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Study into In-orbit Servicing of the Rosetta Mission

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This paper presents results of a research project defined by the Luxembourg Space Agency in collaboration with the University of Luxembourg. The research is undertaken by a group of researchers and students from the Interdisciplinary Centre for Security, Reliability and Trust of the University of Luxembourg. The project is based on the ESA's deep space mission, Rosetta, that was operational from 2004 until 2016 with the objective of studying comet 67P/Churyumov-Gerasimenko.
keywords: orbital refueling, in-space servicing, mission design, mission architecture, Rosetta, space systems

1. Introduction

The current study is undertaken as a part of a larger project defined under a collaboration between the Luxembourg Space Agency (LSA) and SNT Research Center of the University of Luxembourg. The goal of the project is to demonstrate benefits of in-situ resource utilization (ISRU) in reference with legacy ESA missions. Three previous ESA mission are selected to be studied and redesigned considering application of in-situ resource utilization. The selected mission are: Mars Express (launched 2003), Herschel Space Observatory (launched 2009) and Rosetta space probe (launched 2004). The current study is focused on reverse engineering the Rosetta mission and spacecraft in presence of in-orbit refueling capability assuming presence of a propellant-depot in the cislunar space. Several options are being considered for the propellant depot orbit in the study.

The original Rosetta mission had a complex mission profile and performed several planetary fly-bys on its 10 plus years long journey to the comet. Thus, the study is driven by trajectory analysis where two primary mission phases are modelled. The first mission phase addresses the trajectory from Earth to propellant depot while the second phase addresses trajectory analysis from propellant depot to the destination of the Rosetta spacecraft, Comet 67P.

The spacecraft system configuration and related subsystem changes associated with both these phases are then addressed following the trajectory analysis. In addition to the trajectory derived factors (delta-V, time-of-flight etc.), refuelling operations also affect how the spacecraft design changes.

The aim of the first mission phase (Earth-to-Propellant Depot) is to minimize the launch mass for any configuration of the spacecraft. The second phase (Propellant Depot-to-Comet 67P) presents a greater number of options where multiple trajectory scenarios are possible. The simplest scenario being a direct transfer trajectory from the Propellant Depot to Comet while another scenario could be to perform a lunar fly-by on the way to the comet. Similarly, other planetary fly-bys are also possible. The objective of the second phase is to perform a trade-off of the viable interplanetary trajectories and select the combination that optimize the total launch mass and/or total time of flight.

The second section presents an overview of the original Rosetta mission and the spacecraft. This is referred to as Rosetta Baseline as the subsequently redesigned Rosetta mission is measured with respect to the baseline mission. The redesigned mission and spacecraft are referred to as Rosetta 2.0. The third section presents the overall methodology and tools applied in the current study. Section four provides details of the trajectory analysis. Section five addresses ISRU related aspects considered for the refueling operations. The sixth section presents the results for the mission architecture and spacecraft modeled for Rosetta 2.0.

2. Rosetta Baseline

The Rosetta mission originated in 1984 as a Comet Nucleus Sample Return Mission with the ESA and NASA but was eventually approved in 1996 by ESA as an ESA-only mission. Launched in 2004 from Kourou on an Ariane 5 with the objective of studying comet 67P/Churyumov–Gerasimenko (also simply referred to as 67P). Rosetta was a complex mission with a long and complex trajectory with several planetary and asteroid flybys, a long hibernation phase, a deep space journey, close rendezvous operations with a natural body (comet 67P) before eventually landing a robotic lander onto the surface of the comet. It was the first mission to achieve various milestones including landing on a comet.[2,8]

2.1 Mission Trajectory

The objective of the baseline trajectory for the Rosetta mission was to rendezvous with the comet 67P/Churyumov-Gerasimenko in August 2014. To achieve this, Rosetta would carry out a series of planetary flybys interspersed with Deep-Space Maneuvers (DSMs). The baseline trajectory was designed by considering, all relevant orbital perturbations, such as third body gravitational attractions and solar radiation pressures. One of the most important optimization parameters for the trajectory computation is the spacecraft mass after the comet rendezvous. The launch window is derived after the general trajectory is fixed: gravity assist conditions, DSMs, ephemeridis of celestial bodies, etc.

The baseline trajectory for the nominal launch date of Feb 26, 2004, can be deconstructed into the following phases:[1]

Launch: Ariane 5 was the selected launcher for Rosetta Mission. It will deliver the spacecraft into a heliocentric trajectory with an excess velocity of 3.545 km/s and asymptote declination of -2.0.

Earth-to-Earth: Lasted for 370 days and a Deep Space Maneuver (DSM-1) of about 173.5 m/s was performed.

First Earth Swing-by: The first Earth swing-by was performed at an altitude of 4287 km. No maneuvers were conducted except for navigation corrections when required. The relative approach and departure velocities were around 3.90 km/s.

From Earth to Mars: Lasted for 730 days. The spacecraft completed a full revolution around the Sun and prepared to swing-by around Mars during the second perihelion to aphelion arc with the DSM-2 of about 64.3 m/s performed near perihelion.

Swing-by of Mars: The purpose of the Mars swing-by was to impart the maximum deflection to the spacecraft velocity vector. Hence, a minimum distance of 200km at pericenter altitude was fixed. No maneuvers were performed during this phase. However, proximity to Mars' moons Phobos and Deimos affected navigation corrections during this swing-by. The relative approach and departure velocities were around 8.77 km/s.

From Mars to Earth: This phase lasted for about nine months, and no DSM was performed.

Second Earth swing-by: The spacecraft's altitude was about 13893 km from Earth' surface at the second Earth fly-by. No maneuvers were planned other than routine navigation corrections. The relative approach and departure velocities were around 9.33 km/s.

From Earth to Earth: Lasted for 727 days and the spacecraft was hibernated for a certain period of this time. The spacecraft completed another revolution around the Sun reaching distance up to 3.3 Au from the Earth. DSM-3 of about 129.4 m/s was planned at the aphelion of this arc.

Third Earth Swing-by: The purpose of the third Earth swing-by was to impart the maximum deflection to the spacecraft velocity vector at the closest possible altitude of 300 km. The relative approach and departure velocities were around 9.98 km/s.

From Earth to DSM: Cruise phase of 546 days with the spacecraft hibernated.

DSM to Enter Deep Space: Delta-V of 532.6 m/s was imparted at a distance of 4.4 AU from the Sun.

From DSM to Churyumov-Gerasimenko: Lasted for 1108 days during which the spacecraft was hibernated.

Churyumov-Gerasimenko orbit matching maneuver: With a mission duration of 3739 days (10.2 years), the spacecraft captured the comet, approaching it from behind with a braking-maneuver of 773.6 m/s.

Near Comet Operations: This phase lasted for 445 days after rendezvous with the comet and 4184 days (11.5 years) from the launch. It consisted of drifting with the comet while conducting the primary science experiments of the mission.

2.2 Rosetta Spacecraft

The Rosetta spacecraft design was based on a cuboid central frame with an aluminum honeycomb main platform and two solar panels extended from opposite sides of the cuboid. The exploded view is

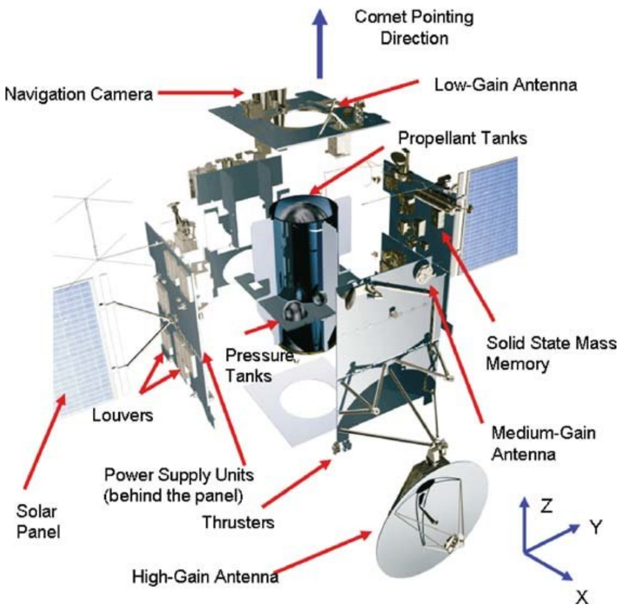

Fig. 1: Exploded view of the Rosetta spacecraft and its major components. Credit: ESA

seen in Figure 1 displaying primary subsystem components of the spacecraft.[10] It consists of three modules:

1. *Bus Support Module (BSM)* holds the spacecraft subsystems in the lower part.

