DEIMOS SAMPLE RETURN TECHNOLOGY REFERENCE STUDY

Deimos Sample Return Technology Reference Study

-Executive Summary-

Science Payload and Advanced Concepts Office
Planetary Exploration Studies Section (SCI-AP)

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Reference	SCI-A/2006/010/DSR
Issue	2
Revision	1
date of issue	28/08/2006

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DISTRIBUTION

Name	Organisation

CHANGE LOG

reason for change	issue	revision	date
Updates based on comments from Peter Falkner (SCI-AP)	2	0	17/01/06
Minor editing changes	2	1	28/02/06

Deimos Sample Return TRS Summary

Scientific objectives:		
Primary Objectives	• Return 1 kg of Deimos regolith to Earth	
Stages	**Earth Return Vehicle (ERV)**	**Propulsion Stage**
Components	• Earth Return Capsule (ERC)	• Pneumatic Sampling System
	• ERC Sample Canister	• GNC Instruments and Navigation Control
	• Sample Transfer System	• Landing Pads
	• Communications, Power (Battery and Solar Array), AOCS, Data Handling, 2nd Propulsion stage, Thermal Control	• 1st Propulsion stage, Power, Thermal Control
Transfer:		
	• Soyuz Fregat 2-1b launch from Kourou into 200 x 25 000 km orbit	
	• 0.5 rev. (or 1.5 rev.) transfer to Mars - Transfer duration ~ 250 days (~700 days for 1.5 rev. case)	
	• After Mars Orbit Insertion (500 x 100 000 km) the S/C will co-orbit with Deimos (20 069 km circular)	
	• Deimos Observation Orbit (DOO) (relative elliptical motion about Deimos)	
	• Touch-and-go sampling sequences (3 nominal)	
	• Return to 500 x 100 000 km Mars orbit	
	• 0.5 rev. (or 1.5 rev.) transfer back to Earth – Transfer duration ~250 days (~800 days)	
Launch Opportunities	**Earth Departure**	**Earth Return**
	May 2011	August 2014
	May 2013	October 2016
	January 2016*	October 2018*
	May 2018	May 2021
	July 2020	August 2023
Mission Duration		
	• ~ 3 years	
Spacecraft details:	**ERV**	**Propulsion Stage**
Stabilisation	3-axis	N/a
Mass	Mass figures include 5-20% component margin (depending on maturity) and 20% system margin.	
Dry (kg)	**343.1**	**400.3**
Total Dry Mass (kg)	**743.4**	
Margin w.r.t. launcher (kg)	~22-33% (* ~6 % for 2016 launch case)	
Power (W)	143 (max. required)	
Data rate (Mbps/day)	360 (max. during DOO)	
TM band	X	
Antenna	HGA, LGA (x2)	
Data storage (Gbit)	64	
P/L power (W)	17	
P/L data rate (kbps)	7	
ROM cost (MEUR)	TBD	
Challenges:		
	• Development of passive Earth Return Capsule for hyperbolic entry (~12 km/s) • Highly Autonomous GNC system for Deimos proximity operations • Efficient sampling mechanism for a touch-and-go on a low gravity body in a vacuum environment (due to uncertainties must be able to cope with for multiple surface compositions) • Planetary protection: sample contamination should be prevented, no back contamination protection required • Solar power generation is a driving factor as the solar array must be fixed, limiting available size	

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1 INTRODUCTION

Determining the origin and composition of asteroids is a key step in understanding the nature of the solar system. Believed to be a captured asteroid, Deimos, Mars' moon, is therefore of dual scientific interest. The upper regolith of the moon contains Martian material accreted during the late heavy bombardment period. Retrieving a sample from Deimos would contain both asteroidal and Martian material. The perceived scientific interest in Deimos, and for small body sample return missions, are the key reasons that Deimos Sample Return (DSR) was chosen as one of ESA's Technology Reference Studies.

1.1 Technology Reference Studies

Technology Reference Studies (TRS) are a technology development tool introduced by ESA's Science Payload and Advanced Concepts Office, whose purpose is to provide a focus for the development of strategically important technologies that are of likely future relevance for scientific missions. This is accomplished through the study of several technologically demanding and scientifically interesting missions, which are currently not part of the ESA science programme.

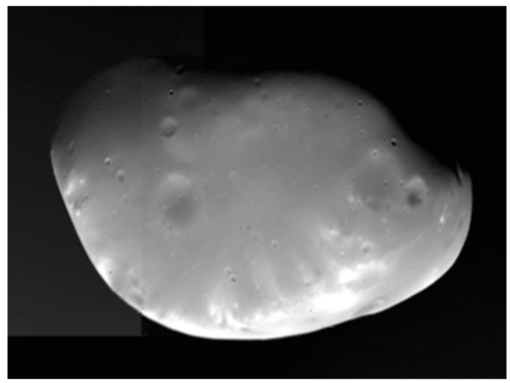

Figure 1.1-1: Deimos

The goal of the DSR TRS is to study the feasibility of and the technologies required to collect a scientifically significant sample of regolith from Deimos' surface and return it to Earth. The DSR mission profile consists of a single spacecraft, launched on a Soyuz-Fregat 2B. After transferring to the Martian system, the spacecraft will enter into a co-orbit with Deimos where it will undertake remote sensing observations and ultimately perform a series of sampling manoeuvres. Upon completion of sampling the spacecraft will return to Earth, where the sample canister, in an Earth Return Capsule, will perform a direct Earth entry.

1.2 Deimos

Deimos is one of two moons that are in orbit around Mars. The origins of Deimos and Phobos are unknown, although it is believed that they are asteroids that were captured into orbit about the planet.

Asteroids are attractive targets for sample return. They are believed to be similar to fossils, retaining some records of the formation of the planets. Deimos is smaller than Mars' other moon, Phobos, with an acceleration due to gravity less than 0.1 % that of Earth. It is also less irregular in shape than Phobos and has a smoother appearance due to partial filling of some of its craters. Although Phobos is also of scientific interest, these factors, along with Deimos' larger orbit, make it a more attractive target for such a mission.

Deimos is classified as a D-type class asteroid. D-type asteroids have low albedos and a generally featureless spectrum. Their spectrums have high values in the infrared region and albedos ranging from 0.04 to 0.07. Deimos has the highest albedo of any D-type asteroid at 60 % higher than the average. The cause of this is believed to be the presence of Martian ejecta on the asteroid surface.

Deimos Properties	
Mass	2.24 x 10^{15} kg
Dimensions	15.0 x 12.2 x 10.4 km
Mean Diameter	12.6 km
Density	2.2 g/cm^3
Surface Gravity	0.0039 m/s^2
Albedo	0.07
Regolith Mean Depth	10 m
Rotational Period	Synchronous
Distance to Mars Surface	20 069 km
Orbit Eccentricity	0.0005
Orbit Inclination	1.79 deg
Orbital Period	30.1 hrs
Orbital Radius	23 460 km

Table 1.2-1: Deimos Physical Properties

2 Scientific Objectives

The main objective of the DSR TRS is to examine the feasibility of returning a meaningful sample from the Deimos surface to Earth. According to expectations a 1 kg sample will contain about 100 g of Martian dust, which is expected to be the minimum required for the desired research [1]. A sample of this size will allow both complete coverage of Deimos and a clear view of several Martian ejecta originating from different episodes and different places. Therefore, the goal of DSR is set to return a 1 kg sample of material.

The sample should consist of regolith material from the surface. Optimally this should also include several small pebbles. In addition, the sample should not be composed entirely of 'surface dust' and should have some subsurface material, providing a good mix of regolith. Due to the homogeneous nature of the surface, the composition of the sample does not depend on sampling location, however newly formed craters should be avoided.

3 Sampling Strategy

There are various methods that can be used to retrieve a sample from a small solar system body such as Deimos. In a fly-by scenario the spacecraft would collect the sample without making direct contact with the surface of the asteroid. This type of sampling could be accomplished through creating a debris cloud and then directing the spacecraft though the cloud to collect the sample. However, this method presents a considerable risk to the spacecraft and was therefore discarded.

The other way to collect a sample is directly from the surface. This can be done a variety of ways, remotely from a hovering position above the surface, or by landing on the surface, either for a sustained sampling operation or temporarily. The main disadvantage with direct collection, when compared to a fly-by approach, is that the spacecraft must be safely manoeuvred down to the asteroid's surface. Therefore, the approach speed and transverse velocities must be strictly controlled.

3.1 Short Term Landing or "Touch and Go"

For the short term landing approach, the spacecraft lands upon the surface and then remains there for only a short time period. Due to the abbreviated time on the surface an anchoring system, required for a longer term landing, would not be required. There are several collection mechanisms that could be used with such an approach, including a gas injection collector, a Penetrator with funnel collector (as used on the Japanese Sample Return Mission, Muses-C [1]) and a rotary broom. The main concern with the different options is the amount of sample that can be collected in a single operation and the method of packaging and transferring the sample to the Earth Return Capsule (ERC).

For all of the mechanisms examined it would be extremely difficult to collect a 1 kg (estimated 600 cm^3 volume) sample in a single collection operation, therefore multiple operations will be required. This will likely increase the scientific return by providing samples from multiple sites. However, each operation requires extensive GNC planning to select and coordinate the approach to the site and each approach significantly increases the overall risk to the spacecraft. Therefore, the number of operations required should be minimised in the design, with a target of 2-3 and no more than 5.

From preliminary investigations of potential collection mechanisms it should be feasible to collect a sample of an adequate size so that no more than 5 operations are required. Further optimisation could also reduce this number. Therefore, with the capability to obtain the required sample and without the added complexity of requiring anchoring to the surface the short term landing approach is very attractive for DSR.

3.2 Remote Sampling

To accomplish a remote sampling the spacecraft would approach the surface without landing. A collection mechanism would be extended downwards and would collect the sample without the need for the main spacecraft body to land on the surface. Depending on the time required for collection the spacecraft might have to hover at a specific distance above the surface for a certain period of time. This could be accomplished using the spacecraft thrusters and the maximum permitted sampling time would be determined by the required propellant mass.

