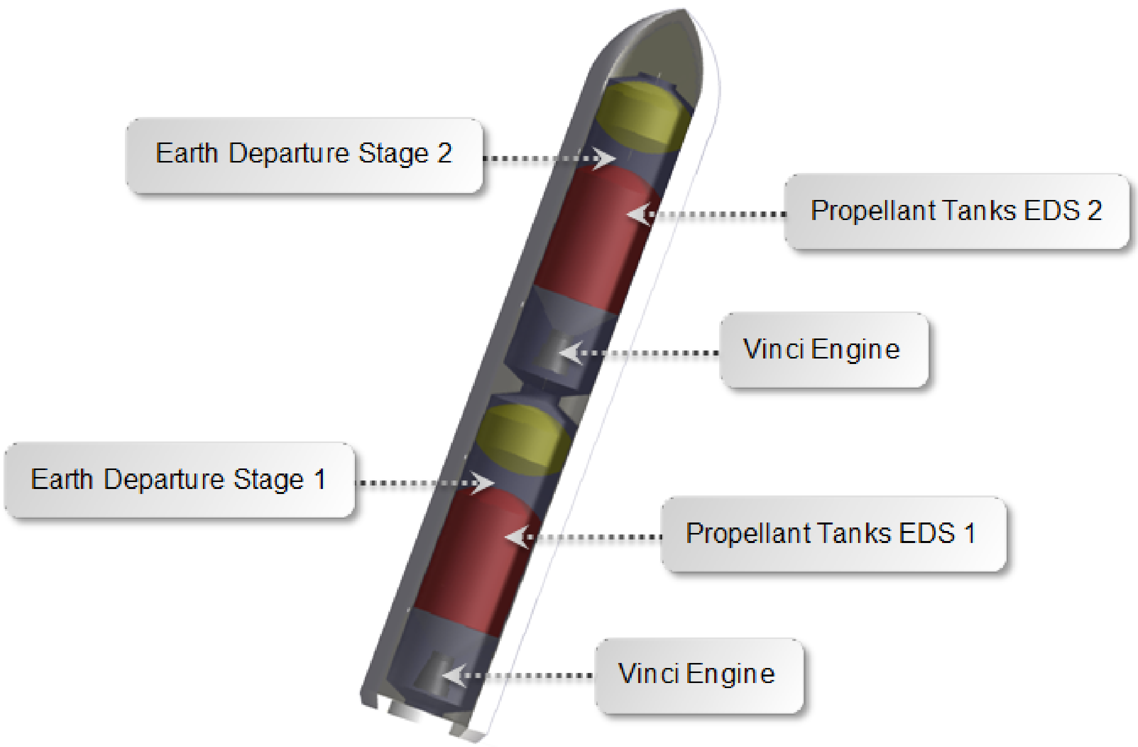

Figure 7-4: Cross-section view of Fairing II, containing the two ED stages.

To cover the two possible launch options, i.e. either carrying EDS 1 and 2 or AAS, ADS and the habitat into orbit, there have to be two different fairings, labelled here Fairing I (F I) for the latter case and Fairing II (F II) for the former.

8. Overall Mission Strategy

8.1. Assumptions

Major assumptions for the overall mission strategy are the European prominence in the mission design as well as the timeframe of about 2020 to 2040, with more probability towards the later parts of this.

While international cooperation and partners are not only allowed but also envisioned, the overall leadership of the mission, as laid out during this study should lie with Europe. Consequently technology and infrastructure used for the mission should be available in Europe.

For the detailed mission plan it has been assumed that for the design case all mission maneuvers at the asteroid target are conducted by individual stages and the transfer trajectory injection is split into two stages as this is in all cases the most demanding maneuver wrt ΔV. There are three maneuvers necessary for the mission:

- Earth Departure Maneuver (EDM)
- Asteroid Arrival Maneuver (AAM)
- Asteroid Departure Maneuver (ADM)

It has also been assumed that no maneuver at Earth occurs for re-entry, but that the spacecraft will execute a direct re-entry, applying aerocapture. This significantly relaxes the ΔV requirements for the missions and thus the overall mass, as any propellant used for such a maneuver would need to be carried along through the whole mission, driving necessary propellant masses for the previous maneuvers.

8.2. Requirements and Design Drivers

Design drivers for the mission strategy are the logistics of transporting the mission relevant components into space and from there towards the target asteroid. Requirement ST-PE-0020 states that the transfer stage system should have a total ΔV-capability of at least 7 km/s.

In a backwards calculation beginning with the mission payload of 40,000 kg (crew capsule and habitat, requirement ST-PE-0010) the sizing of the various transfer stages has been conducted, which in turn provides the data for the masses that need to be carried into orbit by a launch vehicle. The detailed designs for the launcher and the transfer stages are provided in Sections 5 and 6.

The total mass of components that need to be transported into orbit is 360,000 kg, with individual masses of close to 100,000 kg (for the Earth Departure stages).

8.3. Options and Trades

At the beginning of the mission various possible concept ideas for different aspects of the mission have been collected to weigh them against each other and consequently find a suitable and likely mission strategy.
The trade aspects investigated are summarized in the following list:

- Single-Launch vs. Multi-Launch
- Existing Launcher vs. New Launcher
- Identical Launchers vs. Different Launchers
- Identical Launch-Sites vs. Different Launch-Sites
- Multi-Transfer to Target vs. Single-Transfer to Target

The trades are summarized in the following according to their advantages and disadvantages.

8.3.1. Single-Launch vs. Multi-Launch Trade

Generally the mission components for CERMIT need to be carried into orbit by a launcher. To do this, a launch scheme with a single launch could be envisioned, carrying the whole payload at once – obviously reducing launch campaign complexity and costs, because e.g. personnel and launch preparations are only required once per mission. This would be a scenario similar to the Apollo-Programme, where all components, i.e. Lunar Lander and Command/Service Module, have been transported into LEO by one launcher (Saturn V).
An alternative would be to use several launches for one mission, which would reduce the amount of payload mass of each individual launches significantly (the maximum being about 100 tons for the ED stages).
In the following the advantages and disadvantages of the single-launch option over the multi-launch option are discussed (note that the disadvantages of one option are the advantages of the other and vice versa).

Compared to a multi-launch scenario, a single launch has less failure probability. Only one launch can fail, which in a total loss would of course mean also the total loss of the mission hardware. However as all components are mission relevant even the loss of one component means a failure of the whole mission, i.e. there is no advantage in launching several times.
If the payload is launched by a single vehicle, it is also easier to assemble, even if it is not completely assembled under the fairing. The fact that it is easier to reach identical orbits with a single launcher than with multiple ones, benefits the former option. As a multi-launch-campaign is complex and certain components, e.g. the ED stages, have a restricted amount of possible dwelling time in space, a single launch allows more flexibility towards the launch window and the overall mission. If the whole mission needs to be postponed, no hardware is already in orbit. For a multi-launch mission a possible failure scenario would be to have one

component in space, which could make a delay impossible or at least costly if the already launched component needs to be replaced (e.g. because it re-entries before the mission can be attempted once more).

Overall the complexity, operations and mission scenario is easier to accomplish with a single launcher. The same is true for the logistics and infrastructure.

Disadvantageous is the fact that there is a very high system complexity for a launcher able to carry, like in this case, 360 tons of payload, which increases the development uncertainty and risk. For the same reason it is costly and challenging to qualify such a complex launch vehicle for human crew transports.

Furthermore a multi-launch scenario allows more discrete distribution of work packages or launch responsibilities of whole launches, distributing responsibilities of a single launch vehicle might result in high coordination effort and thus negate positive effects of work sharing.

The biggest disadvantage of a single-launch solution however is the restriction in payload mass. It has been assessed by the study team that a single launcher cannot carry more payload than about 150 to 200 tons, rendering the single-launch option invalid for the current complete mission design (although less ΔV demanding missions could still be accomplished with this kind of launcher).

Table 8-1: Advantages and disadvantages of the single-launch option over the multi-launch option.

Advantages	Disadvantages
Lower failure probability	Higher development uncertainty (due to complexity)
Lower assembly effort of payload	Reduced opportunity for international cooperation
More flexibility (wrt launch window and mission)	More difficult to human rate due to complexity
Simpler mission design	Limit of maximum practical payload capacity (ca. 150 to 200 t)
Single infrastructure and logistics	

Table 8-1 summarizes the advantages and disadvantages from the single-launch point of view wrt the multi-launch option. During the study it has been decided that the single-launch's limit on the payload mass rules this option out and therefore the multi-launch option has been selected for further consideration.

8.3.2. Multi-Launch Trades: Existing vs. New Development

As there currently is no single-launch option capable of handling the large payload demands, only for the multi-launch option a trade needs to be conducted regarding the utilization of existing launchers or the development of a new one.

Using an existing launcher would reduce the failure probability of the launcher because it would be a system with significant flight heritage. The gain in safety when compared with a new launcher development is especially beneficial for the crew transport into orbit.
Problematic however is the fact that there is no heavy launcher in usage at the moment. The only comparable project currently in development is the Space Launch System of the United States. From existing launchers in Europe ARIANE 5 is the heaviest option, capable of carrying about 20 tons into a LEO. This would mean at least 18 launches would be required, to accomplish the mission (not taking into account payload mass increases due to smaller mission modules, e.g. need for more docking adapters). The trade is summarized in Table 8-2.

Table 8-2: Advantages and disadvantages of the multi-launch: existing launcher option over the multi-launch: new development option.