2. *Payload Support Module (PSM)* houses the science equipment, instruments and sensors that face the comet during the operational phase of the mission. This module was mounted on top of the BSM.

3. *Lander Support Module (LSM)* hosts the Surface Science Package (SSP) named PHILAE lander.

The BSM and PSM together formed the Rosetta Orbiter spacecraft which conducted most of the mission and carried the Philae lander all the way to the Comet 67P. The lander Philae was released upon arrival at the comet and was the first spacecraft to land on a comet. For this study we focus primarily on the subsystems of the Rosetta orbiter as we assume the payload instruments remain unchanged. Further details addressing major subsystems of the Rosetta spacecraft are as follows.

2.2.1 Rosetta Orbiter Subsystems

Propulsion Subsystem: Rosetta's main propulsion system consisted of 24 paired bipropellant thrusters from which 12 thrusters were for the sake of redundancy. Each thruster could generate 10 N force for trajectory and attitude control. Mounted inside the vertical thrust tube were two large, equally sized, propellant tanks that were used to store the fuel and oxidizer. The top tank was filled with MMH (monomethyl hydrazine). The lower tank contained N2O4 (nitrogen tetroxide) that would chemically interact with MMH. Propellant pressurization was provided by two 68 L high pressure tanks filled with helium. The spacecraft carried 1720 kg of propellant at launch in which 660 kg of it was monomethyl hydrazine (MMH) fuel and 1060 kg nitrogen tetroxide oxidizer, contained in two 1108 liters, grade 5 titanium alloy tanks. The thrusters were able to provide delta-v of at least 2300 m/s over the course of the mission.[13]

Power Subsystem: Rosetta was the first deep space mission to rely exclusively on solar arrays for its electrical power generation. Two solar arrays spanning 32 m^2 produced the electrical power required by the spacecraft. Rosetta's electric power generation was based on a very high efficiency solar cells known as LILT (low intensity, low temperature) cells to generate sufficient power at solar distances of more than 5 AU (800 million km), where the sunlight was only four percent of the intensity at Earth. The cells were optimized for low solar intensity (40 W/m2), low temperature (-130°C) operation, with a sunlight conversion efficiency of 25 percent. In addition, rechargeable lithium-ion batteries were utilized as backup power source during periods of eclipse or darkness.

Avionic Subsystem: The Avionics Subsystem includes the Attitude and Orbit Control Monitor System (AOCMS) and OnBoard Data Handling (OBDH) subsystems. The AOCMS subsystem has interfaces with most of the other spacecraft subsystems (i.e. with the Solar Array and Telecommunications subsystems for pointing purposes, or with the Propulsion subsystem for the attitude control). The OBDH subsystem controls the telecommunications with the ground segment, managing the telecommand and telemetry traffic. The OBDH distributes the commands to the different subsystems and receive from them their respective status and telemetry. The spacecraft was 3-axis stabilized during normal operations, but during the prolonged hibernation period in deep space when most of its systems were shut down, it was placed in a spin stabilized configuration.

Communication Subsystem: Rosetta's communications suite included a steerable High Gain Antenna (HGA), a fixed position Medium Gain Antenna (MGA), and two omnidirectional Low Gain Antennas (LGA). S-band was used for telecommand (up-

Rosetta Spacecraft Subsystems	**Mass (kg)**
Philae Lander	100
Rosetta Orbiter	
Payload Support Module (PSM)	179.53
Bus Support Module (BSM)	
Attitude Determination and Control	121.58
Power	342.09
Propulsion	177.34
Structure and Mechanisms	277.59
Thermal Control	59.06
Communication	61.99
Data Handling	36.7
Dry Mass	**1355.88**
Propellant	1720
Total Mass	**3075.88**

Table 1: Mass budget for Rosetta

Event	**Delta-V (m/s)**	**Propellant (kg)**
Launch Window	10	9,28
First Correction	160	144,78
Cruise Navigation	75	65,48
DSM 1	173,5	145,89
DSM 2	64,3	52,15
DSM 3	129,4	101,91
Asteroid Steins Flyby	11	8,48
DSM 4	532,6	378,49
Asteroid Lutetia Flyby	11	7,22
Enter deep space	773,6	451,72
Contingencies	75	38,42
Near Comet Ops	120	59,67

Table 2: Delta-V budget for the main events of the Rosetta Mission

link) while the downlink was done using S and X-band channels. The communication equipment included a 28W X-band TWTA (Traveling Wave Tube Amplifier) and a dual 5 W S/X band transponder. Onboard heaters kept the equipment from freezing when the spacecraft was far from the Sun.

2.3 *Mass and Propellant Budgets*

In order to study the effect of ISRU on the Rosetta mission detailed evaluation of mass and propellant budget was conducted. A mass budget of the Rosetta spacecraft is developed in terms of major subsystems and is provided in Table 1. This will provide a baseline to compare the savings in the total mass of the Rosetta 2.0 spacecraft which will make use of the in-orbit refueling capabilities.

Delta-V Budget: The trajectory of the Rosetta spacecraft is a highly complex one with multiple planetary flybys. During the trajectory, Rosetta performed several maneuvering burns. The delta-V budget of the Rosetta mission is shown in Table 2. Major events corresponding to propulsive maneuvers are shown with the amount of fuel utilized for each maneuver. This data is further visualized in Figure 2. This information provides a reference of dynamically changing mass of the spacecraft throughout the trajectory and is used in developing the mission scenarios as explained in next section.

3. Working Methodology

The overall objective of this study to estimate benefits of in-space refueling for the Rosetta mission. The benefits could range from technical capability, reliability or cost perspective. It has been assumed that any resources required for the mission are available in a Lunar orbit. In respect with the Rosetta baseline, the capability to refuel a spacecraft in-space can either reduce the total launch mass (smaller launch vehicle), increase the scientific capability (more payload) or enhance the journey by reducing the journey time or the necessity to have a hibernation phase (increased reliability).
The study is conducted in two phases. Phase one of the study has already been completed while phase two is ongoing currently and is targeted to be completed by end of the current year 2021. The first phase of the study focused mainly on familiarization of the baseline Rosetta mission and spacecraft in addition to analyzing important aspects related to the in-orbit refueling application. The ongoing second phase is focused on developing new mission scenario to include in-space refueling and eventually resizing of Rosetta with refueling considerations. The final goal is to demonstrate significant savings in the total launch mass primarily driven by reduction in onboard propellant mass. This paper focuses on the second phase and results obtained thus far.

3.1 *Study Process*

The process developed for the study is a scenario-based analysis. Given the open ended nature of the project, several mission scenarios are identified to be analyzed. Each mission scenario is representative of a unique mission architecture. For the current study, a mission architecture has four main elements. They are; 1) Launch Vehicle, 2) Rosetta Spacecraft, 3) Propellant Depot, and 4) Mission Design. The first three are physical systems that interact with each other while the fourth element is how the three systems interact with each other. Mission Design