[1] JAXA, Institute of Space and Astronomical Science, Missions, Hayabusa, http://www.isas.jaxa.jp/e/enterp/missions/hayabusa/scenario.shtml

There are two main challenges to this approach, the connector between the collector and the spacecraft and the ability to collect the required size of sample. A flexible system, such as an inflatable rod, used as a connector, would not be compatible with a collector that would be able to collect more than a few grams of sample. While potentially interesting as a secondary sampling tool, this type of system would require numerous operations before collecting 1 kg of material. An increase in the number of operations significantly increases the risk to the spacecraft and this approach becomes impractical.

A rigid connector, such as an extendable boom or arm, would be extremely vulnerable to transverse velocities. In order to collect an adequate volume, and to meet scientific criteria, the collector would need to penetrate the surface. Therefore, if even a small transverse velocity remains during collection the connector, and potentially the spacecraft, could be adversely affected. It would also be difficult for the collector to produce the necessary force to significantly penetrate the surface and collect a sizeable sample. Together, the risk to the spacecraft and the remaining difficulty in collecting an adequate sample make this approach unattractive as well.

Another possible approach would be to have the spacecraft fire a Penetrator into the surface from its hovering position. The firing mechanism could be adjustable to allow for uncertainties in the surface composition, starting with a smaller force and increasing it on subsequent operations if the penetration depth was inadequate. The University of Arizona is currently developing this kind of device to be used for core sampling of a comet. Their device is oversized for DSR purposes as it was designed for ice penetration, but modifications could potentially reduce the mass and size. Currently the Penetrator has a mass of 20 kg and collects a 200 cm^3 core sample from solid ice. A similar device for DSR could collect the entire required sample in only 3 operations.

For DSR, however, there is a challenge in either designing the Penetrator so that it can be reusable, or to reduce the mass to allow for a dedicated Penetrator for each operation. If it is required that the Penetrator be reused, it would be extremely difficult to ensure that it could be safely extracted from the regolith with its undetermined properties.

The extraction itself also creates several challenges. A cable of some kind connecting the Penetrator to the spacecraft would be required. If the penetration of the device fails it could rebound from the surface and, due to the cable, impact the spacecraft. Or if the surface density is extremely low, the regolith could fail to stop the Penetrator and it would pull the spacecraft towards the surface. If the penetration works properly the extraction also poses a hazard. Whether extracting the Penetrator or simply the sample, the same risk of collision with the spacecraft exists while it is being reeled in.

For the remote sensing approach, most of the collection mechanisms are incapable of collecting the required sample size without numerous sampling operations (<< 5). This would significantly increase the risk to the spacecraft. A Penetrator collection device could potentially collect a sample of adequate size; however it also creates a significant risk to the spacecraft. Therefore, due to the substantial amount of sample that is required for DSR the remote sampling approach is deemed to be unappealing.

3.3 Long Term Landing

For long term sampling, the spacecraft would land upon and then attach itself to the surface. Due to the low gravity of Deimos, the spacecraft will not remain on the surface without some additional force. For a long term sampling approach this force would be applied by an anchoring system. Several mechanisms could then be used to obtain the sample. Traditional devices, such as drills and coring devices could all be used with this approach as they could act against the anchor to obtain the penetration force. A mechanical arm or similar device could also be used, since the spacecraft would remain on the surface for a sufficient time to allow the slow procedure of manipulating the arm to collect a rock/sample. With the large available collection time, once anchored on the surface, there should be little difficulty in obtaining the size of sample required.

The challenge in this approach therefore is not in selecting a sampling mechanism but in designing the anchoring system and in the amount of time spent on the surface. Due to the unknown nature of the surface it would be difficult to design a system that would reliably provide the necessary anchoring. If the surface density were too low, the anchor would not imbed in the regolith and thus would not provide the necessary counterforce to keep the spacecraft in place during sampling. If the regolith is too dense the anchor could fail to penetrate the surface and the anchoring would also fail.

An anchoring system also limits the potential to have multiple sampling sites. Given the unknown nature of the surface, it would be extremely difficult to design an anchoring system that could safely detach from the surface and then be reused. Therefore the long term landing approach would likely be constrained to a single sampling operation on a single site. This could reduce the scientific return, but multiple sampling sites are not, at this time, a scientific requirement.

The time spent on the surface could also pose additional problems. Real time communication with the Earth will not be possible and all operations, including the complex sampling operations, will need to be autonomous. There could also be power concerns depending on the solar array orientation on the spacecraft.

Due to the increased complexity in requiring an anchoring system and the extended time on the asteroid surface, this approach is less advantageous than the short term landing approach. However, since this approach does meet the science requirement it will not be discarded and will be re-examined if additional challenges are uncovered with the short term landing approach.

3.4 DSR TRS Approach

Remote sampling poses considerable problems in obtaining the required sample size and is therefore unattractive for DSR. The short and long term landing approaches are both capable of obtaining the required sample, however, the long term approach has the added complexity of an anchoring system. Therefore, the short term landing approach or “touch and go” is the selected method for the DSR TRS.

4 PLANETARY PROTECTION STRATEGY

Planetary protection covers two distinct areas; the contamination of a foreign body with material from Earth (*forward contamination*) and the contamination of the Earth with material from a foreign body (*back contamination*).

4.1 Protection against Forward Contamination

According to the COSPAR Planetary Protection Policy [2], to date Deimos has no officially designated forward contamination protection guidelines. It has not been specifically identified for the more stringent categories and will likely fall into Category I or II. The best fit is category II, which includes carbonaceous chondrite asteroids. Categories I and II do not require any methods to prevent forward contamination. Therefore to meet planetary protection guidelines it is likely that no measures will be required with regards to protection against forward contamination. However, for scientific purposes it will be necessary to take steps to avoid contaminating the sample.

4.2 Protection against Back Contamination

The COSPAR Policy outlines the requirements for small body sample return and Deimos does not fit the necessary profile for bodies requiring containment procedures. For a body to be qualified as restricted Earth return, an answer of 'no' or 'uncertain' must be identified in response to six qualifying questions. At this time, it is not believed that Deimos meets this qualification, so it would most likely be classified as unrestricted Earth return.

The Task Group on Sample Return from Small Solar System Bodies [3] agrees with this classification, placing Deimos in the category of bodies where "no special containment and handling is warranted beyond that which is needed for scientific purposes."
However they do qualify this recommendation by declaring that the assessment for Deimos has a lesser degree of confidence than for some other bodies. The task group advises an additional evaluation to confirm their recommendation if a mission is planned.

4.3 DSR TRS Approach

Since, at the current time, there is no recommendation for the implementation of forward or back contamination procedures for a sample return mission to Deimos, the DSR concept does not plan for the implementation of any measures beyond those required for scientific purposes. This approach should also lead to a mission concept that is more applicable to other small solar system bodies, as the majority do not require protection against contamination. If it is later determined that protection procedures are required for Deimos, the DSR concept could be re-assessed.

It should be noted that if back contamination protection is required this will have a severe impact on the mission. Ensuring the containment of the returned sample as well as the cleanliness of the return capsule to meet the required regulations would add significant complexity to the spacecraft. This is currently one of the chief challenges for a Mars sample return mission. In the case that back contamination protection is required, DSR could take advantage of related developments from Mars sample return programs.

5 MISSION ANALYSIS

The mission analysis for the DSR TRS was performed as an initial step in the study for the 2010-2020 timeframe, in a contract with EADS/Astrium. Both low and high thrust scenarios were analyzed along with gravity assists, optimal stay times and Martian orbits, as well as other ΔV reducing measures. The study was used to determine the initial feasibility of the mission concept and it was concluded that it is feasible to return a significant mass using both chemical and combined chemical and Solar Electric Propulsion (SEP) scenarios. However, due to the higher cost of a SEP system, the purely Chemical Propulsion (CP) scenario was selected as baseline.

During the main TRS study, additional mission analysis was performed in the frame of the mission architecture contract with Alcatel Space. This second mission analysis was done in concert with the mission and spacecraft design and thus concentrated on the baseline high thrust option. It was also refined throughout the study for optimization with the selected design.

5.1 EADS/Astrium Mission Analysis

5.1.1 Low Thrust Transfers

To obtain the most favourable low thrust scenarios, a Lunar Gravity Assist is required for Earth departure. A parabolic approach and departure is then used at Mars, and a direct re-entry is implemented upon Earth return. The nominal case also assumes the use of high thrust for orbit insertion and departure at Mars, which would require a separate high thrust stage. The nominal thrust assumed is 200 mN/tonne at 1 AU and is reduced to 100 mN/tonne at Mars to account for solar flux reduction.

Segment	Initial ΔV (CP)	Transfer ΔV (SEP)	Final ΔV (CP)	Transfer Time
Earth-Mars *March 2018*	0.70 km/s	4.52 km/s	0.55 km/s	1.08 yrs
Stay Time	------------	--------------	---------------	78 days
Mars-Earth *July 2019*	0.55 km/s	1.91 km/s	0 (Direct Entry)	0.93 yrs
Total		**6.43 km/s SEP total**	**1.8 km/s CP total**	**Mission Duration 2.22 yrs**

Table 5.1.1-1: Transfer performance for low thrust transfers (2018 launch)

For this scenario the maximum useful spacecraft mass returned to Earth is typically 800 kg for the optimal stay time (which is in general, between 100 and 200 days). Useful mass is defined as the available mass for the spacecraft not including the propulsion system. The total low thrust ΔV is approximately 7 km/s (in the 6.6 - 9 km/s range). As an example of this baseline scenario the 2018 launch case is outlined in Table 5.1.1-1 and 5.1.1-2.