Advantages	Disadvantages
Lower failure probability	No heavy launcher available
Increased safety for astronauts	

Due to the fact that currently no heavy launcher exists in Europe that could carry out such a mission and therefore only unrealistically high launch numbers would enable it, the study team selected the option of developing a new launch vehicle.

8.3.3. Multi-Launch Trades: Identical vs. Variable New Launcher

Under assumption of a new development for a multi-launch scenario, it is important to decide if all launchers are identical or various launcher types should be taken into account and thus development.
A single new launch vehicle would reduce the costs of development and testing as the effort is reduced (although certain systems could probably be shared even if the complete launch system is different). As a disadvantage the launchers could not be adapted to their purpose and possible different launch-sites if only a single launcher type is used.
Favouring the reduced development costs for the study, the identical launcher option has been selected.

8.3.4. Launch-Site Trade: Identical vs. Variable

If a multi-launch scenario is applied, it would be one possibility to use several launch sites or a single one. The latter has an advantage in logistics, e.g. transport of components to the launch site is easier if there is only a single one, i.e. only one infrastructure and process is needed. Also the mission design is simplified as e.g. no different orbit characteristics have to be regarded as it would be the case for the usage of multiple launch-sites with different geographical latitudes.

One advantage of a single launch-site especially with regard to a single launcher type would also be the fact that there are fewer constraints to account for, e.g. regarding launch direction or timing of staging (to prevent debris dropping on settled areas).
A major disadvantage is that it is likely that the launch frequency would be reduced as a single site can only handle a certain amount of personnel and components at a time. The trade is summarized in Table 8-3.

Table 8-3: Advantages and disadvantages of the single launch-site option vs. the multi-launch site option.

Advantages	Disadvantages
Easier logistics (e.g. transport of components)	Lower launch frequency
Easier mission design (e.g. similar orbit inclination)	
Less constraints (only of one site) on individual launcher (wrt design and performance)	

No final decision has been made on the launch-site selection during CERMIT, but for the further course of the study the single launch-site option has been assumed due to the fact that the launcher designs that have been drafted during the study did not suggest very large launch numbers rendering this option impracticable.

8.3.5. Transfer Scenario Trade: Multi-Transfer vs. Single-Transfer

Due to the large mission payload masses that need to be transported towards the NEA target, it is worth to investigate the possibility to distribute the mission payload onto two or more transport vehicles, only one of them carrying the crew. This would reduce the maximum mass to be transported by a single spacecraft and therefore increase the feasibility of the mission as a whole, due to the relaxed constraints on the transfer stages.
However unlike in LEO where there is always the ability to abort the mission by the crew, this is not the case if assembly is postponed to later parts of the mission, e.g. at the asteroid. Generally a scenario where parts of the mission components (e.g. the asteroid departure stage) are sent ahead on their own, has an increased mission complexity. There are additional rendezvous maneuvers necessary, the launch campaign needs more effort and also the fact that two (or more) instead of one spacecraft need to be surveyed during the mission likely requires more personnel. Two spacecraft e.g. with two (even if identical) propulsion systems have an increased risk of failure.
Concluding no safety relevant equipment can be sent ahead, e.g. no propellant or system required for a safe return to Earth. The advantages and disadvantages as explained above are summarized in Table 8-2.
The fact that there is no bulky equipment on EXPLORER, which is not mission or safety-relevant, rules out the utilization of a multi-transfer scenario for CERMIT.

Table 8-4: Advantages and disadvantages of the multi-launch: existing launcher option over the multi-launch: new development option.

Advantages	Disadvantages
Reduce maximum mass to be transported by one spacecraft	No abort ability in case of assembly failure distant to Earth
Increase feasibility of mission (due to relaxed constraints and demands on transfer stage)	Complex mission design
	No safety relevant equipment can be sent ahead

8.3.6. Re-entry Scenario Trade

Regarding the return of the spacecraft to Earth and especially re-entry of the crew capsule, two possible scenarios are thinkable:

1) Direct re-entry (aerocapture)
2) Entry into parking orbit and subsequent re-entry

Entry into a parking orbit relaxes the requirements for the thermal protection system (TPS) of the re-entry capsule, due to a smaller re-entry velocity for a subsequent landing. One possible scenario could even be to have the spacecraft dock with a re-entry module only in LEO, to prevent carrying the heavy TPS all along through the mission, even though it is only needed at its end. In a similar scenario, the spacecraft could also dock with a LEO infrastructure (like the International Space Station) and re-enter Earth's atmosphere from there. However the maneuver for entering into an Earth orbit would require extra fuel, which also has to be carried along for the whole mission (which would only be reasonable if the TPS mass reduction would be larger) and add the risk of another engine ignition and a maneuver. So generally for this scenario there would be two major risks: engine failure and TPS failure.
If on the other hand, the spacecraft conducts a direct re-entry without previous maneuvers, the stress on the TPS is increased in comparison to the parking orbit variant. However the risk of an engine failure and a maneuver are removed, consequently reducing the overall risk for the human crew. A direct re-entry scenario has been applied for the Apollo-Programme.
As risk reduction is – despite the more severe demands on the TPS – gained by the direct re-entry option, it is selected for the mission as planned in the CERMIT study. A summary of this trade is given in Table 8-5.

Table 8-5: Advantages and disadvantages of the direct re-entry over the parking orbit variant.

Advantages	Disadvantages
Less risky (no maneuver necessary)	Heavy TPS required
No propellant required	No ability to dock with infrastructure/ module

8.3.7. Crew Transport into Orbit

For the crew transport into orbit several scenarios exist. First of all the crew needs not to be transported into orbit with the crew capsule, but could use a separate system, possibly with more heritage (e.g. a manned ATV variant). This way, the actual crew capsule could be transported into space very early during the mission preparation and thoroughly be tested and the crew needs only to be sent up once the launch is certain (that way also reducing the amount of consumables necessary, because only a short dwelling time in orbit would occur). On the other hand it would require another capsule and an additional launch of a spacecraft as well as a docking maneuver to transfer the crew to EXPLORER.

Therefore another option would be to transport the crew capsule (probably in combination with some or all other mission components) into space, along with the crew. This way launch numbers are reduced, however there are still the costs and effort for qualifying the new launcher for transporting a human crew.

A third variant would be to qualify an existing human rated launcher for transporting the crew capsule into orbit. For this option a docking maneuver for completing the assembly of EXPLORER is also necessary.

Figure 8-1: Proposed design for the Liberty launch vehicle [EADS Astrium].

In the terms of this study, it has been decided that qualifying a possible new heavy launcher for human spaceflight is costly and would result in large effort. Therefore the crew capsule along with the crew will be transported into space via a by then established and human rated launcher, either an ARIANE 5 derivate, a human rated Falcon 9, possibly the Liberty launch vehicle by EADS Astrium (s. Figure 8-1), which is a proposed launcher inheriting technology from both the US Space Transportation System (1st stage) and ARIANE 5 (upper stage) or any other suitable vehicle.

8.4. Baseline Design

During the course of the study, two general mission scenarios have been discussed, which only differ wrt the launch campaign, mainly the number of launches used. The remaining maneuvers and mission steps are similar for both approaches. As the actual mission scenario needs to be selected once the launcher is available, the two examples are explained below as baseline, as they depend on the actual launcher used.

The description also applies only to the design cases of the mission – as mentioned before, there are mission opportunities with relaxed requirements regarding the ΔV and therefore allowing several maneuvers to be executed by one stage. In the design case each stage (with the exception of Earth departure) is only used for a single maneuver.

8.4.1. Launch Campaign

Besides the requirement of transporting the mission components into orbit, the launch campaign is subject to two constraints. To limit the risk for the crew and the effort to sustain it, the crew is transported into orbit last. An alternative could be transport to an infrastructure in LEO (similar to ISS) and later transfer to EXPLORER, but currently there are no definite plans for such an infrastructure and it is therefore not included in the considerations.

The two before mentioned options are depicted in Figure 8-2 and Figure 8-3. For the four-launch variant, it is assumed that the crew is transported by an existing capsule into orbit, which docks to EXPLORER for crew transfer.

However as shown in the two-launch strategy, it is also possible to transport the crew along with the crew capsule into orbit (which is then lost, in case the mission needs to be aborted and the crew return to Earth). In any case the crew transport occurs separately and therefore adds another launch to the campaign besides the SIRIUS-launches, even though for both cases the crew capsule can be added to one of the other launches.