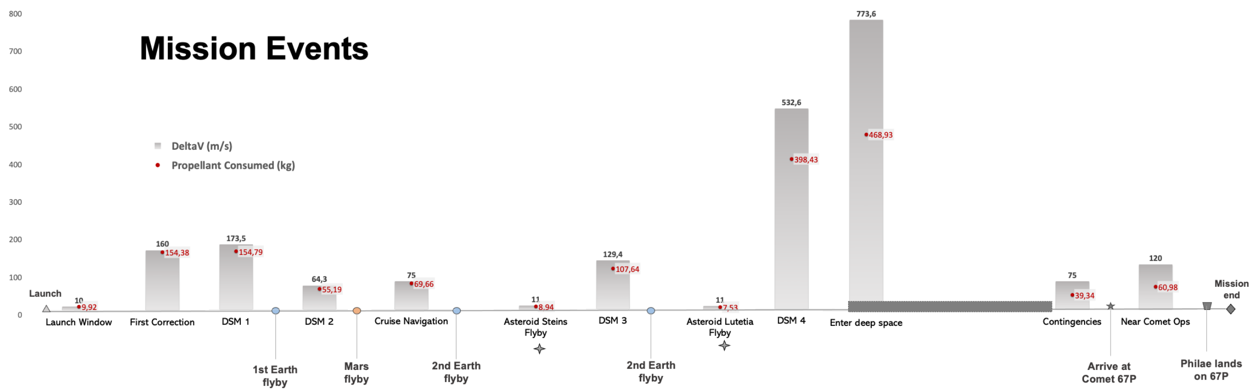

Fig. 2: Delta-V Events and Propellant consumed

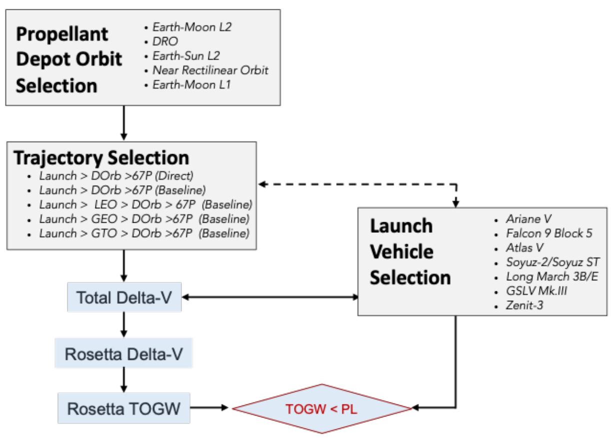

Fig. 3: CDF-LU Layout and Configuration

addresses questions such as: "*In what orbit will the propellant depot be located?*", "*How does Rosetta reach the depot?*", "*What is the role of the launch vehicle?*", "*How much delta-V is provided by the launcher and the Rosetta spacecraft and in what trajectory segments?*"

The process flow diagram shown in Fig.3 explains the working methodology developed for the study. The first step is the selection of an orbit for the propellant depot. Several options are identified as feasible candidates based on the literature review,[4],[3].[16] These options include Earth-Moon Lagrange points (EM-L1 and EM-L2), Earth-Sun Lagrange points (ES-L1 and ES-L2), Distant Retrograde Orbits (DRO), Near Rectilinear Orbits (NRO) et al. as shown in Fig.3. The location of the propellant depot determines rest of the trajectory and selection of the launch vehicle. Multiple options are considered for both of these decisions. Also, both decisions impact upon each other as well.
Trajectory analysis is conducted in two mission phases. The first mission phase addresses the trajectory from the Earth to the propellant depot while the second phase addresses trajectory analysis from propellant depot to Comet 67P.
Multiple launch vehicle options are identified as feasible options based on current available launchers of varying size and performances. Similarly, several ways are possible to reach the depot (direct launch to depot orbit or via a parking orbit) and later to the destination comet. Selection of trajectory and launch vehicle thus determines the total delta-V required to complete the mission and also how much delta-V is needed from the Rosetta spacecraft. The delta-V required from Rosetta will drive the total mass of Rosetta at the launch and the size of the propellant tanks of the spacecraft. The desirable goal would be to launch Rosetta with only enough fuel to reach the depot where it can be fully refueled and can reach its destination. It is important to consider that the mass and volume of Rosetta are under the payload carrying capacity of the launch vehicle.

3.2 *Mission Scenario Definition*

The study process shows how multiple combinations of launcher, propellant depot location and trajectory are possible. Each combination is a unique mission architecture and refers to a specific mission scenario. Figure 4 visualizes the mission scenario definition. A launch vehicle (LV) places the Rosetta Spacecraft (RS) in a parking orbit from where the Rosetta carries on to the propellant depot orbit. Upon arrival at the depot, the Rosetta refuels completely and continues to match the trajectory of the

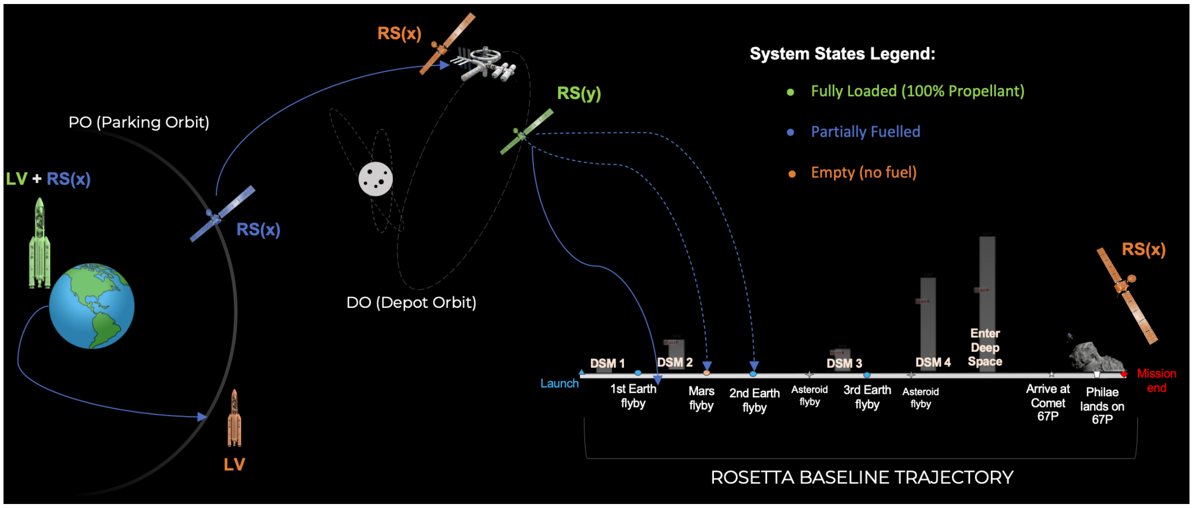

Fig. 4: Mission scenarios development for Rosetta 2.0

original Rosetta missions at one of the major trajectory events. From this point onwards, Rosetta continues to its destination as per the original trajectory profile. The original Rosetta performed flybys near two asteroids which were important scientific mission milestones. Thus, it is decided that post-refueling Rosetta spacecraft will meet the original trajectory before the first asteroid flyby.

As seen in Figure 4, the Rosetta spacecraft will be in different states from mass point of view throughout the trajectory. Initially at launch, Rosetta is carrying X amount of fuel and is represented as RS(x) in the figure. This X amount of fuel is what is required for Rosetta to leave the parking orbit and reach the propellant depot. Thus, Rosetta 2.0 will need to carry only this X amount of fuel (plus safe margin) at the launch. Rosetta can be completely refueled at the propellant depot, depart from the propellant depot and meet the trajectory of the original mission (Rosetta baseline) at a specified point and carry on the further path as the baseline Rosetta mission. The specific point at which the Rosetta 2.0 will join the baseline trajectory will be dependent on the delta-V required. This will determine the total fuel required to complete the rest of the mission and is represented as RS(Y) in Figure 4. RS(Y) also determines the size of the propellant tanks.

It should be noted that if RS(X) is less than RS(Y), then Rosetta 2.0 will be launched with partially full tanks, which can then be completely filled at the propellant depot. There is a potential problem with liquid sloshing during the launch if the tanks are too empty. This aspect is not addressed in the current analysis and will be considered in future analysis if needed.

3.3 *Tools*

the following software tools were used in implementing the methodology for the current study. They are described as follows:

Architecture Modeling Platform: A model based systems engineering approach is implemented to model the mission architecture and implement the study process described previously in this section. The Concurrent Design Facility at the University of Luxembourg (CDF-LU)[19] is applied for this purpose. RHEA Group's Concurrent Design Platform (CDP)[20] is used to develop the model of the main elements of the mission architecture. CDP is a concurrent design platform where different system elements can be modeled and assigned with parameters defining the characteristics and performance for each element. Model elements can be physical components and non-physical aspects such as trajectory or cost analysis.

Figure 5 shows the models developed in the CDP for the Rosetta Baseline and Rosetta 2.0 missions. As it can be seen, the baseline model is more detailed and includes components for each subsystem of the spacecraft. Every component is assigned with parameters such as the mass of the component, the power required, and the physical dimensions. The model for Rosetta 2.0 is more broad and contains elements like launcher and the propellant depot.

CDP is found to be an appropriate modeling tool for this study as it provides a platform where the parametric information flow between mission architecture elements can be established and alternative scenarios can be modeled. CDP also provides the capability to integrate alternatives within the model and conduct trade studies more effectively.

SPICE Toolbox: For this study, all trajectory computation routines were implemented in the SPICE Toolkit. SPICE was developed by the Navigation and Ancillary Information Facility, JPL, NASA. SPICE is an information library with a large collection of user-level APIs, subroutines and functions provided as source code to enable the user to compute the state of planets, satellites, comets, asteroids and spacecraft. It has a database of size, shape and orientation of planets, satellites, comets and asteroids. These observation geometry datasets called 'kernels' are sourced from reliable sources such as Mission Operations Centres and accompanied with additional metadata which are consistent with flight project data systems and SPICE standards.
The Spice Toolkit was originally implemented in ANSI FORTRAN 77 but is now available in many languages. For this paper, SPICE Toolkit version N0066, released April 10, 2017 is used. MATLAB R2020a environment is used to interact with the toolkit.