Stay Time (days)	18	48	78	108	138
Chemical Stage 1 Fuel Mass (kg)	129.9	129.9	129.9	129.9	129.9
Chemical Stage 1 Dry Mass (kg)	503.9	503.9	503.9	503.9	503.9
Ion Stage Dry Mass (kg)	310.2	307.3	305.8	308.1	310.8
Ion Stage Fuel Mass (kg)	357.9	348.9	344.2	351.3	359.7
Chemical Stage 2 Fuel Mass (kg)	145.1	145.5	145.7	145.4	145
Chemical Stage 2 Dry Mass (kg)	613.9	616.8	618.3	616	613.3
Mass at Atmospheric Entry (kg)	829.9	838.4	842.9	836.1	828.2
Transfer Rate (kg/year)	376.3	378.3	378.9	377.7	376

Table 5.1.1 -2: 2018 Launch Mass Analysis (2018 Launch)

Eliminating the high thrust stage and using SEP for orbit insertion and departure yields significant mass gains to the nominal scenario, at the expense of time. The use of SEP at Mars for orbit insertion and departure would increase the useful mass from 800 kg to 1180 kg, while adding approx. 150 days in spiral times. The useful mass could be further increased to 1280 kg by also using SEP for Earth escape.

Launch Injection	Soyuz 50000 km	Soyuz 50000 km	Soyuz Direct	Dnepr 40000 km	Dnepr 71000 km	Soyuz 50000 km	Soyuz Direct
Stages	CP-SEP-CP	CP-SEP-CP	SEP-CP	CP-SEP-CP	CP-SEP-CP	SEP	SEP
Mission Duration (days)	1187	1187	1187	1187	1187	1581	1306
Thrust Required at 1.7 AU (mN)	226	74	74	50	52	57	49
Chemical Stage 1 Propellant Mass (kg)	129.9	77.7	0	69.6	56.9	0	0
Chemical Stage 1 Dry Mass (kg)	503.9	172.6	0	131	72.4	0	0
Ion Stage Dry Mass (kg)	312	137.4	137.4	125	126.1	166.1	147
Ion Stage Propellant Mass (kg)	363.5	116.7	116.7	77.7	81.2	206.2	146.5
Chemical Stage 2 Propellant Mass (kg)	145.7	83.1	83.1	70.8	71.9	0	0
Chemical Stage 2 Dry Mass (kg)	618.3	202.7	202.7	136.8	142.6	0	0
Mass at Atmospheric Entry (kg)	817.5	200	200	89.1	98.9	200	200
Launch Mass (kg)	**2890.8**	**990.3**	**740**	**700**	**650**	**572.3**	**493.4**

Table 5.1.1-3: Summary of Performance for 2013 Launch Case

However, such large returned masses are likely greater than that required for DSR. Therefore, an analysis was performed to examine the required launch mass to obtain a specific mass at Earth entry. This was done for the 2013 launch nominal scenario. For a useful return mass of 200 kg, a spacecraft with a total mass of 990 kg must be launched into the optimal 50 000 km apogee elliptical Earth orbit. If launched directly into a trans-lunar orbit, the total spacecraft mass can be reduced to 740 kg.

Further mass gains could be achieved using SEP for Earth apogee raising and Mars insertion and departure, resulting in a total spacecraft mass of 570 kg when using an initial apogee of 50 000 km and 490 kg for a direct launch to trans-lunar orbit. With the mass sufficiently low, alternative launch vehicles can also be considered such as the Dnepr, in order to reduce costs. Table 5.1.1-3 shows the results for the various cases. The first column shows the nominal scenario for the 2013 launch case, with a maximum launch mass, chemical propulsion Earth escape, Mars insertion and Mars escape, and a solar electric propulsion transfer. This scenario returns an 817.5 kg mass at atmospheric entry. The second column shows the reduced mass scenario with a 200kg returned spacecraft mass. The remaining cases vary injection apogee, launch vehicle and planetary escape/insertion propulsion methods for a 200kg returned mass.

5.1.2 High Thrust Transfers

The baseline scenario for high thrust transfers uses half revolution transfers to and from Mars. A direct entry is envisioned at Earth return since an Earth orbit insertion would not be feasible with the mass constraints of using a Soyuz Fregat 2B launch vehicle. The main concern of this scenario is the long required stay times at Mars of about 450 days. These can be decreased with a 1.5 revolution transfer scenario, however the transfer time is increased and the overall mission time remains relatively unchanged.

For the nominal high thrust scenarios the optimum total ΔVs are around 7 km/s and the optimum stay time ranges between 330 and 550 days. The optimum transfers are outlined in Table 5.1.2-1. The type of transfer is also noted for each segment, whether it is a 0.5 (short) or 1.5 (long) revolution transfer.

Launch Date	Earth-Mars Transfer	Mars Departure Date	Mars-Earth Transfer	Stay Time (days)	Mission Duration (yrs)	Total DeltaV (km/s)
10-Nov-11	0.5 rev.	8-Aug-13	0.5 rev.	331	2.71	2.67
7-Dec-13	0.5 rev.	15-Mar-15	1.5 rev.	169	3.45	3.04
17-Jan-16	0.5 rev.	19-Mar-18	0.5 rev.	515	2.75	3.29
25-Oct-17	1.5 rev.	16-Jun-20	0.5 rev.	122	3.16	3.03
12-May-18	0.5 rev.	21-Feb-19	1.5 rev.	79	3.05	2.73
10-Nov-19	1.5 rev.	10-Jul-22	0.5 rev.	154	3.42	3.08

Table 5.1.2-1: Optimal High Thrust Transfers

The useful masses available at Earth return were also analyzed for the optimum transfers and can be found in Table 5.1.2-2. The analysis assumes the use of a CP system with a specific impulse of 320s and that the maximum capacity of the Soyuz Fregat 2B (2890.8 kg) is employed to launch into a HEO (200 x 25 000 km). The transfers have two CP stages in order to maximize the Earth returned mass as it was found that the CP-CP staged transfer has a mass advantage over a single CP transfer.

The masses at atmospheric entry in Table 5.1.2-2 represent the maximum spacecraft dry mass remaining upon reaching the Earth's atmosphere. For these transfer scenarios, the masses range between 200 and 300 kg. Leaving mass behind at Mars or increasing the performance of the CP system (i.e. increasing Isp) could increase this mass. However, preliminary analysis indicates that the mass at atmospheric entry for these transfer cases should be adequate for the needs of DSR.

Launch Epoch	**2011**	**2013**	**2016**	**2017**	**2018**	**2020**
Chemical Stage 1 Dry Mass (kg)	746	757.7	805	761	724.8	746.2
Chemical Stage 1 Fuel Mass (kg)	162.5	164	170	164.4	159.7	162.5
Chemical Stage 2 Dry Mass (kg)	1065.3	1114.3	1101.5	1110.4	1160.3	1132.6
Chemical Stage 2 Fuel Mass (kg)	198.3	203.1	201.9	202.7	207.6	204.9
Transfer Rate (kg/year)	121.2	75.9	80.9	82.9	79.3	74.4
Mass at Atmospheric Entry (kg)	**328.7**	**261.7**	**222.4**	**262.2**	**248.3**	**254.5**

Table 5.1.2-2: Mass Available at Atmospheric Entry for High Thrust Transfers

5.2 Alcatel Mission Analysis

The mission analysis was again examined for launch in the 2010-2020 timeframe by Alcatel Space; with a baseline of high thrust transfers. Three launch opportunities for each optimal launch year were analyzed for compatibility with mission requirements. The transfers for the 2020 launch case were modified from optimal in order to incorporate a minimum stay time of 90 days at Mars. The resulting required deltaVs, transfer durations and available stay time at Mars for each case are depicted in Figures 5.2-1 and 5.2-2. From a preliminary mass assessment the 2016 launch cases were discarded due to their low performances. Of the remaining cases, the 2013 launch case provided the lowest performance and was therefore used for the design case. The design will consequently be compatible with the 2011, 2013, 2018 and 2020 launch cases.