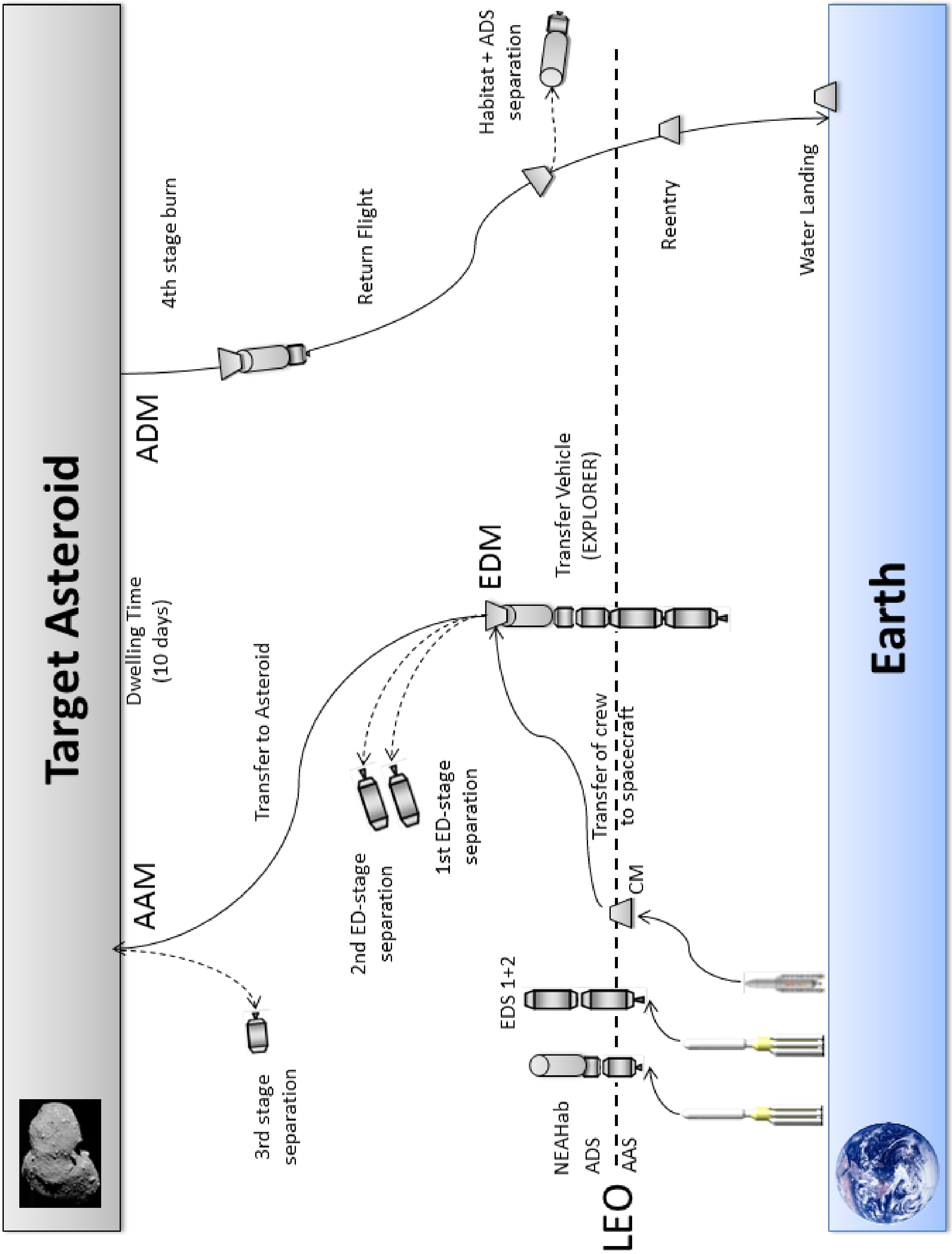

Figure 8-2: Possible mission scenario for a utilization of SIRIUS-H.

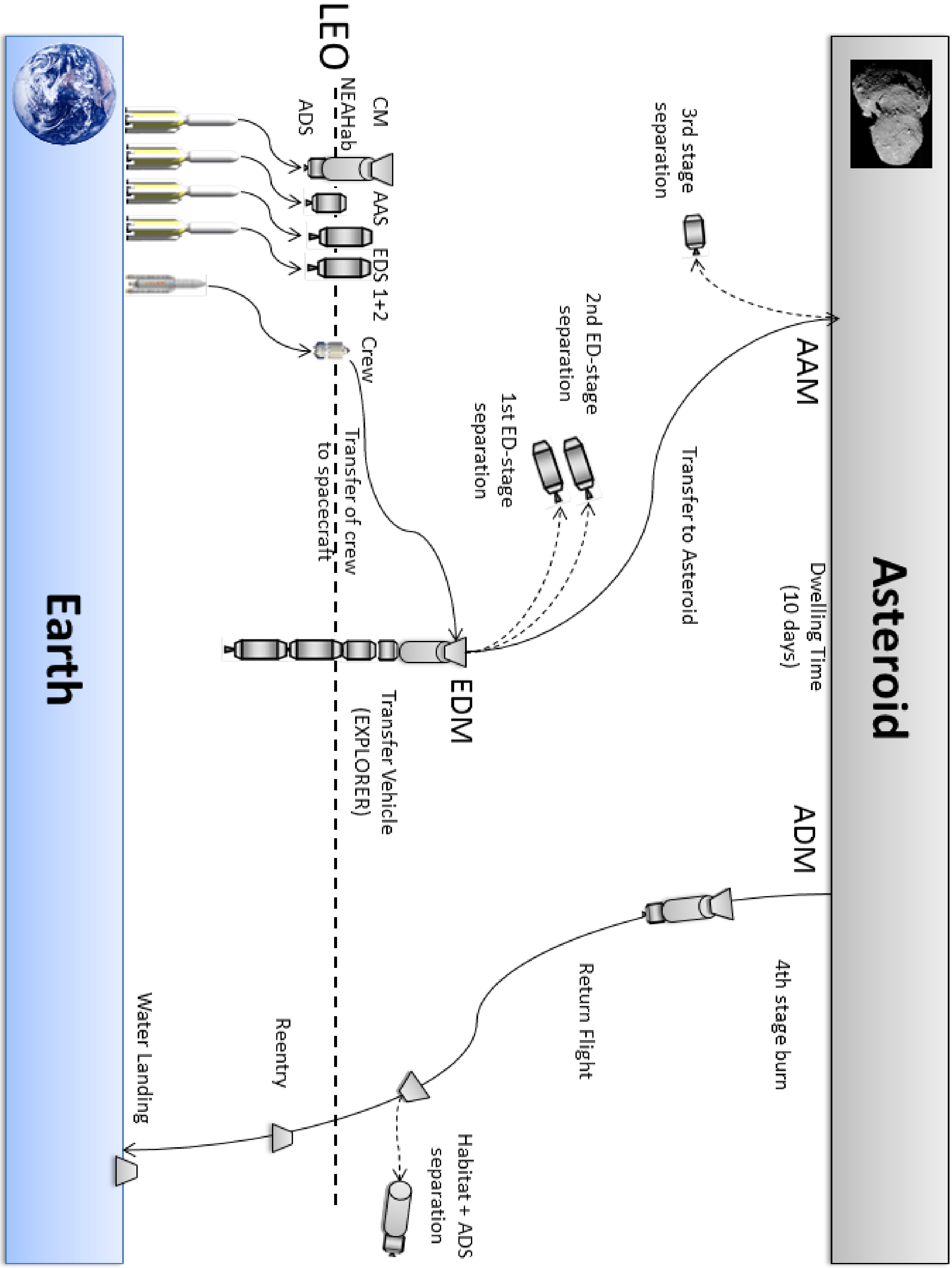

Figure 8-3: Possible mission scenario for a utilization of SIRIUS-L.

Due to the restrictions of cryogenic propellants, as used of the Earth Departures maneuvers, regarding storage, the two ED stages are launched last. As can be seen in Figure 8-2 and Figure 8-3, the launch of these two stages would occur either with a single launcher (with a

payload mass of 199,922 kg), in case of SIRIUS-H, or with two launches (with a mass of 99,911 kg each) of SIRIUS-L. These payload mass requirements are the drivers for the two launcher designs (s. Section 5), which can carry 200 resp. 100 tons into orbit.
As a consequence of the limited amount of storing capability, the launch of the ED stages can only occur 30 days prior to the opening of the launch window (s. Section 6.4.3). An early launch allows a relaxed schedule for docking maneuvers with previously launched components and testing of the two stages.
As can be seen in Figure 8-2 and Figure 8-3, before the ED stages, the remaining components can be launched. Due to the extensive storability of the non-cryogenic propellant, time is no essential factor for this part of the launch campaign.
In the four-launch option with SIRIUS-L therefore the AA stage can be separately launched from the remaining components (AD stage and crew modules). Also as an example in this version of the launch campaign, the command module is supposedly part of the first payload. This way all systems relevant for the crew safety (crew capsule, habitat and AD stage) can be extensively tested in orbit. Once their reliability is established, the next components of EXPLORER are launched.
Especially for the cases where not all stages are required for a successful mission, i.e. for a ΔV below the 7 km/s, a variable number of stages is transported into orbit, therefore various fairing need to be developed for SIRIUS (either version).
All launches and the assembly occur to resp. in a circular parking LEO of 300 km altitude. An assembly procedure has to be developed at a later time.

8.4.2. Transfer to Asteroid

Once EXPLORER is assembled the spacecraft can execute the departure maneuver from Earth, escaping for the first time Earth's gravity field with a crewed spacecraft. It will burn both ED stages for this purpose (however not in all cases they are fully tanked), the only case where two stages are required for one maneuver. Afterwards the transfer to the asteroid is conducted, which can include experiments, equipment preparation and generally work supporting the subsequent arrival at the asteroid. The flight time is variable as is described in Section 3.
For the purpose of the CERMIT study, a constant dwelling time of 10 days at the asteroid has been assumed for the mission. This will include extra vehicular activities for experiments on the asteroid surface.

8.4.3. Return to Earth and Re-entry

Return to Earth is initiated by the AD stage, bringing EXPLORER on its way home. For the duration of the flight, which is also variable, the AD stage will be docked at the spacecraft, because it houses the attitude control system and is needed to attain the correct attitude at Earth for beginning re-entry procedures.

Once in Earth proximity and at the right attitude, the re-entry capsule is separated from EXPLORER, leaving behind the habitat, the AD stage and the service part of the crew capsule. Considering comparable missions and current technology it is likely that the landing will occur on water. Likely positions for this are the Atlantic, possibly west coast of Europe or Africa, where European access is easy to realize.