4. Trajectory Analysis

For the current study, refuelling the Rosetta spacecraft is considered beneficial if the following conditions are met:

1. the total launch mass of Rosetta 2.0 is less than the Rosetta Baseline spacecraft;
2. Rosetta 2.0 autonomously reaches the depot orbit;
3. Rosetta 2.0 rendezvous with the baseline trajectory at the same (or lower) delta-v cost of the portion of orbit that will be not performed in the Rosetta 2.0 misison.

The trajectory redesign of Rosetta is performed backward in time from a given point along its nominal trajectory. Considering that the depot draws resources from the Moon, it shall be bounded to the Earth-Moon system. The outgoing legs of Earth flybys are considered as points-of-interest for Rosetta 2.0 to rendezvous with the baseline trajectory. For this paper, only the first Earth flyby was evaluated.

The trajectory design of Rosetta can be split into three parts: the launch from the Earth surface to a given parking orbit, the transfer from the parking orbit to the depot, and finally the rendezvous trajectory from the depot to the outgoing leg of the original Rosetta trajectory resulting from the first Earth flyby.

While the launch is subjected to the gravitational attraction of the Earth only, the transfer and the rendezvous trajectories are modeled in the three-body dynamics of the Sun-Earth system. The interaction with the Moon gravity is ignored.

A description of the depot orbit initially described, followed by the rendezvous trajectory and the launch and transfer orbits.

4.1 *Propellant Depot Orbit*

The trajectory design begins with the selection of a depot orbit. In the literature, distant retrograde orbits were proposed as a possible candidate given their stability enabling inexpensive station-keeping,[5],[16].[3] Although their high costs of insertion, comparable to the injection into an interplanetary trajectory, and to escape prevent saving any delta-v for interplanetary probes and demand to explore other solutions. In the cis-lunar space and the low-energy regions surrounding the Sun-Earth L1 and L2, several orbits needs to be evaluated as potential candidates. For this study, a halo orbit at L2 was selected based on the reasoning that at low-energy the L2 becomes the access door to interplanetary space. The orbit selected as candidate to host the depot is a largo Lissajous orbit characterized by an orbital period of six months and by and an amplitude Ay-Az of 800'000 km.[12]

Fig. 11 shows in blue the depot orbit in the circular restricted three-body problem of the Sun-Earth system. The red and yellow dots identify the location of the Earth and L2 respectively.

4.2 *From depot to the Rosetta baseline trajectory*

The rendezvous with the original trajectory of Rosetta baseline is formulated as an optimisation problem. The proposed method, summarised in Eq. 1, is standard in trajectory design and consists of splitting the trajectory in two. A delta-v is applied at the extrema and the trajectories are propagated forward from the depot and backward from the exit condition at the Sphere of Influence, increased to two Hill sphere radii.[22] The optimizer imposes that the two trajectories (forward and backward) merge at end of the integrations and minimizes the total delta-v computed as the sum of ones given at the extrema and the velocity difference at the contact point.

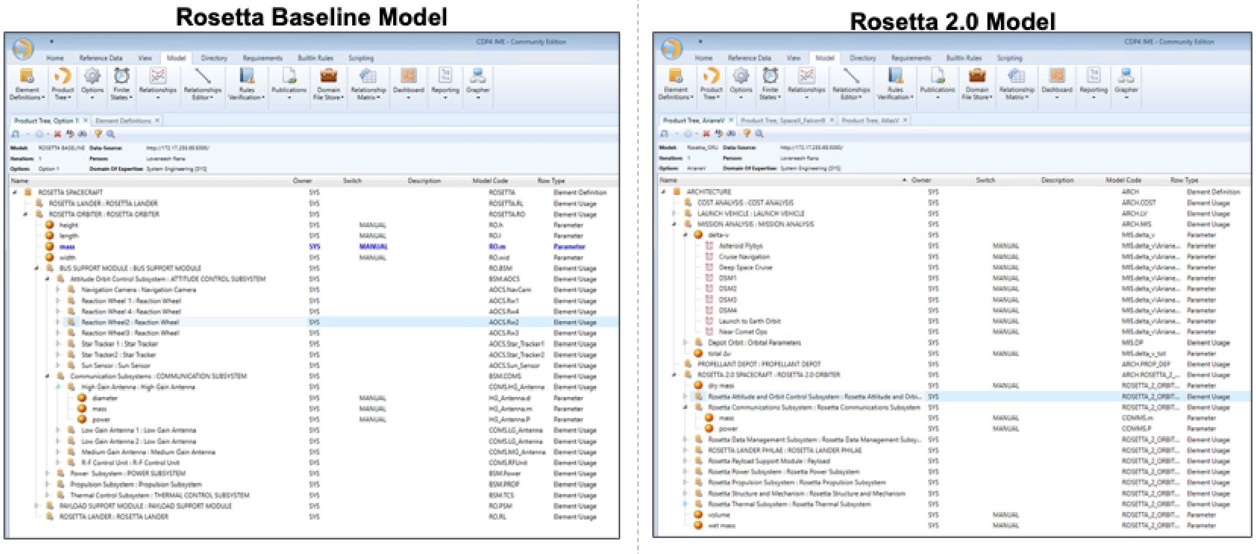

Fig. 5: Mission Architecture Model developed for Rosetta Baseline and Rosetta 2.0

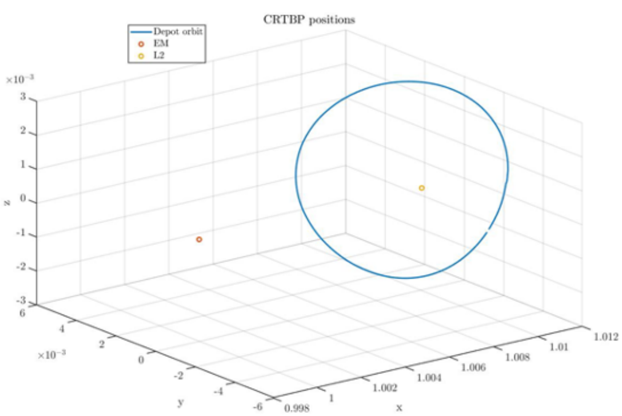

Fig. 6: Propellant Depot Trajectory around Earth-Sun L2

$$\begin{aligned} \min_{x} \quad & \Delta v^- + |\mathbf{v}^-(t^-) - \mathbf{v}^+(t^+)| + \Delta v^+ \\ \text{subject to} \quad & \mathbf{r}^-(t^-) = \mathbf{r}^+(t^+) \end{aligned} \quad [1]$$

where $\mathbf{r}(t)$ and $\mathbf{v}(t)$ are three dimensional values of the Cartesian position and velocity of Rosetta propagated for a the time t, Δv denotes the delta-v at one extrema and the $\mp$ signs indicate the propagation direction, backward or forward respectively. The delta-v at the extrema are aligned with the orbital velocity.

The total delta-v for the rendezvous trajectory is limited to 333.5 m/s. This value comes from summing up the first correction with the first deep space maneuver. Matching the original position and velocity of Rosetta at the Sphere-of-Influence (SoI) within this premise is simply impossible given the different three-body energy of the depot and outgoing leg, thus a similar approach is repeated rendezvousing Rosetta's nominal trajectory at the close approach, the perigee of the Earth flyby.

Figure 7 shows the SoI of Earth in green, the trajectory of Rosetta leaving the depot in blue, and rendezvous at the closest approach of the first Earth flyby of the baseline mission in dashed lines.

If the total delta-v is smaller/equal than the selected value, Rosetta will be able to transfer from the depot to the comet 67P following its nominal trajectory.