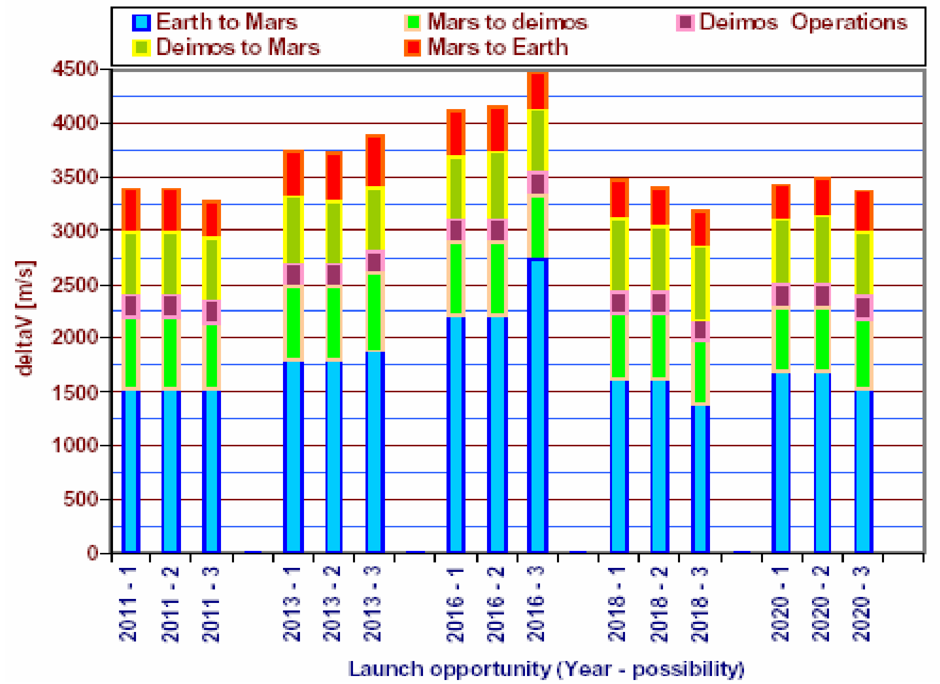

Figure 5.2-1: Velocity Impulse Budget for Various Launch Opportunities

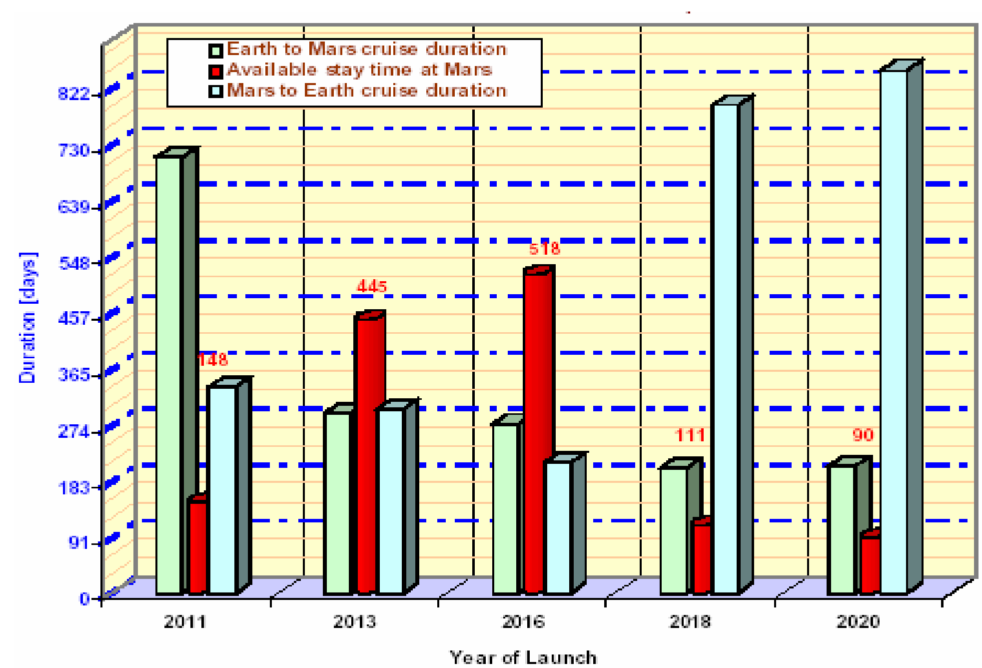

Figure 5.2-2: Mission Timeline Aspects

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Figure 5.2-3 shows the mission performance for the various launch epochs. The figure depicts the launch mass margin evolution based on the spacecraft dry mass associated with the sizing case (2013 launch) as well as with spacecraft dry masses adjusted to each mission opportunity. The adjustment consisted of modifying the size of the propellant tanks for the different mission deltaV budgets. The mission performances demonstrate that the DSR concept would be feasible with a launch mass margin greater than 20% for several launch opportunities in the 2010-2020 timeframe.

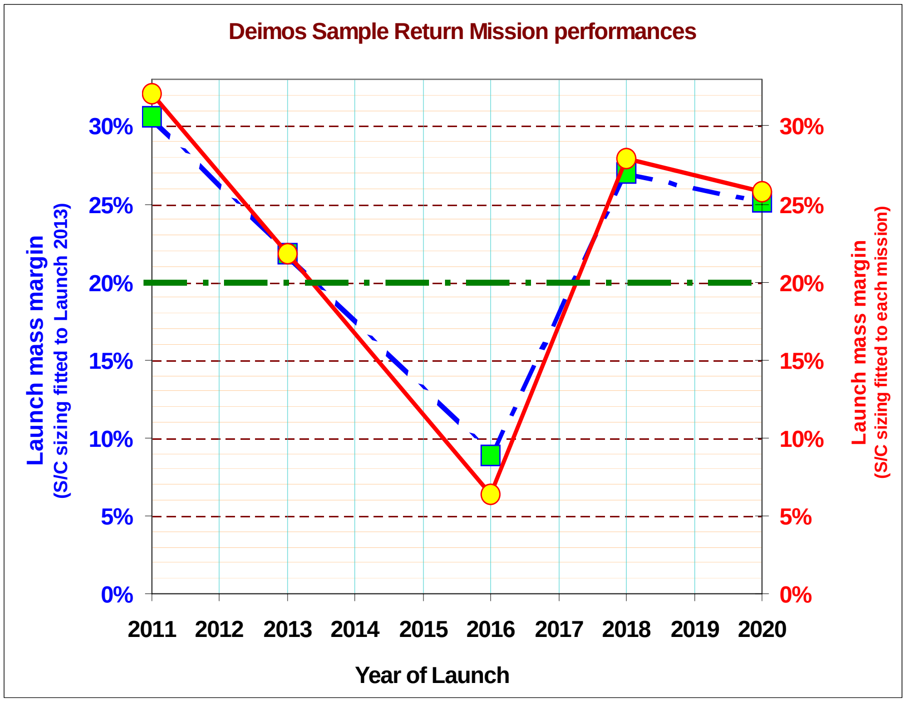


Table 5.2-1: Evolution of the Launch Mass Margin vs. the Year of Launch

5.3 Launch Windows

Earth Departure	Earth Return
May 2011	August 2014
May 2013	October 2016
January 2016*	October 2018
May 2018	May 2021
July 2020	August 2023

* Does not have a Launch Mass Margin of ≥ 20%

Table 5.3-1: DSR Mission Opportunities

6 DSR ARCHITECTURE

The DSR spacecraft will be launched into the Mars-Deimos System on a Soyuz Fregat 2B. The SF2B will launch the spacecraft into a 200 x 25 000 km Earth orbit, after which the spacecraft will begin its transfer to the Martian system. Upon reaching Mars, the spacecraft will be placed into an intermediate elliptical orbit (500 x 100 000 km) before performing a series of manoeuvres to enter into a co-orbiting trajectory with Deimos (20 069 km circular orbit). The spacecraft will then enter into an observation orbit and will perform measurements of Deimos' surface and gravitational properties before obtaining the samples. Once the samples are obtained they will be transferred into a canister inside the Earth Return Capsule (ERC). Components of the spacecraft that are no longer required, such as the sampling mechanism and empty tanks, will then be separated and left in Martian orbit to reduce propellant requirements for the transfer back to Earth. Upon approaching Earth, the ERC will separate and perform a direct re-entry while the main spacecraft body will be diverted into open space.

6.1 Deimos Observation Phase

Once the spacecraft has entered into a co-orbiting trajectory with Deimos it will be placed into a Deimos Observation Orbit (DOO) in order to observe the surface before performing sampling manoeuvres. This observational orbit (Figure 6.1-1) will be achieved by slightly modifying the eccentricity of the spacecraft's orbit, with respect to that of Deimos.

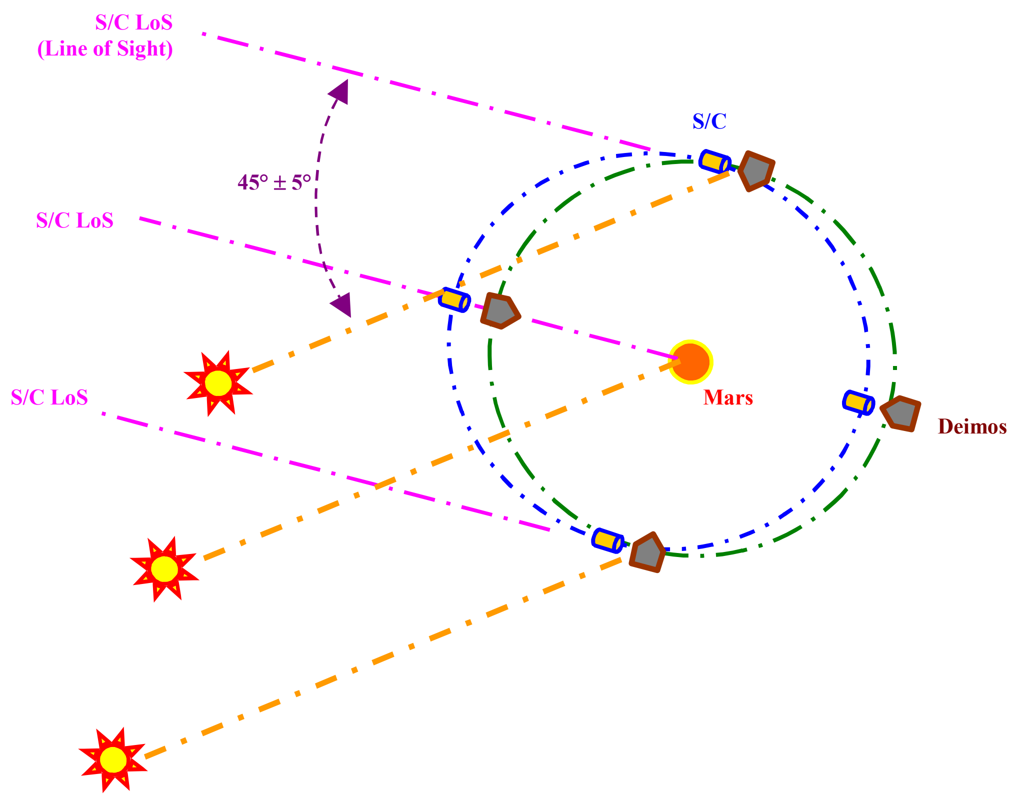


Figure 6.1-1: Deimos Observation Orbit Configuration

The difference in eccentricity will produce a relative elliptical motion about Deimos with a 30-hour period. This observational orbit will permit the examination of a large number of sampling sites on the surface, enabling the selection of the most optimal locations for sampling manoeuvres. The repetitive motion about Deimos will also aid in the accurate mapping of its gravitation field, which will be needed for the determination of the navigation sequence for sampling manoeuvres.

After initial observations, the distance to the surface can be decreased, in order to further investigate potential sampling sites, by varying the eccentricity. Altitude changes only require about 0.2 m/s of deltaV for a change of +/- 15 km. However, the cumulative deltaV for maintaining the spacecraft in a low orbit is not negligible and must be considered for all low altitude operations. The deltaV required for maintenance at various altitudes is depicted in Figure 6.1-2. Therefore the spacecraft will nominally maintain a higher altitude with short durations at lower altitudes as required for observational measurements.