8.5. Long Term Evolution

Comparable e.g. to Apollo it is advisable to plan an increasing complexity and difficulty of NEA missions to slowly gather experience with this kind of mission without risking the crew.

It is likely that the spacecraft components are tested and verified in Earth orbit first, before further steps are taken. Validation opportunities might also be circling of the Moon.

As not all asteroid targets require the total amount of ΔV for all launch windows, it is reasonable to plan missions in a sequence with increasing demands. For example a first mission could be conducted to 2000 SG_{344} (s. Section 3) using only one ED and the AD stages and 4 years later to 1999 CG_9 with the complete set of components.

9. Cost

Within the context of the CERMIT study, two lots of costs were deduced. Give the embryonic stage of the study itself, it is important to note that only the development costs have been considered. Manufacture and operations costs remain a point for further investigation to be conducted in the future, perhaps when more technical and mission details have crystallised.

The first group of development costs relates to the launcher system itself, including four different configurations which were considered. The second group of development costs addresses the transfer modules and their respective various configurations which were studied. Both are summarised below.

For the cost estimation, the parametric approach was assumed, and the model used was the [RD 8]. This model is a dedicated model for launcher systems, and as such splits up a launcher into its constituent stages, as well as engines and boosters. More information about TransCost model and the underlying parametric approach can be found in the SART document [RD 9].

9.1. Launcher System Development Costs

Four configurations were given serious consideration within the scope of the CERMIT study, being:

- EAP-H800-H32 (four Vulcain 3 engines, two Vinci engines)
- Zenit-H600-H32 (three Vulcain 3 engines, one Vinci engine)
- Zenit-H800-H32 (five Vulcain 3 engines, one Vinci engine)
- Zenit-H800-H32 (five Vulcain 3 engines, two Vinci engines)

Some assumptions had to be made in order to be able to perform the cost estimations for each of the new stages and engines. These assumptions are outlined below:

- EAP booster stage for Configuration 1 is identical to the Ariane 5 ECA EAP stage, therefore development costs for this within the context of CERMIT are negligible
- Ukrainian Zenith stage is already in existence and fully operational and therefore incurs zero development costs
- Vinci engine is also fully operational and therefore bears a development cost of zero.
- Development costs are only incurred for the following components:
    - Vulcain 3 engine
    - H600, H800 and H900 stages
    - H32 stage

- The TransCost factor for cost growth by deviation from optimum schedule (f6) is 1 (i.e. the schedule is perfectly adhered to)
- Calculation of the TransCost program organisation factor (f7) with three prime contractors for the project, namely one for the Vulcain 3 engine, one for the H32 stage, and one for the H600/H800/H900, in accordance with which configuration if being developed
- The fairing is part of the upper stage structure for costing purposes when applying the relevant TransCost CERs. Its mass was therefore combined with the total mass of the upper stage, instead of being calculated separately. This may have resulted in a higher cost, since the fairing is a simpler structure than that implied by the stage CER which has been applied
- TransCost table for the cost of one Work Year has been applied to convert effort into a monetary 'cost' value
- The development is conducted in Europe, and therefore the TransCost productivity factor (f8) for Europe has been applied on all calculations
- 100 test firings for the Vulcain 3 engine

Slightly different costs are observed for the Upper stage of seemingly identical Vinci stage with two Vinci engines (EAP-H800-H32 and Zenith-H900-H32). This is the result of data inputs which were obtained from calculations using different payloads.
The summary of the four different configurations and their development costs are given in Table 9-1 to Table 9-4.

Table 9-1: EAP-H800-H32 development costs based on the TransCost model.

EAP-H800-H32	Cost (M€ 2011 e.c.)	Cost with factors
H800 (4 x Vulcain)	15914.4	19178.4
H32 (2 x Vinci)	814.3	981.3
Vulcain 3	1743.1	2100.6
Total		**22260.3**

Table 9-2: Zenit-H600-H32 development costs based on the TransCost model.

Zenith-H600-H32	Cost (M€ 2011 e.c.)	Cost with factors
H600 (3 x Vulcain)	12439.7	14991.1
H32 (1 x Vinci)	819.3	987.3
Vulcain 3	1743.1	2100.6
Total		**18079.0**

Table 9-3: Zenit-H800-H32 development costs based on the TransCost model.

Zenith-H800-H32	Cost (M€ 2011 e.c.)	Cost with factors
H800 (5 x Vulcain)	14881.7	17933.9
H32 (1 x Vinci)	819.3	987.3
Vulcain 3	1743.1	2100.6
Total		**21021.8**

Table 9-4: Zenit-H900-H32 development costs based on the TransCost model.

Zenith-H900-H32	Cost (M€ 2011 e.c.)	Cost with factors
H900 (5 x Vulcain)	16005.9	19288.7
H32 (2 x Vinci)	813.4	980.2
Vulcain 3	1743.1	2100.6
Total		**22369.5**

9.2. Transfer Stages Development Costs

The development of three different transfer stages had to be costed within the scope of the CERMIT study, being:

- The Vinci engine used for the ED2 Stage is fully operational and therefore incurs no development cost
- Development costs are only incurred for the following components:
  - AD Stage
  - AA Stage
  - ED2 Stage
  - Aestus II Engine
- The TransCost factor for cost growth by deviation from optimum schedule (f6) is 1 (i.e. the schedule is perfectly adhered to)
- Calculation of the TransCost program organisation factor (f7) with three prime contractors for the project, namely one for the AD stage, one for the AA stage, and one for the ED2 stage
- TransCost table for the cost of one Work Year has been applied to convert effort into a monetary ‘cost’ value
- The development is conducted in Europe, and therefore the TransCost productivity factor (f8) for Europe has been applied on all calculations
- All calculations for the Aestus 2 engine were made from inputs and data obtained from a previous calculation of Aestus 2 within the context of a VENUS study.
- 30 test firings for the Aestus 2 engine

The summary of the four different configurations and their development costs are given in Table 9-5 to Table 9-7.

Table 9-5: AD stage development costs based on the TransCost model.

AD Stage	Cost (M€ 2011 e.c.)	Cost with factors
Stage	1540.6	1716.5
Aestus 2	272.9	304.0
Total		**2020.6**

Table 9-6: AA stage development costs based on the TransCost model.

AA Stage	Cost (M€ 2011 e.c.)	Cost with factors
Stage	2151.5	2397.2
Aestus 2	272.9	304.0
Total		**2701.2**

Table 9-7: ED stage development costs based on the TransCost model.

ED Stage	Cost (M€ 2011 e.c.)	Cost with factors
Stage	3245.4	3615.9
Aestus 2	272.9	304.0
Total		**3920.0**

The previous calculations show the total cost per stage only. However, the development cost for the Aestus 2 engine will be incurred only once for the transfer modules, which, combined, can be seen as constituent components within a single system. Therefore the overall total sum of development costs is more adequately grouped in Table 9-8.

Table 9-8: Complete costs for the transfer stage development.

Component	Cost (M€ 2011 e.c.)
AD Stage	1716.5
AA Stage	2397.2
ED2 Stage	3615.9
Aestus 2	304.0
TOTAL	**8033.7**

9.3. Ground Segment Cost

For handling of the launchers carrying the different parts of the CERMIT spacecraft (4 Transfer Stages, 1 habitat and 1 crew capsule) into orbit two dedicated ground segments are needed. One to launch the crew capsule and one to launch the four Transfer Stages and the habitat. In the following subchapters the cost estimation for the construction of the ground segments of the crew capsule and of the unmanned parts, i.e. Transfer Stages, habitat and crew capsule, is elaborated.

9.3.1. Ground Segment for Crew Launcher

It is foreseen to start the CERMIT Crew Module on top of a modified Ariane 5. So the existing ground segment infrastructure of the Ariane 5 has to be adjusted to the requirements for handling a crewed spacecraft. In this subchapter the cost estimation for such a ground segment infrastructure is presented.

During the cost estimation for the ground segment infrastructure of the CERMIT Crew Module several ‘global’ assumptions have been made:

- Costs are displayed in k€ and FY 2010.
- For the Project Office cost estimates on WP 1000-Level, 8% on Project Mgmt., 12% Systems Engineering and 4% on PA & Risk Mgmt. have been considered on the general sub level cost items.
- With respect to the Phase-A accuracy there will be three different margins set on each cost item (low: 10%; medium: 15%; high: 20%) to reflect the maturity level.
- No labour costs for the nominal operation procedures of Phase B-E1 have been estimated, nor are any other labour cost calculated that are associated with maintenance duties of the different new facilities at Kourou and Europe.
- Nominal astronaut recruitment costs & nominal astronaut training costs (up to ~5 years) are not included in the ground segment cost analysis.
- The typical thermal environment within the air-conditioned CSG facilities is kept at a temperature of around 23°C ± 2°C and a relative humidity of 55% ± 5%.
- The costs (especially for the ground infrastructure, e.g. buildings) over the different phases have been distributed by a Beta-Curve spread according to ground infrastructure costs derived from the NASA cost handbook. [RD 10]
- The building costs have been calculated according the German “Baukosteninformationszentrum Deutscher Architektenkammern GmbH – BKI”, which displays a collection of parametric building costs, derived from several former construction projects. [RD 11]
- In order to match the specific construction condition at Kourou a special Kourou construction factor of additional 25% has been applied.
- No mock-up costs are included in this cost estimation.
- No task/safety specific equipment and specific tooling within the equipment of the different facilities were estimated.
- The ground segment costs have been taken from [RD 12]. In this thesis a comparable cost analysis for a manned spacecraft (Advanced Reentry Module) was done.
- Dimensions of the ARV Cargo Version (length 11,23 m) > Dimensions of the CERMIT Crew Module (length 5,5 m)

- The costs of the ARV Cargo Version and the ARV Crew Version were resumed for the cost of the ground segment for the CERMIT Crew Module because there is no two step development approach (first Cargo than Crew Version) like in case of the ARV.
- In the cost estimation of the ground segment for the CERMIT Crew Module no Control Room Operation Costs for LCC & MCC, no Communication Network infrastructure and no Recovery Infrastructure were considered. (no procedure data known)

The present cost analysis is organized into three sub cost domains:

- Launch Site Infrastructure
- Mission Operation Infrastructure
- Training Infrastructure EAC

A cost analysis was done for each of these sub cost domains.