Table 3 collects some feasible solutions that allows to rendezvous with the original trajectory of Rosetta

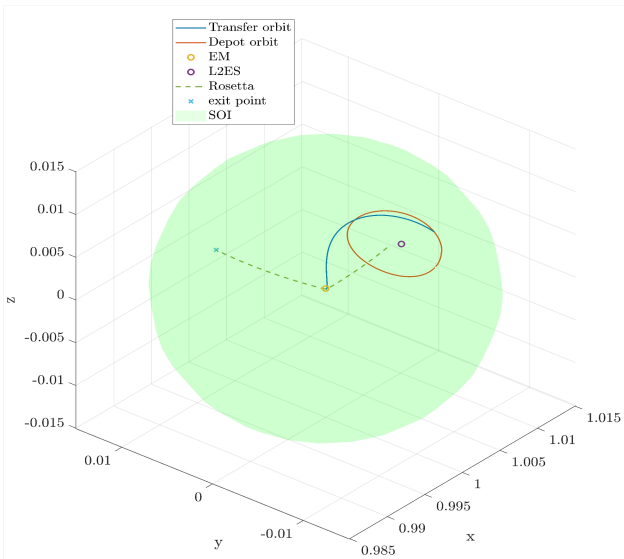

Fig. 7: Trajectory from Propellant Depot to Baseline mission trajectory

θ_H, deg	$\Delta v, m/s$	$\Delta t, days$
357	112	65.238
358	138	63.541
359	121	62.169
360	127	60.721

Table 3: Rendezvous trajectory solutions

at the periapsis of the first flyby with a total delta-v smaller than the maximum acceptable one. From left to right you find the angular position at the depature from the depot, $\theta_H = \tan 2(y_H, z_H)$, the total delta-v and time of flight.

Since there is no/insignificant advantage in delta-v and time-of-flight, rendezvousing with the nominal Rosetta trajectory represents a simple check that Rosetta will be able to reach 67P without consuming additional propellant. The benefit of on-orbit servicing comes from the possibility to select a smaller launcher to insert Rosetta in a parking orbit and from limiting its propellant mass to the level required for Rosetta to transfer to the depot where it will be refilled. A trade-off is performed on the altitude of the parking orbit. The objective is to minimise the launch cost understanding how this variable impacts from one side the amount of propellant that Rosetta shall burn to transfer to the depot and from the other side the payload of the launcher, accounting for the wet mass of Rosetta.

4.3 From the Earth surface to the depot

The launcher trajectory is approximated with an Hohmann transfer leaving the surface of the Earth at 0 km/s and injecting Rosetta into a circular parking orbit.

The possible parking orbits' inclination are limited to a range of 30 degrees with respect to the ecliptic plane corresponding to the maximum oscillation for Kourou launch site, induced by precession of the Earth rotation axis during the year.

Modelling the launch trajectory as an Hohmann transfer, as shown in Eq. 2, from the surface to Earth to the parking orbit underestimates the delta-v by $2km/s$ induced by gravity losses, associated to the misalignment of the thrust vector to the velocity one and to the atmospheric drag. Although this error appears large, it can be eliminated by considering the delta-v difference.

$$\Delta v_H = \sqrt{\frac{R+h}{R}\frac{2\mu}{2R+h}} + \sqrt{\frac{\mu}{R+h}} - \sqrt{\frac{R}{R+h}\frac{2\mu}{2R+h}} \quad [2]$$

where μ, R and h are respectively the standard gravitational parameters of the Earth, its radius and the altitude of the parking orbit, and the first and last term account for the velocity of the Hohmann at the peri-/apo- gee respectively, while the second term is the velocity of the circular orbit of radius $R{+}h$. There is no contribution at the perigee, located on the surface of the Earth, since the launcher is assumed to be

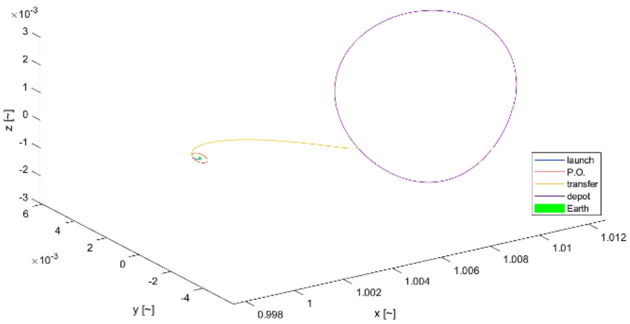

Fig. 8: Trajectory from Earth surface to depot orbit

accelerated from zero velocity.

Finally the transfer is obtained by back-propagating the trajectory from the depot, applying a small delta-v Δv_H, and collecting those that impact the spherical segment of parking orbit radius bounded to $\pm(R+h)\cos i_{MAX}$, where i_{MAX} corresponds to the $30deg$ inclination.

Fig. 8, the launcher and transfer trajectories are represented in blue and yellow respectively. The parking and depot orbits are displayed in red and purple instead.

Among all possible impacting solutions, the stored ones are those belonging to the pareto front of the minimum $\Delta v - \Delta t$, defined in Eq. 3:

$$\Delta v = \Delta v_H + \left| \mathbf{v}(\Delta t^-) - \frac{\mathbf{h}(\Delta t^-)}{h(\Delta t^-)} \times \frac{\mathbf{r}(\Delta t^-)}{R+h} \sqrt{\frac{\mu}{R+h}} \right| \quad [3]$$

$$\Delta t^- : \quad r(\Delta t^-) = R + h \quad \& \quad \left\| \frac{z(\Delta t^-)}{R+h} \right\| < \cos i_{MAX}$$

where h is the angular momentum of the impacting orbit. The pareto front is obtained from a grid search on the angular position, θ_H, and the insertion burn at the Halo orbit, Δv_H.

Fig. 9 collects the pareto fronts obtained for parking orbits ranging from $7500km$ to $90000km$ altitude and limited to 140 days. Lower altitudes are not reachable considering the amount of propellant stored on Rosetta. It is interesting to notice that for increasing the time-of-flight (tof) ensures savings in delta-v, even if the delta-v reaches a plateau at around 80 days. Increasing the altitude induces a shift in the pareto front, meaning lower delta-v values for the same values of tof.

Fig. 10 shows the difference in delta-v coming from targeting larger parking orbit. The delta-v of the

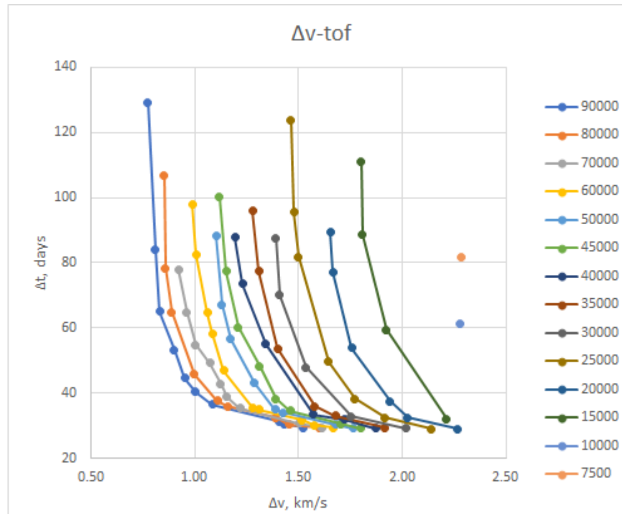

Fig. 9: Trade off between the altitude of the P.O

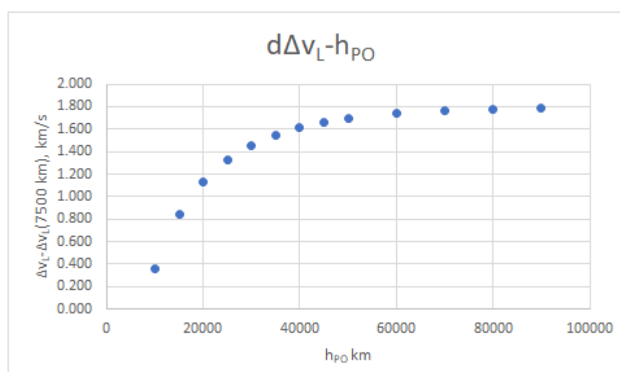

Fig. 10: Delta-v difference originating by different altitude of the parking orbit

launchers computing from Eq. 2 are subtracted from the value obtained for smallest altitude, $7,500km$. It is interesting to notice that from $60,000km$ altitude, the delta-v difference settles down to $1.8km/s$. A reference value of insertion at $7,500km$ altitude parking orbit can estimated to $12.0 - 12.4km/s$ accounting for gravity loss.

5. ISRU Considerations

This section outlines ISRU-related aspects that were investigated and considered in context of the current study. These include details regarding the propellant depot design, rendezvous and docking operations, and refueling procedures.