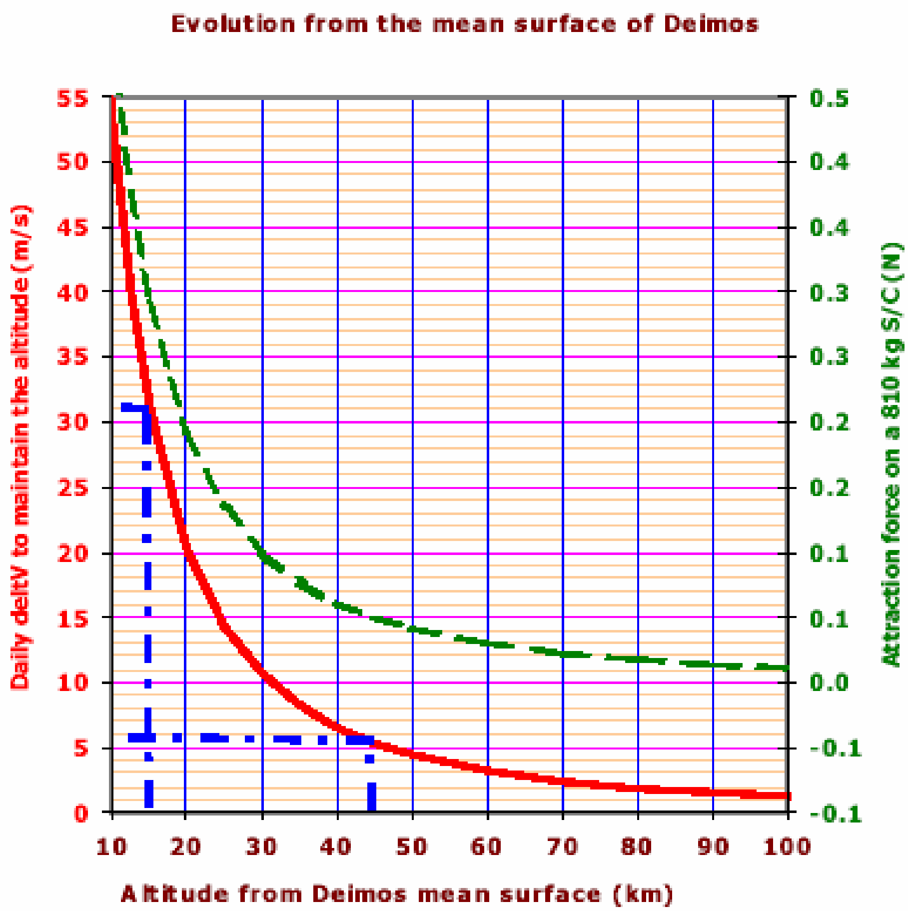


Figure 6.1-2: Required DeltaV to Maintain Altitude for Deimos Observations

6.2 Sampling Operations

A touch-and-go sampling method has proven optimal, in terms of spacecraft and mission requirements, although it introduces several challenges for collecting the sample. The sample has to be collected in a very short time frame (~3 sec.) and it is unlikely that a 1 kg sample could be collected during a single manoeuvre so several manoeuvres would have to be performed. However, this provides the added benefit of allowing multiple sampling sites. The spacecraft dynamics during the touch-and-go sampling manoeuvre are depicted in Figure 6.2-1.

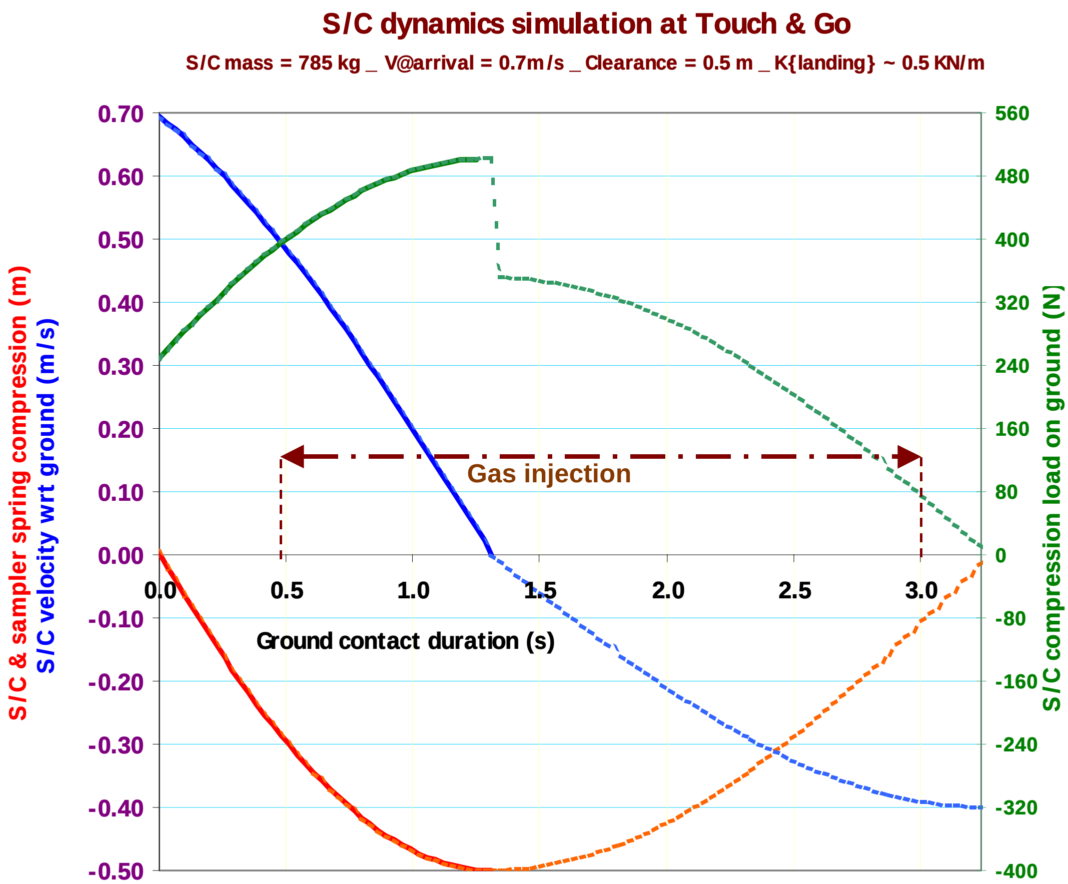


Figure 6.2-1: S/C Dynamics during the Touch-and-Go Maneuver

Performing a rendezvous or landing manoeuvre on the surface of a small body, with a small gravitational field, presents several challenges, mainly in terms of guidance, navigation and control requirements. Deimos' irregular shape also produces an irregular gravitational field that must be compensated for, in order to ensure spacecraft survival. The survival of the spacecraft before, during and after the sampling manoeuvre is critical, so the navigation approach for these operations must be strictly controlled. The details of the autonavigation approach for the sampling operations can be seen in Table 6.2-1.

S/C altitude	S/C Vertical Velocity	Sustained deltaV (g[Deimos])	Performed deltaV	Duration	Used sensors	Freq. (TBC)	Comments
15000 m --	0 m/s			start			Start from departure orbit
 1500 m --	-3 m/s -6 m/s	-3 m/s	- 3m/s +6 m/s	56.1 min	NAC WAC uLAT	1 Hz 1 Hz 8 Hz	Navigation on NAC with WAC assistance Descent on the vertical direction Control of lateral velocity (counteracted) Arrival at Altitude = 1500m
1500 m -- 1500 m --	-0.42 m/s +0.42 m/s	-0.83 m/s	+1.25 m/s	5 min	NAC WAC uLAT	1/32 Hz 1/32 Hz 1/32 Hz	5 minutes parabollic flight for refined imaging Target adjustment WAC / NAC calibration (free fall conditions) Start of navigation on WAC
1500 m -- 100 m --	-1 m/s -3.24 m/s	-2.24 m/s	-0.56 m/s +2.99 m/s	11.4 min	WAC uLAT	1 Hz 8 Hz	Descent on the vertical direction Control of lateral velocity (counteracted) Arrival at Altitude = 100m
100 m --	-0.25 m/s -0.25 m/s	-1.69 m/s	+1.69 m/s	6.7 min	WAC TLR	1 Hz 8 Hz	Slow descent on the vertical direction Accurate control of lateral velocity (counteracted) Accurate control of the S/C trim angle
0 m -- 0 m --	0.25 m/s ~ 0 m/s			5 sec	WAC TLR	1 Hz 8 Hz	Touch and Go
0 m -- 6100 m --	+5.5 m/s 2 m/s	-3.5 m/s	+ 5.5 m/s	30 min	WAC TLR	1/32 Hz 8 Hz	Deimos departure in Rough GNC mode Acquisition of the Nominal GNC mode
6100 m -- 15000 m --	3.33 m/s 0.15 m/s	-3.18 m/s	+ 1.33 m/s	102 min	WAC uLAT	1/32 Hz 1/32 Hz	Refined navigation to return on departure orbit Return on the departure orbit
		total :	*22.3 m/s*	*211 min*			*1h19mn to arrive / 2h12 min to go back*

Table 6.2-1: Touch-and-Go Sampling Sequence

Due to the lag time in communication between the Earth and the Martian system, real time control during these critical manoeuvres will not be possible. Therefore a highly autonomous guidance, navigation and control system must be developed to ensure feasibility of such a mission. The details of the GNC instruments required for the touch-and-go sequence are outlined in Table 6.2-2. These instruments will also be used during the Deimos Observation Phase, to determine the landing location and refine the touch-and-go sequence.