The total ground segment infrastructure cost for the CERMIT Crew Module derives to 156,987 k€ (FY2010). These costs can entirely be assigned to the non-recurring costs and none of them to the recurring cost for Phase C/D and Phase E1. As can be seen in Figure 9-1, main cost driver for the CERMIT Crew Module is set within the launch site infrastructure.

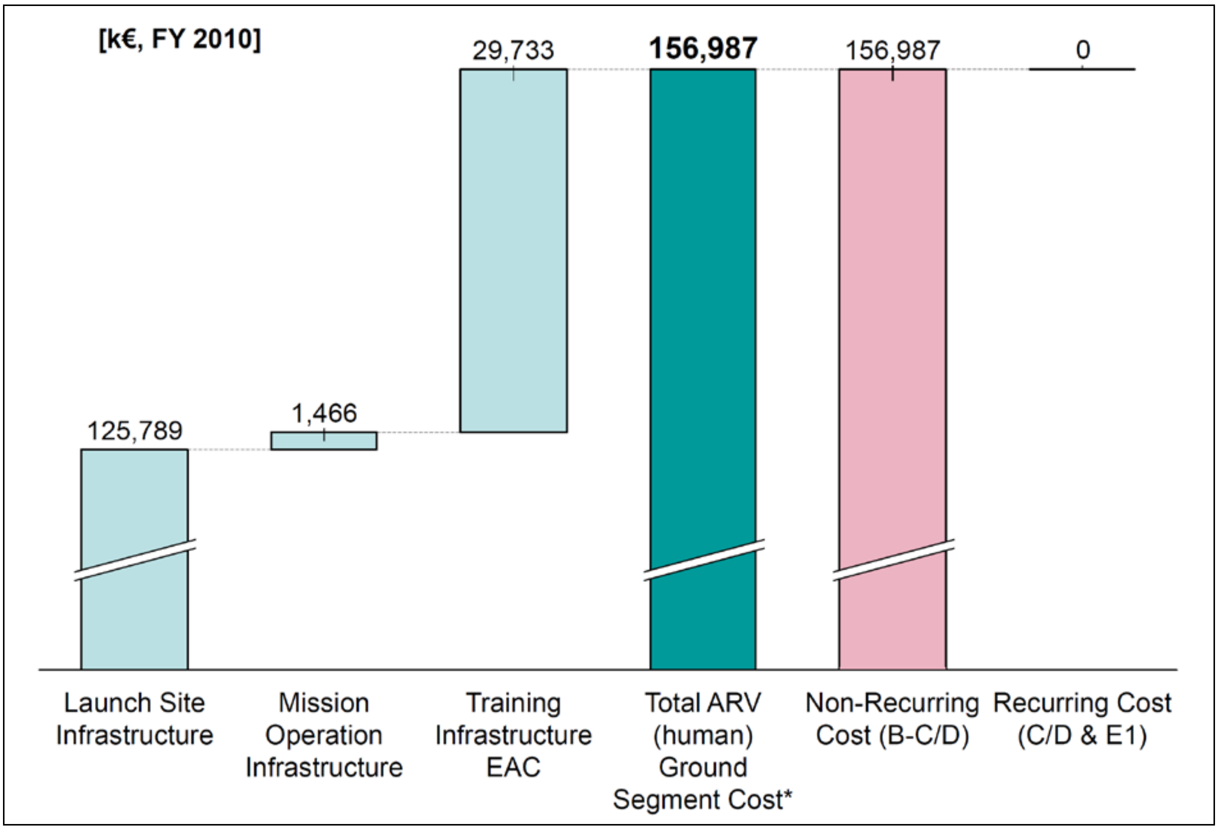

Figure 9-1: Cost summary of ground segment CERMIT Crew Module [*No labour costs included]

According to the life-cycle cost analysis the Phases B1 & B2 require 50,824 k€, which then increases to 106,164 k€ during Phase-C/D. During Phase E1 no costs occur as during operation all elements for the ground segment are completed. Compare Figure 9-2 for a detailed cost split.
It has to be stated that the labour cost are not included in the current ground infrastructure cost estimate. These costs would additionally increase the overall expenditures and need to be estimated in the next phase.

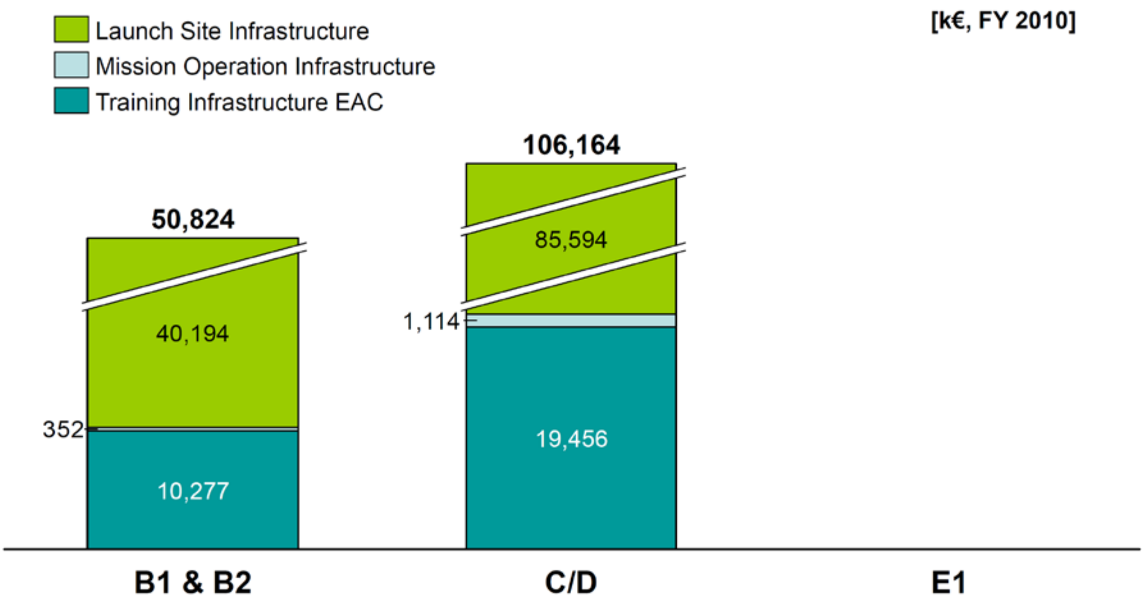

Figure 9-2: Ground segment life cycle cost of CERMIT Crew Module

Cost items like dedicated launch table, CES preparation building, astronaut preparation building and BAF parking hangar are some examples for the cost composition of the launch site infrastructure for the CERMIT Crew Module. The expenditures for the launch site infrastructure as well as several cost items in the Training Infrastructure EAC segment for the astronaut training or the Mission Operation Infrastructure (see Table 9-9) contribute to the costs for the CERMIT Crew Module ground segment infrastructure.

Table 9-9 displays a cost summary of the CERMIT Crew Module (See also [RD 12] for a detailed cost breakdown).

Table 9-9: Cost estimate overview of the CERMIT Crew Module ground segment costs

Cost Breakdown	Phase B2	Phase C/D	Phase E1	Total	Recurrent Phase C/D	Recurrent Phase E1
Ground Segment	**50.823,5**	**106.163,9**	**0,0**	**156.987,4**	**0,0**	**0,0**
Mission Operation Infrastructure	**351,8**	**1.114,2**	**0,0**	**1.466,0**	**0,0**	**0,0**
Project Office	68,1	215,6	0,0	283,7	0,0	0,0
Mission Control Centre (MCC) Infrastructure Modifications	283,7	898,5	0,0	1.182,3	0,0	0,0
Launch Site Infrastructure	**40.194,4**	**85.594,2**	**0,0**	**125.788,5**	**0,0**	**0,0**
Project Office	5.404,6	13.151,3	0,0	18.555,9	0,0	0,0
CES Preparation Building	590,7	867,1	0,0	1.457,8	0,0	0,0
A5 Launch Table	6.243,5	19.771,0	0,0	26.014,5	0,0	0,0
Final Assembly Building (BAF) Modifications	2.815,7	8.916,3	0,0	11.732,0	0,0	0,0
Launch Control Centre Modifications (LCC)	418,6	1.325,4	0,0	1.744,0	0,0	0,0
Launch Pad Modifications	9.372,6	14.169,6	0,0	23.542,2	0,0	0,0
Astronaut Preparation Facility	3.078,1	9.747,4	0,0	12.825,6	0,0	0,0
Payload Preparation Complex Modifications (EPCU)	12.270,6	17.646,0	0,0	29.916,6	0,0	0,0
Training Infrastructure EAC	**10.277,3**	**19.455,5**	**0,0**	**29.732,8**	**0,0**	**0,0**
Project Office	1.989,2	3.765,6	0,0	5.754,7	0,0	0,0
EAC Infrastructure Enhancements	8.288,1	15.689,9	0,0	23.978,1	0,0	0,0

9.3.2. Ground Segment for Cargo Launcher

In this subchapter the cost estimations for the ground segment infrastructure for two of the four possible CERMIT launcher configurations is shown. Because of the height of the two configurations - Zenit-H600-H32 and the Zenit-H900-H32 launcher - it is not possible to integrate and to start these launchers in/from the existing launch complexes of the Ariane 5 in Kourou. So a completely new launch complex has to be built on the site of the CSG. Below the cost estimation for the two launcher configurations Zenit-H600-H32 and Zenit-H900-H32 is presented.