5.1 *Propellant Depot*

In-orbit servicing and refueling and propellant depot concepts have been addressed in multiple programs and studies in the past.[6,7,11,23] An exhaustive literature review was conducted focusing on programs and studies addressing in-orbit refueling and propellant depot concepts. Following this review, a propellant depot concept is developed for the Rosetta spacecraft based on the information gathered. The concept is hypothesized by selecting components of other studies and is not verified through a parametric feasibility analysis as that would be out of scope for the current study.

The depot is assumed to be a large-scale spacecraft capable of storing a substantial amount of propellant. Since Rosetta does not use cryogenic propellant, no additional specifications regarding cryogenic fuels storage are considered. The depot would be a passive participant in the rendezvous and docking process, acting as the target spacecraft with Rosetta as the chaser.

The scalable propellant depot proposed in[21] is used as a reference, as it offers a baseline design compatible with a variety of mission requirements and functions. It is shown in Figure 11.

The propellant depot has a cylindrical body with a planar solar array on either side. It has a dock and fluid transfer interface on one end. The outer shell of the depot is a debris shield to keep the tanks from being punctured. The tanks and most of the other subsystems (pumps, feed lines, avionics, etc.) reside within the debris shield. The propellant depot will consist of the following components:

- The robotic arm
- Flexible cable
- Tool storage area
- Tanks for the propellant and oxidizer

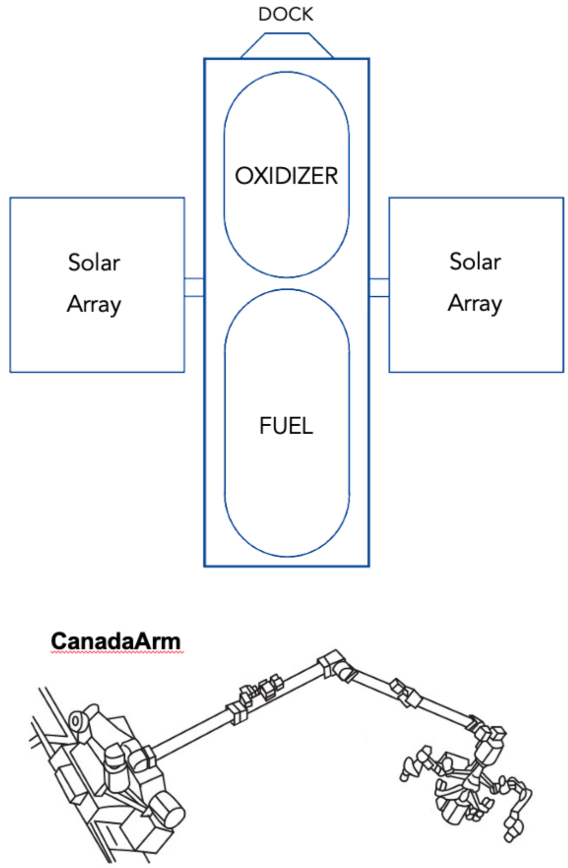

Fig. 11: Propellant Depot design is based on the scalable design concept[21] with a robotic arm similar to CanadaArm2 on the ISS

- Thermal control subsystem
- Attitude Control subsystem
- Communication subsystem
- Power subsystem (solar panels and batteries)
- Boom structure for the docking

It is assumed that the depot is refillable and regularly resupplied with propellant from the Moon. This is an important consideration that drive the decision to consider the propellant depot's location in the vicinity of the Moon.

The propellant tanks are cylindrical with hemispherical end caps. A variety of materials can be chosen for the tanks. The fluid transfer interface and dock are assumed to be similar to the international berthing and docking mechanism.

The Canadarm2 robotic arm[17] and its "hand", Dextre, are used as a reference for how the propellant depot's robotic arm will work and be structured, considering they were used to successfully perform a fuel transfer operation similar to the one described above on the ISS.

5.2 *Rendezvous and Docking*

An important part of the in-orbit refueling is the rendezvous and docking between the propellant depot and the spacecraft. In order to define the rendezvous and docking scenario for the Rosetta ISRU concept, a literature review was conducted addressing these concepts.

For the rendezvous and docking operations, Rosetta 2.0 spacecraft is the chaser, and the propellant depot the target. The propellant depot has a boom extension that can connect with the Rosetta 2.0 spacecraft to so that the docking and fluid transfer operations can be conducted.

When Rosetta is at the appropriate distance and orientation with regard to the depot, the depot extends the boom, which is attached to the cone structure mechanism on Rosetta. This is done considering the design reasons that it would be easier to add a conical structure to Rosetta (and most spacecraft) rather than an extendable probe. Since Rosetta is the chaser, it can use a high-precision optical guidance mechanism to orient itself in a way that the depot's boom attaches to its cone during the docking procedure. For the soft-docking process, Rosetta's cone will have a magnetic structure, and the flexible rope which attaches itself for it will be from a highly ferromagnetic metallic alloy. The common stages of the rendezvous and docking operations are as follows:

1. Rendezvous Stage/Phasing (insertion orbit to 40km)
2. Proximity (40km to 100m)
3. Terminal (100m to Docking Stage)
4. Docking Stage (1m to Mechanical Capture)
a. Soft-Dock (Capture/Latch)
b. Hard-Dock (Rigidization of two spacecrafts)

The rendezvous, and docking process adopted for Rosetta 2.0 is defined in four major steps which are demonstrated in Figure12 and described next in further detail:

1. Station-Keeping: Using inputs from the docking sensor, an autonomous guidance and navigation system guides Rosetta to a station-keeping position, maintaining approximately 1m separation between it and the propellant depot, with an orientation such that Rosetta's cone faces the boom on the depot.

2. Soft-Docking: The soft-dock operation begins with the depot extending a flexible cable toward Rosetta. At approximately $\frac{1}{2}$ m separation, the boom is extended until the cable latches at the bottom of the cone on Rosetta. The two spacecraft are now soft-docked.

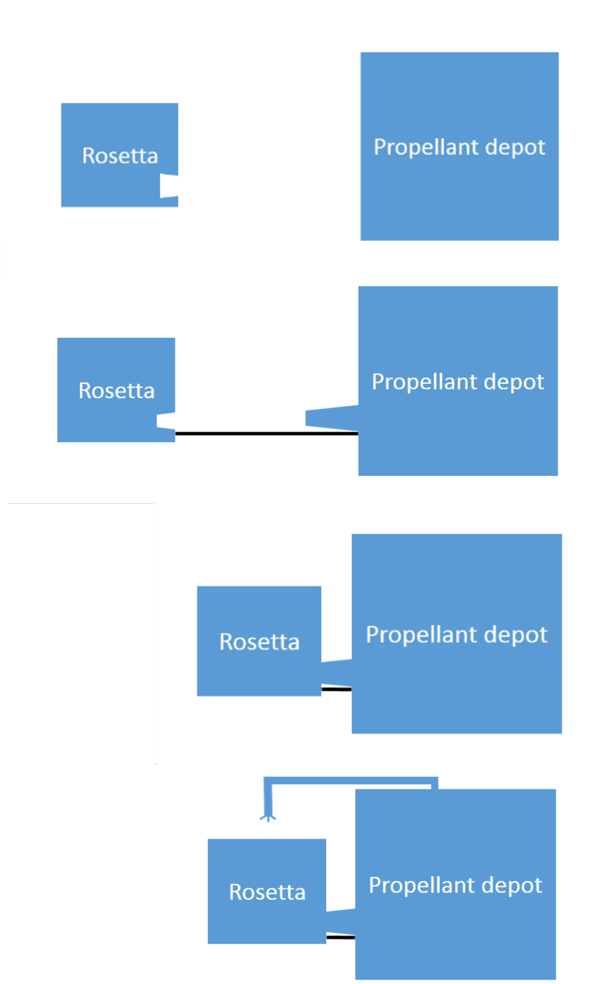

Fig. 12: Rendezvous and Docking Process

3. Hard-Docking: Once the probe is latched on Rosetta's cone, the cable retracts, pulling the boom into contact with the cone. When the probe head is fully seated within the cone, Rosetta and the fuel depot are docked.

4. Servicing: The two spacecraft are now rigidly docked and aligned closely enough to allow fluid and electrical connections to be mated with the propellant depot. The robotic arm is extended from the propellant depot (will be explained below), and fuel transfer can begin. The refueling mechanism and process will be described below.