Instrument	Acronym	# Units	Mass (kg)	Power (W)	Data Rate (kb/s)	Dimensions (cm)	FOV (deg)
Narrow Angle Camera	NAC	2	0.5	0.65	1	150 x 40 x 40	4
Wide Angle Camera	WAC	2	0.12	0.65	1	70 x 80 x 75	40
Micro Laser Ranger	µLat	1	2	2	0.5	150 x 100 x 80	200 µrad
Triple Laser Ranger	TLR	2	1.2	3	0.6	70 x 100 x 50	1 mrad
DPU + CPS		1	1.5	3.31		10 x 4 x 15	
Structures			1				
Margin (20%)			1.63	2.78	1.14		
TOTAL			**9.77**	**16.69**	**6.84**		

Table 6.2-2: GNC Instruments for Touch-and-Go Sequence

6.3 Staging

Several staging scenarios were considered for DSR. The optimal design consists of a single unique spacecraft for the entire mission with a two-stage propulsion system. The Earth Return Vehicle (ERV) consists of all spacecraft components that are needed for both the forward and return journeys. The Propulsion Stage incorporates all systems that can be left behind at Deimos and are not required for the return journey. This includes the first propulsion stage, the GNC payload required for sampling operations as well as the sampling mechanism itself. This stage will be separated after sampling is completed and prior to the return to the Mars intermediate orbit. A breakdown of the components in each stage can be found in the mass budget in Section 10.

6.4 Earth Return

The return to Earth is the reverse of the Mars transfer trajectory. After sampling operations have been completed and the propulsion stage is separated from the ERV, the spacecraft will return to the Martian intermediate elliptical orbit (500 x 100 000 km). From this orbit the ERV will begin its return transfer to Earth.

For Earth return two strategies were considered, a direct Earth entry or an Earth orbit insertion. After inserting into an Earth orbit the spacecraft could rendezvous with the ISS to deliver the samples. However, this option was discarded due to the prohibitively high deltaV required to perform the insertion manoeuvre. Therefore, upon Earth approach the ERC will separate from the ERV. Then the ERV will be diverted into open space while the ERC will perform a direct entry into the Earth's atmosphere. The entry will be targeted towards a specified landing area where the ERC will be retrieved following touch-down.

7 SAMPLING SYSTEM

7.1 Landing Pads

For the touch-and-go sequence, three landing pads have been implemented at the base of the spacecraft to provide shock absorption and stability (120 deg. distribution). The sampling devices have been integrated into the centre of the pads, so that there will only be three contact points with the surface (Figure 7.1-1). These pads have been designed to absorb the impact and provide the 3-second requisite surface contact time before rebounding and pushing the spacecraft away from Deimos. The sampling mechanism can also withstand a tilt of +/- 15 deg. to cope with uneven terrain. The design provides 300 mm of clearance for the sampling mechanism and an extra 200 mm incorporated in the landing legs, which provides a total safety clearance for the spacecraft of 0.5 m.

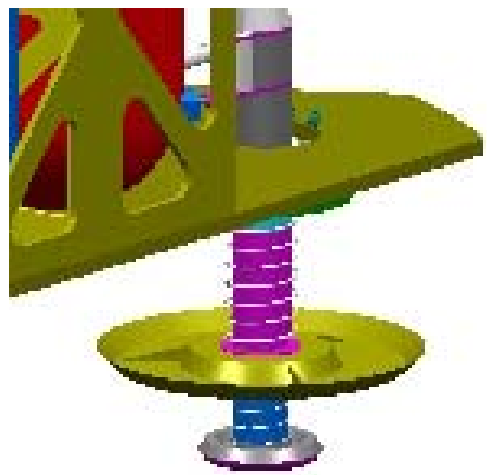
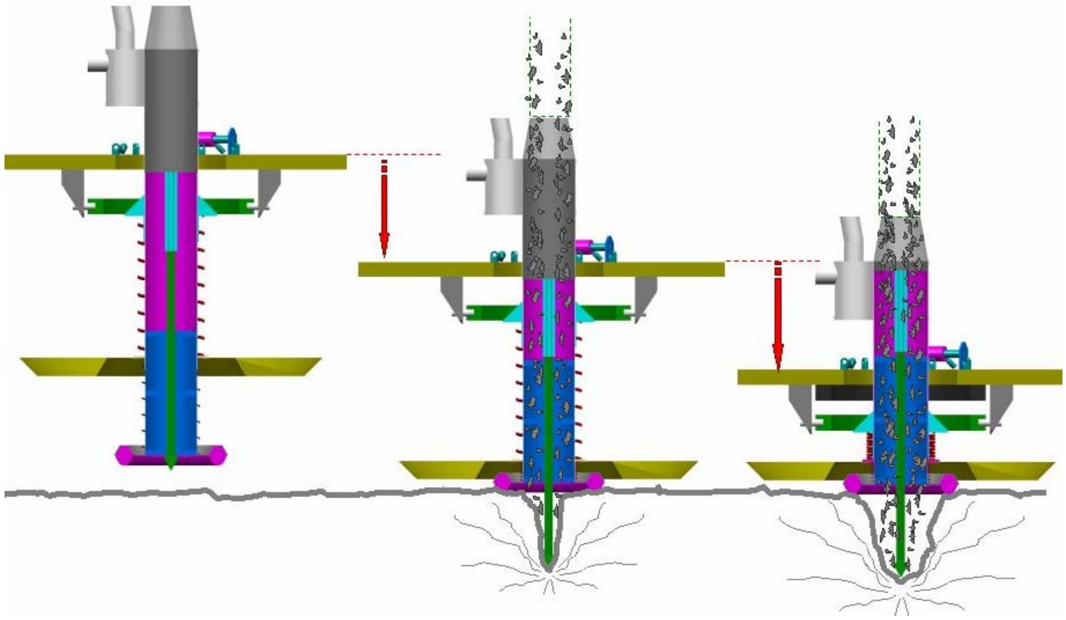


Figure 7.1-1: Sampling Mechanism integrated into Landing Pad

7.2 Sampling Mechanism

The sampling mechanism devised for DSR consists of a pneumatic device with a small sting appendage (Figure 7.2-1). When the device comes into contact with the surface, the sting will be fired into the regolith creating a pocket of loose material. A stream of pressurized gas is then expelled backward from the tip, propelling the material up into the transfer tube. This will all take place during the 3 seconds of contact with the surface. Afterwards, a second gas jet is then used to propel the material further up the transfer tube and into the sample canister. The pressurized gas used for the sampling operations, helium, will be provided by the propellant pressure system from the Propulsion Stage.



Deployed configuration *After first stage compression* *Final compression*

Figure 7.2-1: Pneumatic Sampling Mechanism

8 EARTH RETURN CAPSULE (ERC)

8.1 Baseline Design

Once separated from the main spacecraft the ERC will perform a direct Earth re-entry, with a velocity of ~12 km/s. The design of the ERC consists of a fully passive capsule in order to ensure the integrity of the sample container with no signal point failures such as parachutes or airbags. The capsule is required to cope with the hypersonic entry phase, subsonic descent and hard impact on the ground. The mass budget (Table 8.1-1) is based on the advanced PICA15 (227 kg/cm^3) TPS thermal ablative material with a thickness between 32 mm and 65 mm. The development of this material is ongoing and it should be available in the required timeframe.

ERC Mass	Kg
Front Shield	22.2
Primary Structure	1.8
Protection Cover	2.6
Sample Canister	3.7
Sample	1
Mechanisms	2
Thermal Control	1.1
Avionics	1.5
Shock Absorption Device	4.1
TOTAL	**40.1**
TOTAL (20% margin)	**48**

Table 8.1-1: ERC Mass Budget

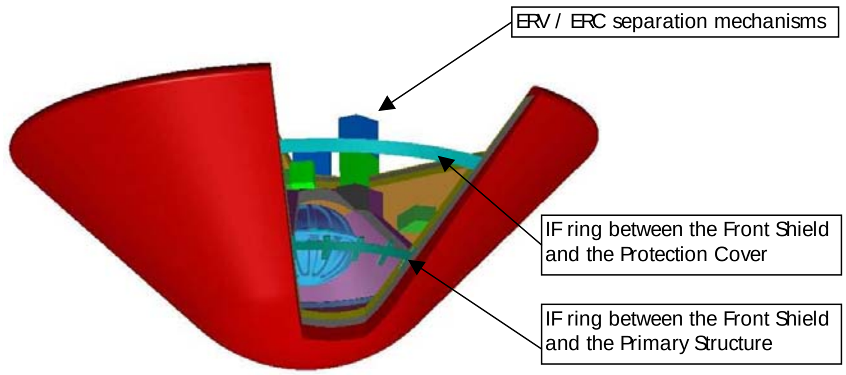


Figure 8.1-1: ERC Baseline Design

The shape of the capsule has a typical ballistic coefficient of about 0.68 during the subsonic phase. It has a diameter of 1.30 m and mass of 48 kg. The design impact velocity of the ERC is 30 m/s (including a 5 % margin). The sample canister and its shock absorption system (Open Cell RVC Foam) have therefore been designed to cope with this impact velocity (impact shock ~550 g) to ensure containment of the sample. The interior design of the capsule is depicted in Figure 8.1-2.

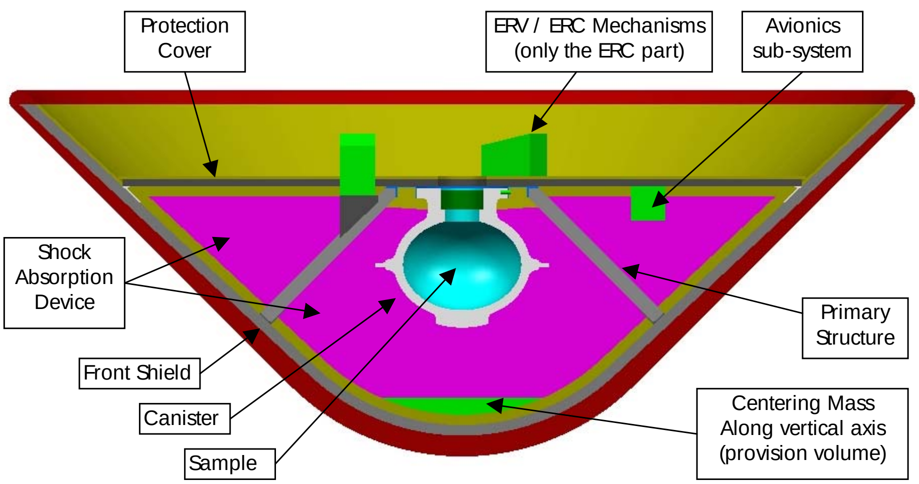


Figure 8.1-2: ERC Cross Section View

The design of the ERC is based on newly developed thermal ablative material (PICA15) and has a low mass fraction compared to traditional entry vehicles. An extensive testing and development phase is therefore recommended for the ERC.