Zenit-H600-H32

For the cost estimate of the ground segment infrastructure for the Zenit-H600-H32 launcher several assumptions have been made:

- Ariane 5 launch complex is not suitable for Zenit-H600-H32
- Completely new launch complex has to be built.
- For the cost estimation of the new launch complexes former building cost data of the present Ariane 5 ground segment infrastructure on the CSG were used. [RD 13]
- The cost data for the existing launch complex contains for example the whole building- and development costs for e.g. buildings, roads or rail tracks, the power supply and miscellaneous supply facilities, complete interior of the buildings e.g. MGSE etc.(see Table 9-12)
- No labour costs for the nominal operation procedures of Phase B-E1 has been estimated, nor are any other labour cost calculated that are associated with maintenance duties of the different new facilities at Kourou.
- Ground segment infrastructure costs for the new launch complexes have been estimated by escalating the former building costs to FY 2010 and multiplying them with adjustment factors depending on differences in volumes or number of launcher segments.
- Adjustment factors (Ratio of Zenit-H600-H32 to Ariane 5 ECA):
  - Launcher complete volume ratio
  - Booster volume ratio x booster amount ratio
  - Fairing volume ratio
  - First stage volume ratio x no. of engines
- To calculate the ratios the Ariane 5 ECA version has been chosen.
- Costs are displayed in M€ and FY 2010.
- Use of Fairing I → Launcher height: 79.13 m
- Amount of launches per CERMIT mission: 4

Table 9-10 and Table 9-11 show the configuration parameters of the Zenit-H600-H32 Launcher and the Ariane 5 ECA used in the cost estimation.

Table 9-10: Data of the Zenit-H600-H32 Launcher

	Name	Amount	Length [m]	Diameter [m]	Volume [m³]
Launcher	Zenit-H600-H32	-	79.13	17.2	18,386
Booster	Zenit	4	39.6	3.9	473
First Stage (Core Stage)	Vulcan 3	1 (3 engines)	31.8	10	2,498
Second Stage (Upper Stage)	Vinci	1	7.9	5.4	181
Fairing I	-	-	35	8	1,759

Table 9-11: Data of Ariane 5 ECA Launcher [RD 4].

	Name	Amount	Length [m]	Diameter [m]	Volume [m³]
Launcher	Ariane 5 ECA		56	11.5	5,817
Booster	P241	2	31.6	3.05	231
First Stage (Core Stage)	Vulcain 2	1 (1 engine)	30.5	5.4	699
Second Stage (Upper Stage)	HM-7B	1	4.7	5.4	108
Payload Fairing	-	-	17	5.4	389

The total ground segment infrastructure costs for the ground segment infrastructure of a new Zenit-H600-H32 Launch Complex (see Table 9-12) derives to 6,168 M€ w/o maturity margin (FY2010).

Table 9-12: Overview of the investment cost for the different ground segment infrastructure parts of a new Zenit-H600-H32 Launch Complex [RD 13].

Launch Facility		Cost (MAU, FY1996)	Cost (M€, FY2010)	Multiplier	Remarks	Resulting Cost (M€, FY2010)
ELA-3		**547 MAU**	**734 M€**	**-**	**-**	**2,278 M€**
	CDL3	53	71	1	-	71
	BIL	55	74	18,386/5,817	Launcher volume ratio	234
	BAF	78	105	18,386/5,817	Launcher volume ratio	332
	BPE-BSP	7	9	473/231 x 4/2	Booster volume ratio x amount ratio	37
	Tables	82	110	18,386/5,817	Launcher volume ratio	348
	Launch Zone	83	112	18,386/5,817	Launcher volume ratio	354
	Check-Out	123	165	1,795/389	Fairing I volume	746

	Facilities				ratio	
	Roads etc.	42	56	1	-	56
	Other	24	32	18,386/5,817	Launcher volume ratio	101
UPG		**199 MAU**	**267 M€**	**473/231 x 4/2**	**Booster volume ratio x amount ratio**	**1,094 M€**
BIP		**51 MAU**	**68 M€**	**473/231 x 4/2**	**Booster volume ratio x amount ratio**	**279 M€**
BEAP		**54 MAU**	**73 M€**	**473/231 x 4/2**	**Booster volume ratio x amount ratio**	**299 M€**
Tests and Validation		**285 MAU**	**383 M€**	**-**	**-**	**1,668 M€**
	BEAP Tests/valid.	33	44	473/231 x 4/2	Booster volume ratio x amount ratio	180
	ELA-3 Tests/valid.	188	253	18,386/5,817	Launcher volume ratio	800
	EPC tests	41	55	2,498/699 x 3/1	First Stage (Core Stage) volume ratio x no. of engines	590
	Propellant supplies	23	31	18,386/5,817	Launcher volume ratio	98
Joint Facilities		**31 MAU**	**42 M€**	**18,386/5,817**	**Launcher volume ratio**	**133 M€**
Propellant Plants		**45 MAU**	**61 M€**	**-**	**-**	**193 M€**
	LOX/LN2 plant	8	11	18,386/5,817	Launcher volume ratio	35
	LH2 plant	26	35	18,386/5,817	Launcher volume ratio	111
	Helium plant	11	15	18,386/5,817	Launcher volume ratio	47
Miscellane ous		**53 MAU**	**71 M€**	**18,386/5,817**	**Launcher volume ratio**	**224 M€**
Total Costs		**1,265 MAU**	**1,699 M€**		**-**	**6,168 M€**

As can be seen in Figure 9-3, the main cost driver for the Zenit-H600-H32 launch complex are the building costs for the modified replica of ELA-3 (37 % of the whole costs).

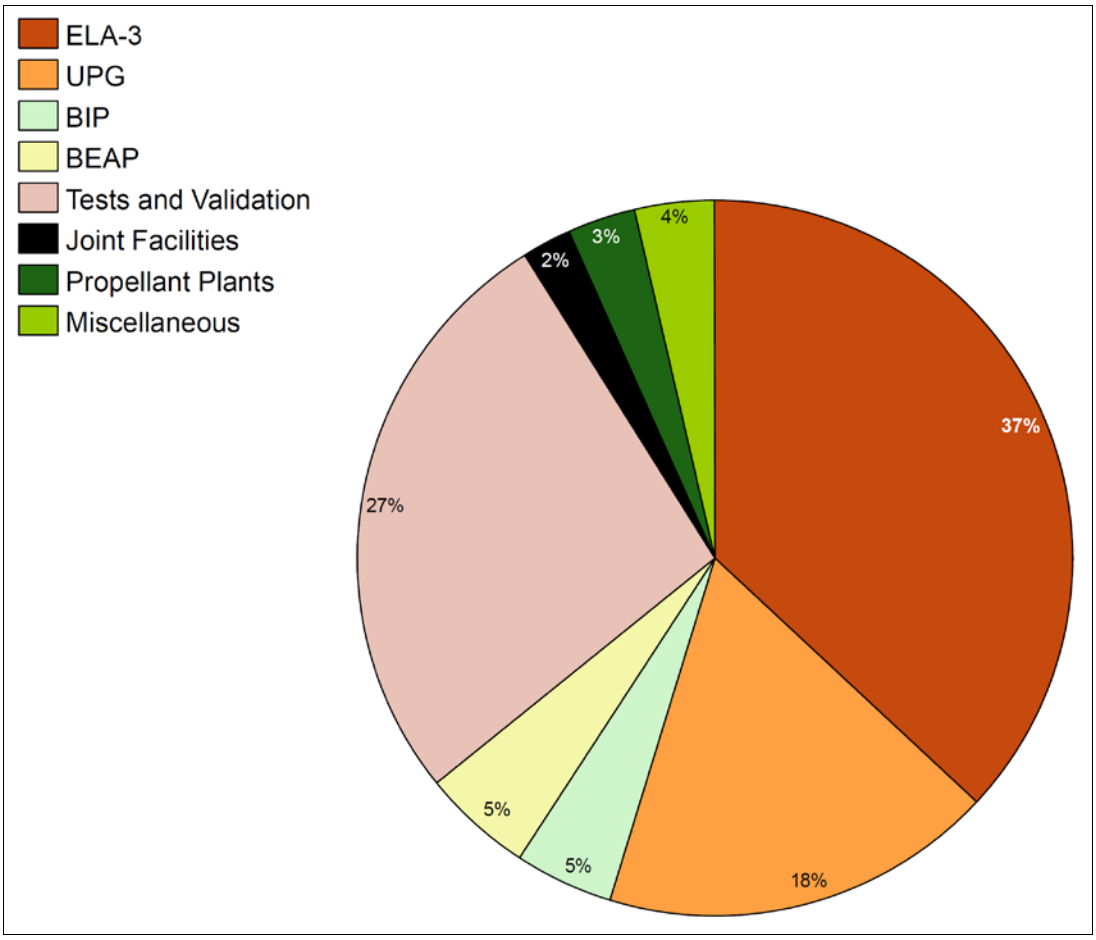

Figure 9-3: Distribution of the percentage for the different ground segment infrastructure parts of the Zenit-H600-H32 launch complex total building costs.