5.3 *Refueling*

NASA's Robotic Refueling Mission (RRM)[18] is used as a reference for the refueling operations at the propellant depot. RRM is selected due to being relatively simple, as well as for having been successfully demonstrated with experiments on the International Space Station. It was also the first in-orbit refueling demonstration that used representative satellite hardware; that is, it successfully demonstrated fuel transfer operations on satellites that were not designed for refueling. Therefore, the process demonstrated in the RRM phase 1 is considered a suitable and reliable choice.
The main fuel transfer involves a robotic arm that is part of the propellant depot. The arm is assumed to be capable of reaching into a storage area of the depot where tools for various stages of the refueling process are stored, grabbing them, and deploying the tools as required for refueling operations. The tools are similar to those used for the RRM, as follows:

- Wire Cutter and Blanket Manipulation Tool
- Multifunction Tool
- Safety Cap Tool
- Nozzle Tool

5.4 *Hardware changes for Rosetta 2.0*

Some necessary changes would have to be made to the Rosetta spacecraft to make it compatible with the ISRU operations described in the previous section.

The addition of an autonomous guidance and navigation system: Rosetta has an automatic GNC system, but it was not capable of the accuracy required by the docking procedure. Since the newly required accuracy is of the order of less than 1-meter, optical solutions will have to be considered. Other supporting sensors and equipment will be required as well.

The addition of a cone for compatibility: The refueling process considered for this study relies on a probe–cone docking system. Rosetta would have a cone placed on it in such a location that the fuel depot would use a probe on its end to attach to it in an orientation allowing the robotic arms to perform operations on the fuel tanks. An end-effector design proposed for RRM mission[15] is considered as the reference for the cone mechanism on the Rosetta spacecraft.

6. Results

Based on the trajectory analysis described in the previous section a complete mission scenario and corresponding architecture is modeled. The results addressing the propellant required, delta-V, and the mass budget for Rosetta 2.0 are presented next.

Event	**DeltaV (m/s)**
DO-Baseline	350
DSM 2	64,3
DSM 3	129,4
Asteroid Steins Flyby	11
DSM 4	532,6
Asteroid Lutetia Flyby	11
Enter deep space	773,6
Contingencies	75
Near Comet Ops	120
RS(y) Delta-V	**2066,9**
Propellant Mass (kg)	**1221.45**

Table 4: Delta-V budget for Rosetta 2.0 Fully Loaded Configuration RS(y)

6.1 *Rosetta 2.0 Delta-V Budget*

As discussed in the trajectory analysis, a backward propagation approach has been applied to compute the delta-V requirements for different steps of the mission. The second phase of the mission, i.e. the trajectory segment after refueling at the propellant depot is computed first. The Rosetta 2.0 spacecraft conducting this mission segment is referred to as RS(y) (see Figure 4), where y refers to the propellant mass required for this segment.

RS(y) - The Fully Loaded Configuration: Rosetta 2.0 spacecraft after completing the refueling departs the propellant depot and connects the trajectory of the baseline mission at the point of the first Earth flyby. This maneuver requires 350 m/s delta-V and from this point onwards, Rosetta 2.0 carries the mission in the same manner as the Rosetta baseline. This configuration is referred to as RS(y) in the Figure 4 and represents the fully-loaded configuration for the spacecraft. The total delta-V budget and the corresponding propellant for the RS(y) configuration is shown in Table 4. The total propellant mass for this loaded configuration is 1220 kg and includes margin for reserved fuel in addition to contingencies. Comparing to the Rosetta Baseline, this is approximately 500 kg less. It should be noted that 1220 kg is the mass of the propellant that Rosetta 2.0 spacecraft should have when it departs the propellant depot so that it can reach Comet 67P. This propellant mass determines the size of the propellant tank for the Rosetta 2.0 spacecraft.

RS(x) - The Launch Configuration: The first mis-

Delta-V(m/s)	**Propellant RS(x)**
1600	812.61
1500	749.46
1400	**688.20**
1300	628.77
1200	571.12
1100	515.19

Table 5: Delta-V options for Rosetta 2.0 Launch Configuration RS(x)

sion segment addresses the journey from the Earth to the Propellant Depot in the selected Earth-Sun L2 orbit. This involves a launch vehicle placing the Rosetta 2.0 spacecraft in a parking orbit from where the Rosetta 2.0 spacecraft will use its own propulsion system to reach the propellant depot orbit and dock at the station to start refueling. This configuration is represented as RS(x) in the Figure 4 and refers to the configuration of the Rosetta 2.0 spacecraft at the launch.

Since RS(x) depends primarily on the parking orbit, multiple orbit options were evaluated in the trajectory analysis ranging from altitudes of 7500 km to 90,000 km around the Earth. These results are discussed previously in section and shown in Figure 9. The parking orbits ranging from 30,000 km to 45,000 km are identified as options-of-interest for the this paper. Figure 13 shows these options-of-interest with delta-V and time-of-flight (tof) requirements to the propellant depot. The 35,000 km parking orbit is a typical GEO option. Table 5 shows propellant required for Rosetta 2.0 spacecraft in its RS(x) or the launch configuration for various delta-V requirements. A 1400 m/s delta-V configuration is selected for this paper which also includes propellant required for rendezvous and docking operations at the depot. The delta-V budget for RS(x) configuration is shown in Table 6 where 100 m/s is reserved for the rendezvous operations at the depot. This value is higher

Event	**Delta-V (m/s)**
Earth to PO	Provided by LV
PO to DO	1300
At Depot	100
RS(x) Delta-V	**1400**
Propellant Mass (kg)	**688.2**

Table 6: Delta-V budget for Rosetta 2.0 Launch Configuration RS(x)

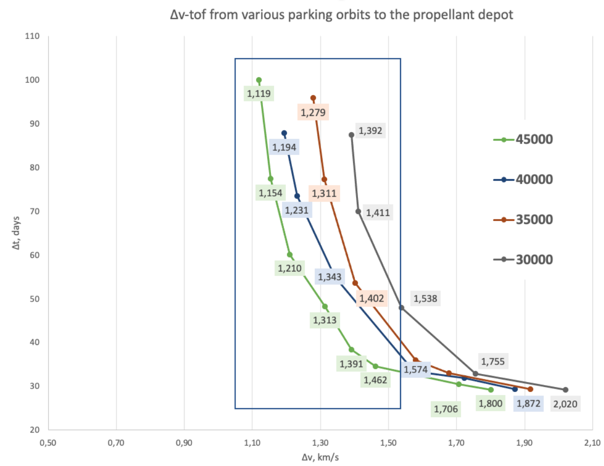

Fig. 13: Trade off between the altitude of the P.O

than average for such operations and is taken so to include the reserved propellant and extra margin. As can be seen in Figure 13, the 1400 m/s delta-V can be obtained from all parking orbits with varying time-of-flight. The propellant mass for the RS(x) configuration for 1400 m/s delta-V is 688.20 kg. This is the propellant mass that Rosetta 2.0 must carry at launch to reach the propellant depot from the parking orbit.

6.2 *Mission Architecture*

A complete mission architecture is now presented based on the RS(x) and RS(y) configurations of the Rosetta 2.0 spacecraft. The modeled mission scenario is shown in Figure 14 and additional details are provided in Table 7.

Parking Orbit Selection: A parking orbit of 35,000 km is selected from the options-of-interest identified in Figure 13. For a delta-V of 1400 m/s, the corresponding time-of-flight is 53 days. This option is a typical GEO option and is within the capability of several launch vehicles.