8.2 ERC Canister

The design of the ERC canister consists of a Kevlar body with metallic liners and a titanium window for the sample insertion. The canister is filled by the pneumatic transfer system, using a filter, which captures the sample material while allowing the gas to escape (Figure 8.2-1). The main design requirement was for the canister to remain airtight under a 550-g load upon ground impact.

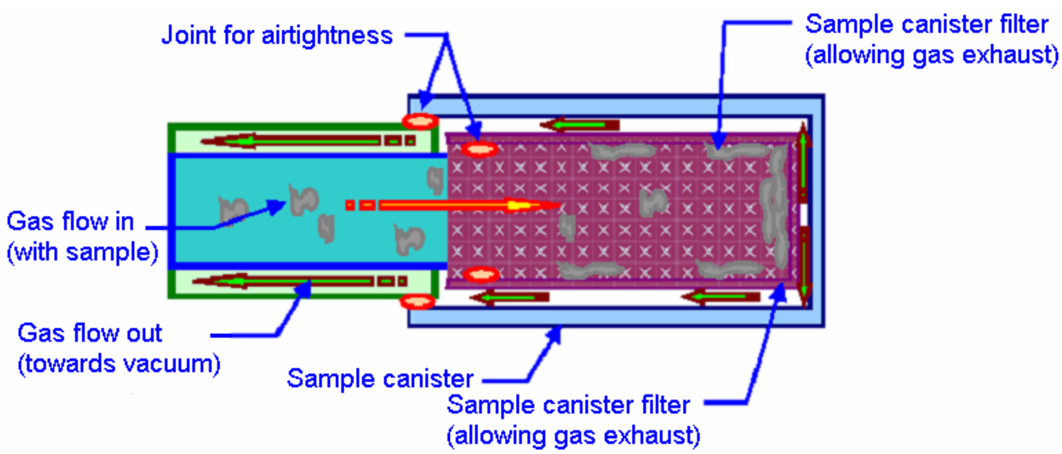


Figure 8.2-1: Filling of the ERC Canister

The design of this system is critical and there are many challenges in being able to fill the canister. Beyond the difficulties of transferring the sample into the container there are further problems that need to be addressed. There is currently no method to determine the amount of material inside the container. If the container becomes overfilled the seal will not be able to close. There is also no way to determine if there is too little sample inside. Depending on the efficiency of the system it might be necessary to measure the contained sample to ensure that the container is adequately filled before Earth return is initiated.

8.3 Localization System

Upon impact the ERC will need to enable rapid localization and recovery of the sample. The primary localization system is based on the Galileo system and will be activated upon release of the ERC from the DSR spacecraft. The system will remain active for 4 days, allowing 2 full days of transmission after landing. However, using the Galileo satellites, the system is capable of detecting the position of the capsule to within an accuracy of 100 m before landing (during the subsonic descent). This could lead to the elimination of the need to protect the localization system on impact, which would provide a mass and stability benefit to the ERC design. A redundant secondary system for localization is also included in the design, based on the Argos positioning system. The ERC will be designed to protect this redundant system so that it will be able to withstand the impact and will operate after landing.

9 DSR SPACECRAFT CONFIGURATION

The main driver for the design of the spacecraft configuration is its capability to perform the touch-and-go sampling manoeuvre. The baseline design consists of two stages, the Propulsion Stage and Earth Return Vehicle (ERV). The Propulsion Stage (Figure 9-1) consists of the main engine for the propulsive manoeuvres prior to sampling (Earth-Mars transfer, MOI, Mars orbit manoeuvres, DOO), the GNC instruments, the mechanisms for obtaining and transferring the sample and the landing pads.

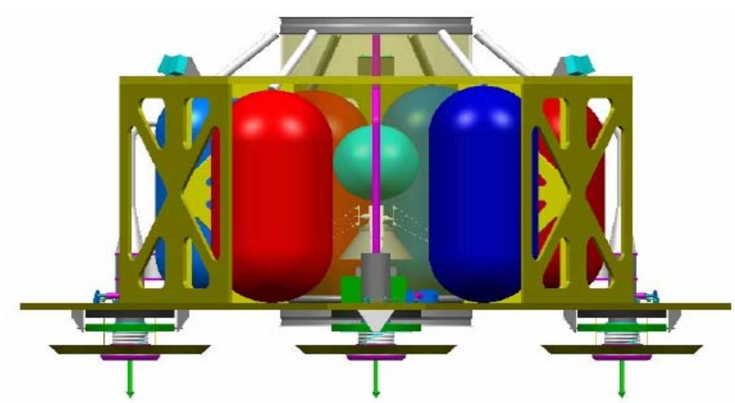


Figure 9-1: Propulsion Stage with Landing Pads in Stowed Configuration

After the sampling operations the Propulsion Stage is no longer required and will be separated and left behind in Martian orbit. For this reason the Propulsion Stage was placed 'below' the ERV (Figure 9-2) so that it could act as a buffer in case of any hard impact during the touch-and-go sequence. All components required for the return transfer are placed on the ERV, at as great a distance from where the spacecraft will contact the surface as possible. An effort has also been made to protect these components from any ejecta that result from the landing by including the top platform. With the delicate components on the top, the platform should block the ejecta.

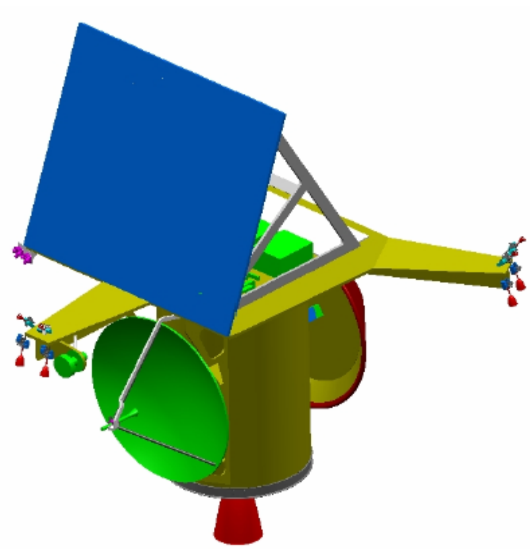


Figure 9-2: Earth Return Vehicle

In order to withstand the impact, the solar array is fixed and placed in a manner to minimize possible damage or degradation from ejecta (on top). The HGA and ERC are located on the sides of the ERV, facing in opposite directions. Two LGA antennas have been placed on the extreme edges of the upper panel. Thrusters for attitude control and to control the touch-and-go sequence are also located on these edges of the upper panel.

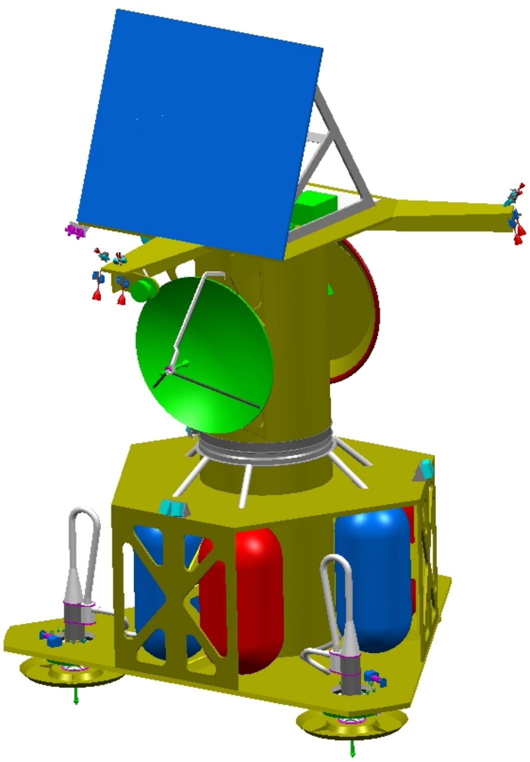


Figure 9-3: DSR Spacecraft

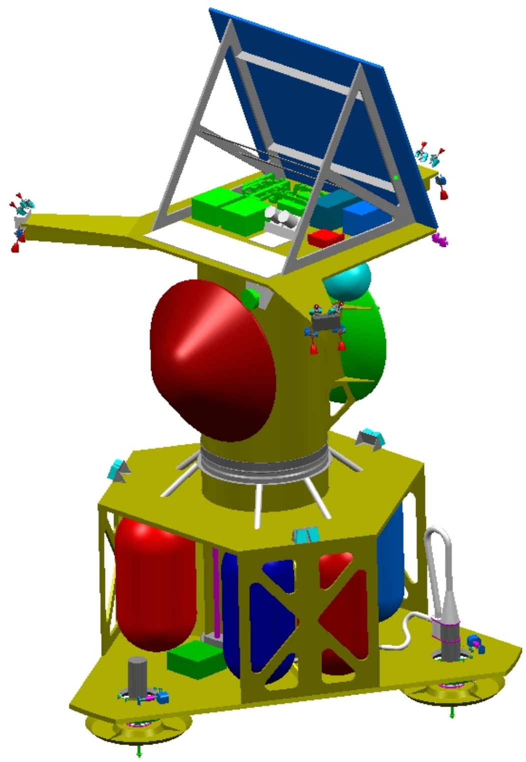


Figure 9-4: DSR Spacecraft

10 DSR MASS BUDGET

Table 10-1 presents a quick overview of the DSR Spacecraft mass budget, with the two spacecraft stages and the ERC.