Zenit-H900-H32

For the cost estimate of the ground segment infrastructure for the Zenit-H900-H32 launcher several assumption were made in order to perform the estimate:

- Ariane 5 launch complex not suitable for Zenit-H900-H32
- Completely new launch complex has to be built.
- For the cost estimation of the new launch complexes former building cost data of the present Ariane 5 ground segment infrastructure on the CSG have been used. [RD 13]
- The cost data for the existing launch complex contains for example the whole building- and development costs for e.g. buildings, roads or rail tracks, the power supply and miscellaneous supply facilities, complete interior of the buildings e.g. MGSE etc. (see Table 9-12)
- No labour costs for the nominal operation procedures of Phase B-E1 have been estimated, nor any other labour cost calculated that are associated with maintenance duties of the different new facilities at Kourou.

- Ground segment infrastructure costs for the new launch complexes have been estimated by escalating the former building costs to FY 2010 and multiplying them with adjustment factors in dependency of differences in volumes or amount of the launcher segments.
- Adjustment factors (Ratio of Zenit-H900-H32 to Ariane 5 ECA):
  - Launcher complete volume ratio
  - Booster volume ratio x booster amount ratio
  - Fairing volume ratio
  - First stage volume ratio x no. of engines
- To calculate the ratios the Ariane 5 ECA version was chosen.
- Costs are displayed in M€ and FY 2010.
- Use of Fairing I and Fairing II→ cost estimation uses launcher height with Fairing II of 101.77 m to calculate the ground segment infrastructure costs of the new launch complexes because the buildings should be suitable for the Zenit-H900-H32 with Fairing I and Fairing II (Launcher with Fairing II is higher than with Fairing I).
- Amount of launches per CERMIT mission: 2
-

Table 9-13 and Table 9-14 show the configuration parameters of the Zenit-H600-H32 Launcher and the Ariane 5 ECA used in the cost estimation.

Table 9-13: Data of the Zenit-H900-H32 Launcher.

	Name	Amount	Length [m]	Diameter [m]	Volume [m³]
Launcher	Zenit-H900-H32	-	(92.54); 101.77	17.2	23,646
Booster	Zenit	8	39.6	3.9	473
First Stage (Core Stage)	Vulcan 3	1 (5 engines)	43.05	10	3,381
Second Stage (Upper Stage)	Vinci	1	7.9	5.4	181
Fairing I	-	-	35	8	1,759
Fairing II	-	-	44	8	2,212

Table 9-14: Data of Ariane 5 ECA Launcher [RD 4].

	Name	Amount	Length [m]	Diameter [m]	Volume [m³]
Launcher	Ariane 5 ECA	-	56	11.5	5,817
Booster	P241	2	31.6	3.05	231
First Stage (Core Stage)	Vulcain 2	1 (1 engine)	30.5	5.4	699
Second Stage (Upper Stage)	HM-7B	1	4.7	5.4	108
Payload Fairing	-	-	17	5.4	389

The total ground segment infrastructure costs for the ground segment infrastructure of a new Zenit-H600-H32 Launch Complex (see Table 9-15) derives to 9,796 M€ w/o maturity margin (FY2010).

Table 9-15: Overview of the investment cost for the different ground segment infrastructure of a new Zenit-H900-H32 Launch Complex [RD 13].

Launch Facility		Cost (MAU, FY1996)	Cost (M€, FY2010)	Multiplier	Remarks	Resulting Cost (M€, FY2010)
ELA-3		**547 MAU**	**734 M€**	**-**	**-**	**2,898 M€**
	CDL3	53	71	1	-	71
	BIL	55	74	23,646/5,817	Launcher volume ratio	301
	BAF	78	105	23,646/5,817	Launcher volume ratio	427
	BPE-BSP	7	9	473/231 x 8/2	Booster volume ratio x amount ratio	74
	Tables	82	110	23,646/5,817	Launcher volume ratio	447
	Launch Zone	83	112	23,646/5,817	Launcher volume ratio	455
	Check-Out Facilities	123	165	2,212/389	Fairing II volume ratio	937
	Roads etc.	42	56	1	-	56
	Other	24	32	23,646/5,817	Launcher volume ratio	130
UPG		**199 MAU**	**267 M€**	**473/231 x 8/2**	**Booster volume ratio x amount ratio**	**2,188 M€**
BIP		**51 MAU**	**68 M€**	**473/231 x 8/2**	**Booster volume ratio x amount ratio**	**557 M€**
BEAP		**54 MAU**	**73 M€**	**473/231 x 8/2**	**Booster volume ratio x amount ratio**	**598 M€**
Tests and Validation		**285 MAU**	**383 M€**	**-**	**-**	**2,846 M€**
	BEAP Tests/valid.	33	44	473/231 x 8/2	Booster volume ratio x amount ratio	361
	ELA-3 Tests/valid.	188	253	23,646/5,817	Launcher volume ratio	1029

	EPC tests	41	55	3,381/699 x 5/1	First Stage (Core Stage) volume ratio x no. of engines	1331
	Propellant supplies	23	31	23,646/5,817	Launcher volume ratio	126
Joint Facilities		**31 MAU**	**42 M€**	23,646/5,817	**Launcher volume ratio**	**171 M€**
Propellant Plants		**45 MAU**	**61 M€**	**-**	**-**	**248 M€**
	LOX/LN2 plant	8	11	23,646/5,817	Launcher volume ratio	45
	LH2 plant	26	35	23,646/5,817	Launcher volume ratio	142
	Helium plant	11	15	23,646/5,817	Launcher volume ratio	61
Miscellaneous		**53 MAU**	**71 M€**	23,646/5,817	**Launcher volume ratio**	**289 M€**
Total Costs		**1,265 MAU**	**1,699 M€**	**-**	**-**	**9,796 M€**

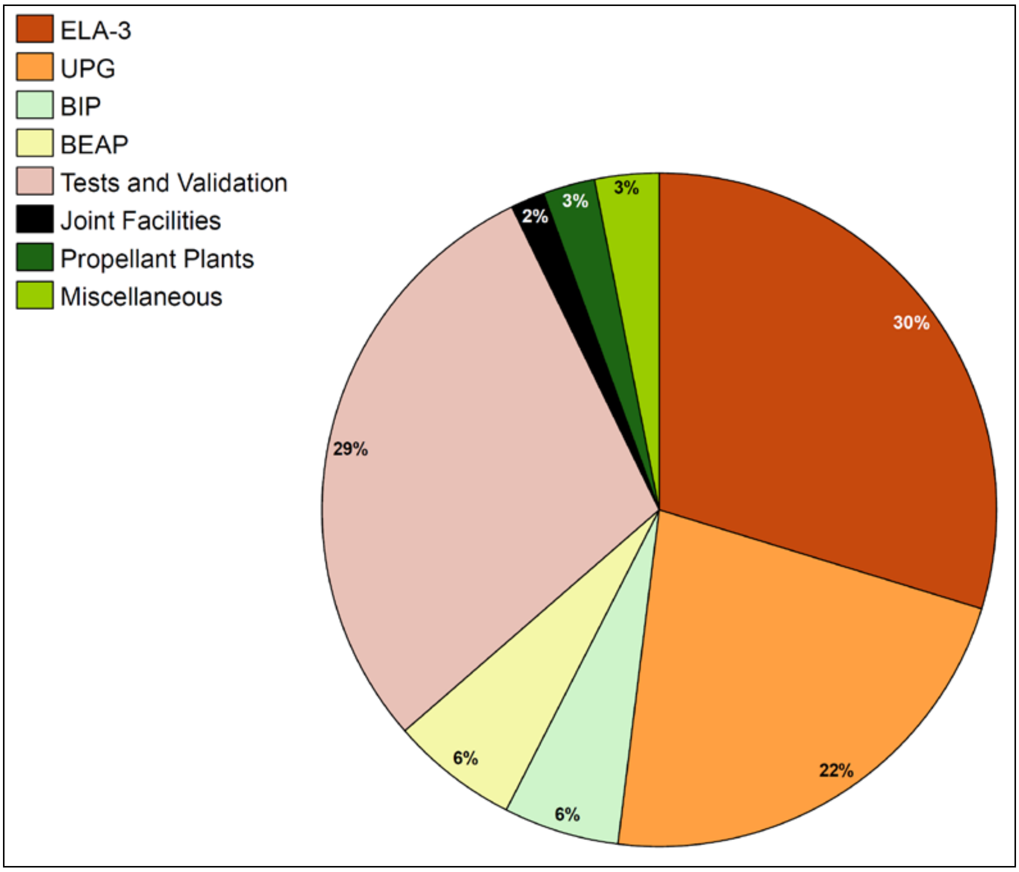

Figure 9-4: Distribution of the percentage for the different ground segment infrastructure parts of the Zenit-H900-H32 launch complex total building costs.

As can be seen in Figure 9-4, the main cost driver for the Zenit-H900-H32 launch complex are the building costs for the modified replica of ELA-3 (30 % of the whole costs) followed by the building costs for the modified replica of the Tests and Validation facilities (29 % of the whole costs).

9.4. Summary

The Launcher Stage development costs are calculated to be in the range of M€18.1 and M€22.4 depending on which configuration is selected. Table 9-16 concisely summarises the total costs per configuration and including the transfer stages.