Launch Vehicle Selection: In order to reach the GEO parking orbit at 35,000 km, several launchers are considered. The primary benefit for the refueling scenario would be to use a smaller launch vehicle than the Ariane V launcher that was used to launch the Rosetta Basleine mission. The delta-V required for the launcher to place the Rosetta 2.0 spacecraft in the 35,000 km parking orbit is 11.9 km/sec. Several heavy-lift and medium-lift launchers are found capable to do so. The launcher options that were an-

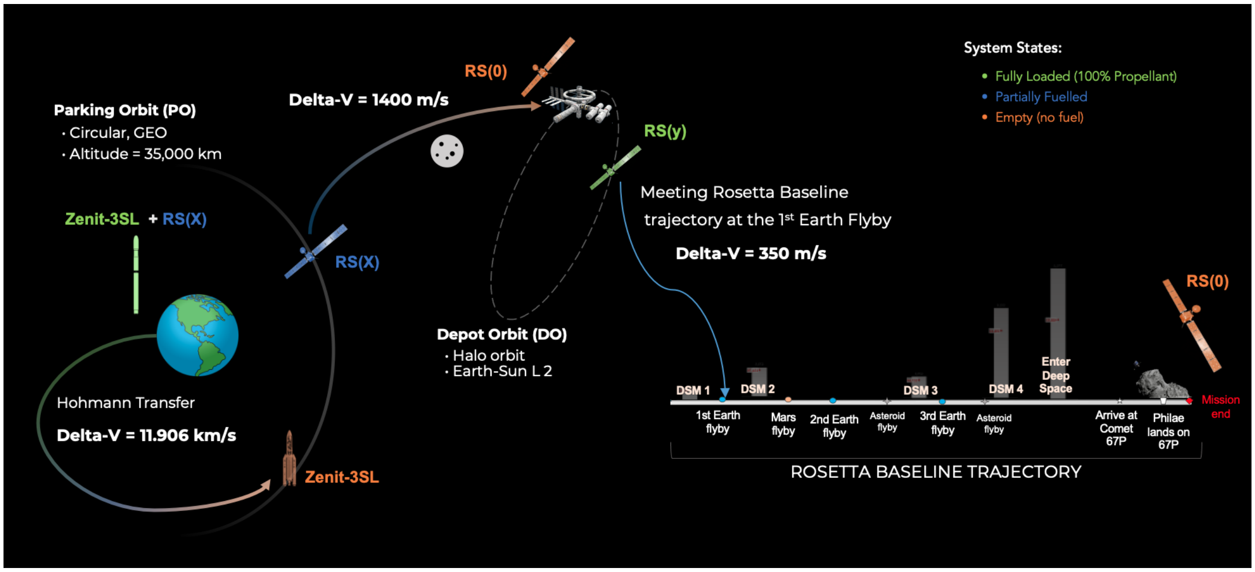

Fig. 14: Mission architecture scenario modeled for Rosetta 2.0

Launch Vehicle	Zenit 3 SL
Parking Orbit(PO)	Circular Orbit Altitude = 35,000 km tof = 53 days
Depot Orbit(DO)	Halo orbit at L2 Sun-Earth(+Moon) system Orbital period = 6 months Ay-Az Amplitude = 800'000 km

Table 7: Rosetta 2.0 mission architecture details

alyzed and found capable to conduct the mission are as follows:

- *Heavy-Lift Launchers*: Ariane V, Falcon 9 FT, Soyuz FG, and Atlas V.

- *Medium-Lift*: Zenit 3 SL, JAXA H-IIA 204.

All these options provide the capability to place a payload mass greater than 3,000 kg in a GEO parking orbit with a total delta-V of more than 12 km/s.

Yuzhnoye Zenit 3-SL is selected as the launch vehicle for Rosetta 2.0 Mission based on its sea-based launch flexibility. Zenit 3-SL is a medium lift launch vehicle that is considerably smaller than the Ariane V that was used for the Rosetta Baseline mission.

Mission Profile: The launcher Zenit-3SL, launches from Kourou carrying a partially filled Rosetta 2.0 spacecraft, RS(x), and conducts a hohmann transfer to place the Rosetta 2.0 spacecraft in the GEO parking orbit at 35,000 km. The Rosetta 2.0 spacecraft then carries out the rest of the mission using its own propulsion system. First, Rosetta conducts the maneuver to de-orbit from parking orbit and reach the propellant depot in the Earth-Sun L2 orbit. The delta-V needed for this segment is 1400 m/s. Rosetta 2.0 arrives at the propellant depot, docks, and refuels its empty tanks. The size of the propellant tanks is determined by the configuration RS(y) as shown in Figure 14. The spacecraft once completely fueled, departs the propellant depot and meets the baseline trajectory at the first earth flyby. The delta-V computed for this step is 350 m/s. From this point forward, the Rosetta 2.0 spacecraft will travel along the same trajectory as the Rosetta baseline spacecraft. In comparison with the baseline mission, the Rosetta 2.0 thus completes all the delta-V maneuvers after the DSM 1 maneuver listed in Table 2. This includes DSM 2, DSM 3, DSM 4, Asteroid Flybys, maneuver to enter the deep space, and near comet operations.

6.3 Rosetta 2.0

Based on the mission scenario and trajectory analysis results, the delta-V for the Rosetta 2.0 spacecraft has been developed in the previous section, see Table 4 and Table 4.

Total propellant corresponding to the RS(x) configuration is found to be 688 kg. This includes the rendezvous operations at the depot and reserved fuel margin.

Total fuel corresponding to the RS(y) configuration is found to be 1220 kg and includes reserved and contingency margins.

Propellant mass is the primary driving factor. The Rosetta Baseline spacecraft carried 1720 kg of the propellant. In comparison, the RS(y) configuration of Rosetta 2.0 requires 1220 kg of the propellant mass. As mentioned earlier, this configuration is the fully fueled spacecraft at the propellant depot. This difference of 500 kg of propellant storage between Rosetta Baseline and Rosetta 2.0 RS(y) would also scale down the overall size and mass of the propellant tanks. This reduction in structural mass is not considered for the current paper and total mass of the Rosetta 2.0 is assumed same as the Rosetta Baseline spacecraft. The reason for this decision is based on the fact that Rosetta 2.0 would need to perform rendezvous and docking operations at the depot. These operations were not part of the baseline mission and thus would require some additional hardware components on the Rosetta 2.0 spacecraft that will increase the spacecraft dry mass. Thus, the decrease in structural mass due to smaller propellant tanks would offset the additional mass.

In summary, the Rosetta 2.0 spacecraft will have propellant tanks large enough to store 1220 kg of propellant (from RS(y)) but at the launch will be loaded with 688 kg of propellant (RS(x)). The mass breakdown is shown in Table 8 comparing the Rosetta Baseline spacecraft with the two configurations of the Rosetta 2.0 spacecraft (RS(x) at the launch and RS(y) as completely loaded at the depot).

Mass Savings: As can be seen from the results, the launch configuration, RS(x) shows 34% reduction in the total mass in comparison with the Rosetta Baseline spacecraft.

Cost Savings: Koelle[14] provides a cost estimate of $30,000 per KG of payload mass to GEO for Zenit 3SL launch vehicle. Using this figure, the 34% mass reduction from Rosetta Baseline amounts to approximately 31 Million US Dllars in 2001. Adjusting with an inflation rate of 55% from 2001 to 2021,[9] this figure increases to 48 Million US Dollars.

7. Conclusion

The paper presents results of a mission and architecture design study. The goal of the study was to demonstrate the benefits of in-space refueling capabilities using an ESA legacy mission, the Rosetta, as a reference. The process developed and applied for the study is a scenario-based approach. Multiple mission scenarios are found to be potentially feasible. The analysis conducted for the paper developed a complete mission scenario that includes multiple space elements. A propellant depot station is modeled in an orbit around the Earth-Sun L2 liberation point. A smaller launch vehicle (than the original Rosetta mission) has been shown as a valid choice for the mission. The mission architecture incorporates considerations regarding the propellant depot design, and rendezvous, docking and refueling operations.

A modified and smaller Rosetta 2.0 spacecraft is derived from the design of the Rosetta Baseline mission. An overall 34% reduction in the total mass has been achieved, which translates to approximately 48 Millions USD in cost savings from launch mass reference.

The study acts as a template for the future interplanetary missions. The final results demonstrates how establishing an in-space refueling capability and in-situ resource utilization can significantly reduce the overall mass and cost of the future spacecrafts and missions.

7.1 Future Work

The current results show one mission scenario for the redesigned mission. As mentioned in the methodology section, there are other possible scenarios and trade options where additional benefits could be found. The study will continue to explore these new mission scenarios in the future work.

From trajectory design perspective, the next objectives are taking into account the arrival and departure positions in the depot orbit. Other initial depot orbits' candidates will also be analysed and the launcher trajectory will be studied with a higher fidelity model. Moreover, the possibility of reconnecting with Rosetta's trajectory after the 2nd and 3rd Earth flyby will be investigated as well in the future work. In addition to the mission design and trajectory analysis, future work will also address a more detailed system design study addressing spacecraft subsystems analysis and further improve the design of the propellant depot.

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	Rosetta Baseline	**Rosetta 2.0 - RS(x)** *Launch Configuration*	**Rosetta 2.0 - RS(y)** *Loaded Configuration*
Propellant Mass	1720	688	1220
Dry Mass	1355	1355	1355
Total	**3075**	**2043**	**2575**
Mass Reduction (%)		**34**%	**16**%
Cost Savings		**48 Million USD**	

Table 8: Mass comparison of Rosetta Baseline and Rosetta 2.0 spacecraft

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