Spacecraft Mass	kg
Propulsion Stage	
Landing Device	2.9
Mechanisms	8.4
Navigation Control	15.6
Power	9.1
Propulsion	120.9
Sampling Device	31.9
Structure	121.6
Thermal Control	16.0
Earth Return Vehicle	
Sample Transfer Device	7.3
AOCS	20.1
Communication (X Band)	16.5
Data Handling	14.4
Power	40.5
Propulsion System	58.0
Structure	70.7
Thermal Control	14.0
ERC Feeding System	3.9
ERC Mass (with Sample)	48.0
TOTAL	**619.5**
TOTAL (20% margin)	**743.4**

Table 10-1: Spacecraft Dry Mass Budget

11 ENABLING TECHNOLOGIES

These technologies fall into two categories, those required to collect a sample from Deimos' surface and those required to return that sample to Earth. The enabling technologies for sample collection include a highly autonomous guidance, navigation and control system, capable of performing the collection manoeuvre on a small, low gravity body, as well as the mechanisms for obtaining and transferring the sample. The main technology required to enable the Earth return is an Earth Return Capsule designed for the 12 km/s entry velocity and for a passive descent.

11.1 Sampling System

The design and development of a sampling mechanism capable of collecting a significant volume of regolith from a small body is critical for the feasibility of a small body sample return mission. The mechanism also needs to be compatible with the touch-and-go sampling method and thus must be able to collect its sample in the approximately 3 seconds of contact with the surface.

Due to the uncertainty of the composition of the regolith on Deimos the sampling mechanism must be compatible with a large variety of particle sizes and surface solidity. It must also be highly efficient in the low gravity, vacuum environment. The other challenges involve making the system reusable and robust enough to survive multiple sampling operations. For these reasons a development and test campaign is recommended for the sampling mechanism.

The sample handling and transfer mechanism is also essential for the mission. This system should be developed in concert with the sampling mechanism itself in order to ensure compatibility and to integrate the systems as much as possible. It might prove necessary at a further stage to determine the amount of sample that is transferred into the sample canister, to prevent overfilling and to ensure an adequate sample before Earth return is initiated. This need should be carefully considered once the efficiency of the sampling system has been assessed.

11.2 Highly Autonomous Guidance, Navigation and Control System

The development of a highly autonomous GNC system for proximity operations around a small solar system body is required. Performing a rendezvous or landing manoeuvre on the surface of an irregular body, with a small, non-uniform gravitational field, presents several challenges and the survival of the spacecraft is critical, so the sampling sequence must be strictly controlled.

Due to the lag time in communication between the Earth and the Martian system, real time control during these critical manoeuvres will not be possible. Therefore the guidance, navigation and control system must be highly autonomous.

One of the key developments for this system is a ground mark tracking system to enable the manoeuvring of the spacecraft towards the selected landing site. The other main development is related and entails the capability to perform autonomous GNC with a quick reaction time using spacecraft sensors.

11.3 Earth Entry Vehicle

A passive return capsule capable of coping with a hyperbolic entry trajectory (12 km/s) is necessary for bringing a sample back to Earth. The ERC requires an advanced thermal protection system and it must be able to provide containment of the sample during impact as well as enable rapid localization and recovery of the sample.

The critical developments for the ERC are the ballistic design and the ablative thermal protection system. The ERC must be able to recover a stable attitude during the subsonic phase. Further analysis and testing will be required to ensure this stability, which will enable a suitable attitude for landing. For the TPS it is essential to develop and qualify a material that can withstand peak heat fluxes of around 5 MW/m^2, act as efficient thermal protection with a reasonable thickness and stay within the specified mass limits. The material is foreseen to be from the Phenolic Impregnated Carbon Ablator family.

12 Applicability for Other Small Body Sample Return

The enabling technologies for the DSR TRS are similar to those required for any small body sample return. Although some of the specifics will change depending on the target body, both a highly autonomous GNC system and a sampling mechanism will be required. The sampling mechanism could change substantially however if the touch and go sampling approach is not used. The highly autonomous guidance, navigation and control system will have similar requirements for any asteroid target body. All asteroids have relatively low gravitational fields, due to their size, and the field will be irregular due to the asteroid's shape. The ERC development is required for any sample return mission. The specifications could change however if it is determined that more (or less) than 1 kg of sample is to be accommodated.

The most critical impact on the enabling technologies for a small body sample return is if it is deemed that protection against back contamination is needed from the selected target body. The impact on the system design is very high to implement the measures necessary to ensure this protection. This would also add some critical technologies to the list of those necessary for the mission. A series of protective covers would be needed to ensure that the ERC is not contaminated by asteroidal material during sampling operations or at any other stage in the mission. A system to seal the sample canister that would be capable of detecting any failure would also be necessary. The addition of this requirement would add significant complexity to the mission.

13 List of Abbreviations

CP	Chemical Propulsion
CPS	Central Processing System
DOO	Deimos Observation Orbit
DPU	Data Processing Unit
DSR	Deimos Sample Return
ERC	Earth Return Capsule
ERV	Earth Return Vehicle
GSTP	General Support Technology Program
HGA	High Gain Antenna
HIPS	Highly Integrated Payload Suite
LGA	Low Gain Antenna
LIDAR	Light Detection and Ranging Instrument
LV	Launch Vehicle
MAV	Mars Ascent Vehicle
MOI	Mars Orbit Insertion
MSR	Mars Sample Return
ROM	Rough Order of Magnitude
SEP	Solar Electric Propulsion
TBC	To be confirmed
TBD	To be determined
TPS	Thermal Protection System
TRS	Technology Reference Study
TRP	Technology Research Programme

14 ACKNOWLEDGMENTS

The Deimos Sample Return Technology Reference Study was performed with the technical assistance of:

- Alcatel Space in the spacecraft and mission conceptual design, under ESA contract 18050/04/NL/HB,
- COSINE Research B.V in the determination of the payload, under ESA contract 18051/04/NL/HB, and
- EADS Astrium in the study of the preliminary DSR mission analysis, under ESA contract 17241/03/NL/HB.

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16 APPENDIX

16.1 ERC Dimensions

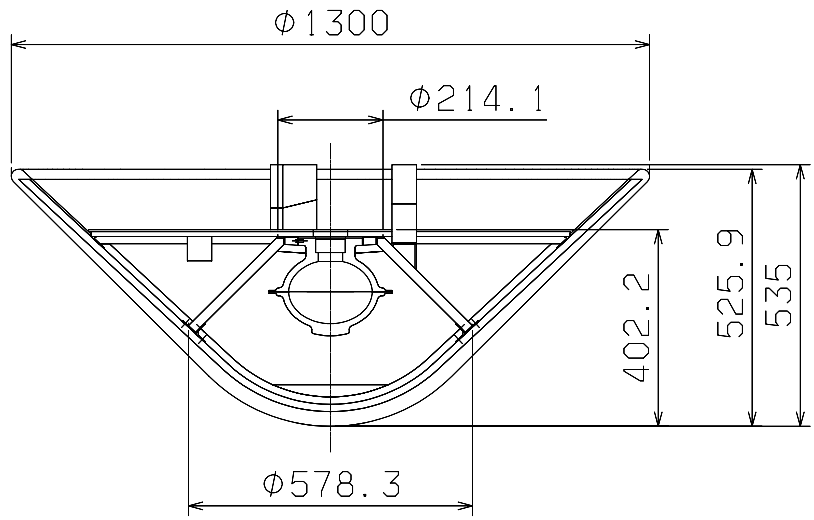

Figure Appendix-1: ERC Dimensions

16.2 DSR Spacecraft Dimensions

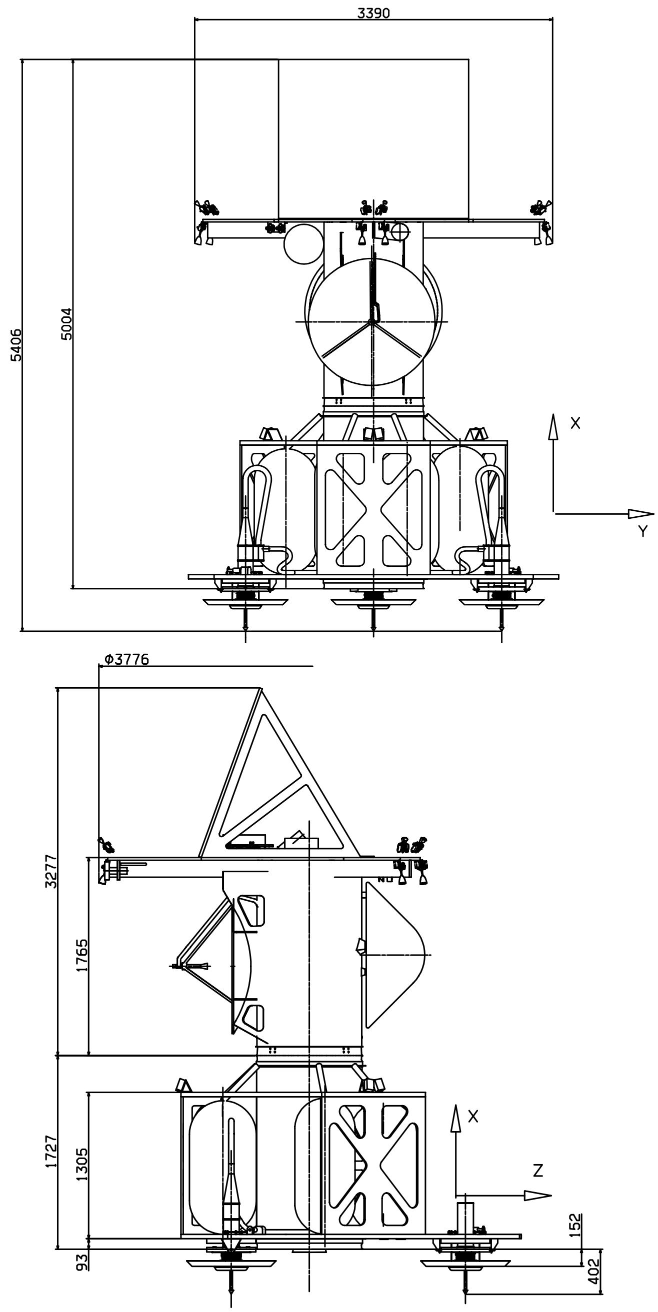

Figure Appendix-2: Dimensions of the DSR Spacecraft