Table 9-16: Summary of development costs in B€ and for 2011 economic conditions.

Launcher Configuration	Launcher Components	Launcher Dev. Costs	Transfer Stage Dev. Costs	Total Dev. Costs
EAP-H800-H32	H800, H32, Vulcain 3	22.3	8.03	**30.33**
Zenit-H600-H32	H600, H32, Vulcain 3	18.1	8.03	**26.13**
Zenit-H800-H32	H800, H32, Vulcain 3	21.0	8.03	**29.03**
Zenit-H900-H32	H900, H32, Vulcain 3	22.4	8.03	**30.43**

As more system and subsystem details become clearer, the cost estimates should be reassessed, to reflect any new information which becomes available.

Table 9-17 summarises the total costs of the ground segment infrastructure building costs for the different CERMIT launchers. Depending on the launch philosophy (2 or 4 cargo launches) the costs of the Zenit-H600-H32 or the Zenit-H900-H32 launcher are applicable.

Table 9-17: Summary of the ground segment infrastructure building costs for the different CERMIT launchers.

Launcher Configuration	Mode	Total costs w/o. margin [M€; FY10]	Margin	Total costs w. margin [M€; FY10]
Zenit-H600-H32	new construction	6,168	15.00 %	7,093
Zenit-H900-H32	new construction	9,796	15.00 %	11,265
CERMIT Crew Module Launcher	modification	140	12.42 %	157

In contrast to subchapter 9.3.2 a cost margin of 15 % with respect to the Phase-A accuracy to reflect a medium maturity level is implemented for the Zenit launcher costs (see Table 9-17).

As more system and subsystem details become clearer, the cost estimates should be reassessed, to reflect any new information which becomes available.

10. Summary and Outlook

This report describes the CERMIT study that investigated the possible layout of a European led crewed mission towards a Near Earth Asteroid and of its necessary components, i.e. the launch vehicle and the transfer stages for the flight towards the asteroid.

10.1. Baseline Design

For the baseline design of the mission, the mission components, completely assembled they are called EXPLORER, have to achieve maneuvers with a total ΔV of 7 km/s, to include a total of 6 possible targets in a timeframe from 2020 to 2040.
Designed to the respective worst cases for the various maneuvers, the following stage designs have been drafted during CERMIT:

- Asteroid departure stage: 36,358 kg
- Asteroid arrival stage: 84,256 kg
- Earth departure stage: 99,911 kg (needed twice for complete maneuver)

Furthermore, based on earlier calculations [RD 1], the crew relevant components are:

- Crew capsule (consisting of service module and re-entry capsule): 9,515 kg,
- Habitat module: 27,890 kg.

Together these components can accommodate a crew of 4 astronauts for a mission duration of 180 days and additional 30 days launch window.
Two launch scenarios are envisioned – one with a heavy lift launcher able to carry 100 tons into orbit (labeled SIRIUS-L) launching four times, another one with two launches of a launch vehicle with 200 tons capacity (labeled SIRIUS-H). The former version would be less demanding technology wise but have increased complexity and logistics for a very demanding launch campaign. The latter would require a launcher beyond the capabilities of any previous rocket, but have a relaxed launch campaign.
In any case the crew would be transported into orbit by a by then existing and validated launcher, to safe the effort and costs for qualifying the new, large and complex launcher for human transport.

10.1.1. Technology Development Requirements

For the design of this mission scenario it has been attempted to reuse as much as possible existing technologies and derive from these only where absolutely necessary. Development needs are mainly present in the propulsion area, where two new engine types are necessary, namely Vulcain 3 and Aestus 2, derived from existing ones.

Overall Europe has no experience with launcher types of this kind and capacity, increasing the system complexity significantly.

10.1.2. Costs

For a cost analysis the development costs have been investigated for CERMIT. Based on the fact that some technology can be reused, it has been calculated that the development of the stages would costs about 8 billion Euros, the launcher – depending on the exact configuration - between 18 and 22 billion. Further investments would be necessary for production of the components, and also creating the infrastructure for this and the ground segment, the latter has been accounted for by ca. 10 billion Euros.

10.2. Open Issues

The goal of the CERMIT study has been to formulate a possible “how to” of conducting a crewed mission (or rather a series of these, to justify the effort) towards a Near Earth Asteroid. Due to the time restrictions this did not include optimization of the various components, e.g. the launcher stages.
Furthermore the trades as presented in the earlier chapters are to a large extent only based on rough calculations or experience of the domain engineers. It is advisable to actually conduct thorough calculations to base these trades on hard facts.
Considerations still open are also the infrastructure and ground segment in general which is necessary to support such a mission.
Finally the development and whole programme behind this kind of missions needs to be thoroughly planned and a European strategy to set these plans into motion has to be established.

10.3. Benefits for Human Spaceflight Missions

Besides the mere scientific gains that can result from a human mission towards a Near Earth Asteroid and which are not part of this study, there are technological reasons for this kind of endeavor. Also social and cultural reasons should be taken into consideration, generally the broadening of humanity’s horizon, although this is also not part of this report. However in the following the benefits for even more challenging goals of human spaceflight are addressed shortly.

10.3.1. Comparison to Moon

Compared to a human return to Moon, a NEA mission is quite different. Considering the fact that Moon is a celestial body with a sensible gravity and a NEA is not, the whole environment for such a mission is very different, also the handling of equipment and generally conduct of astronauts.

For a return to the Moon benefits could be to experience long-term missions without immediate access to Earth (in difference to LEO missions, where a return to Earth is a simple matter). Also if more enduring missions to Moon are planned, life support technology developed for a CERMIT-kind of mission would be applicable to lunar missions as well.

Also the heavy lift launcher vehicles can be re-used to transport infrastructure for Moon into orbit.

10.3.2. Comparison to Mars

A NEA mission would provide experience with spaceflight beyond the terrestrial reach of gravity. While the Moon is a fairly safe target and can be reached on trajectories that include an automatic return to Earth, reaching a NEA is a different matter and in any case means leaving Earth's field of gravity behind – similarly to a Mars mission.

In terms of sustainability this could increase humanity's experience and allow identification of mission critical aspects for these kinds of human exploration missions. Also if missions to Mars include investigation of its Moons Deimos and Phobos, which are likely former main-belt asteroids, a NEA mission can help prepare for this.

10.4. Conclusion

The CERMIT study has proposed a mission architecture that allows a crewed spacecraft to reach a Near Earth Asteroid. While it is apparent that the demands and therefore technological challenges are significant, the fact that a majority of technologies would be further developments of existing ones, is encouraging. During the study no issue arose that would prevent such a mission altogether besides the willingness to undertake the development and subsequently the mission itself. While the occurrence of this willingness would results in high financial effort, this effort would generally benefit the European high technology sector and Europe as a whole by establishing it at the demanding frontier of human achievements.

11. Acronyms

Domain	Abbreviation	Comments
General		
	AA	Asteroid Arrival
	AAM	Asteroid Arrival Manoeuver
	AD	Asteroid Departure
	ADM	Asteroid Departure Manoeuver
	BOL	Begin of Life
	CAD	Computer Aided Drawing
	CEF	Concurrent Engineering Facility
	CERMIT	Crewed European Mission Trail
	DLR	Deutsches Zentrum für Luft- und Raumfahrt
	ED	Earth Departure
	EDM	Earth Departure Manoeuver
	ESA	European Space Agency
	ESTEC	European Space Research and Technology Centre
	EVA	Extra Vehicular Activity
	EXPLORER	European Extensive Personnel Laboratory for Remote Research
	ISRU	In-Situ Resource Utilization
	ISS	International Space Station
	K/O	Kick-Off
	LEO	Low Earth Orbit
	MAD	Mission Architecture Definition
	MPCV	Multi-Purpose Crew Vehicle
	NEA	Near Earth Asteroid
	NEC	Near Earth Comet
	NEO	Near Earth Object
	S/S	Subsystem
	STK	Satellite Tool Kit
	TRL	Technology Readiness Level
Mission	**Mission Analysis**	
	LW	Launch Window
	OCC	Orbital Condition Code
	PHA	Potentially Hazardous Asteroid
	TOF	Time of Flight
Crew Module		
	ARD	Advanced Reentry Demonstrator

	ATV	Automated Transfer Vehicle
	HV	Habitable Volume
	IDSS	International Docking System Standard
	LSSV	Life-Support System Volume
	NPV	Non-Pressurized Volume
	PV	Pressurized Volume
	TPS	Thermal Protection System
Launcher		
	CAC	Calculation for Aerodynamics Coefficients
	CUSP	Coronal Ultraviolet Spectropolarimeter
	EAP	Étages d'Accélération à Poudre
	GLOW	Gross Lift-off Weight
	LFBB	Liquid Flyback Booster
	PMP	Propellant Management Program
	RTS	Raumtransport System
	SART	System Analyse Raumtransport
	STSM	Space Transport and System Mass
	TOSCA_TS	Trajectory optimization and Simulation for Conventional and Advanced Launchers
Transfer Stages		
	AAS	Asteroid Arrival Stage
	ADS	Asteroid Departure Stage
	EDS	Earth Departure Stage
	MLI	Multi-Layer Insulation
	MMH	Monomethylhydrazine
	MON	Mixed Oxides of Nitrogen
Cost		
	ARV	Advanced Re-entry Vehicle
	BAF	Final Assembly Building
	CES	Crew Escape System
	CSG	Centre Spatial Guyanais (Spacecenter Guyana)
	FY	Fiscal Year
	MGSE	Mechanical Ground Support Equipment

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