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SESAME OPENS: A PRECURSOR TO HUMAN ASTEROID MISSIONS

Volker Maiwald
German Aerospace Center (DLR), Institute of Space Systems, Department of System Analysis Space Segment, Bremen, Germany, Volker.Maiwald@dlr.de

Emilien Marchand
TU Delft, Faculty of Aerospace Engineering, Chair of Systems Engineering, Delft, Netherlands, Emilien.Marchand@gmail.com

A common path for human spaceflight is currently often envisioned to lead to Near Earth Asteroids (NEA) within the next decades. While the goal is clear this is not so true for the targets. Just as unmanned probes investigated the lunar surface before humans ever set foot on our natural satellite, it is advisable – considering the current lack of knowledge about NEAs even mandatory – to send an unmanned mission ahead to conduct measurements in preparation of a human mission. Just as with the moon more than one target area should be investigated, i.e. more than one target asteroid.

While many designs for the actual human mission already exist, scenarios for a precursor mission are scarcer. In this paper we present a feasible design for a multi-rendezvous mission to targets suitable for human missions, able to reach up to 5 asteroids with one launch. We will propose a system that will be able to measure various properties of each asteroid, e.g. chemical composition and topography and describe it on a subsystem level, providing mass and power budgets for the whole system. The results show that a spacecraft of about 1,600 kg launch mass and utilizing solar electric propulsion can fly a 5 target mission within 10 years. With a sensitivity analysis we will show the robustness of the design and generally establish the feasibility of such a mission.

I. INTRODUCTION

At the latest since the results from the Augustine Commission interest in a human mission towards a Near Earth Asteroid (NEA) (a large proportion of the Near Earth Objects (NEO)) has increased and such an undertaking is considered as a further step in human exploration of the solar system in a so called "flexible path" [1].

Furthermore there have been several non-crewed missions to NEAs, e.g. Hayabusa or NEAR Shoemaker.

A profound analysis about the possibility to use an Orion-based spacecraft for a NEA-mission has been presented previously in [2], other mission designs involve e.g. a European-based approach deriving from Columbus life-support system heritage and ARIANE technology [3].

Essential to the preparation of a NEA mission with a human crew however is the question of: "What information about NEOs is still needed to support a robust, sustainable human exploration program?" [4]

To allow finding the necessary information, a precursor mission has been investigated at the German Aerospace Center's (DLR) Institute of Space Systems in 2012. Analogously to e.g. the Surveyor-program as a "fact-finder" for the Apollo missions, investigating the landing sites [5], the authors present a mission design in this paper that allows the investigation of multiple targets for a human NEA mission. To reduce costs and effort one spacecraft is intended for multiple rendezvous with NEAs.

The basic idea has been to use existing, very well optimized trajectories from ESA's Global Trajectory Optimization Competition (GTOC) [6] for application on such a multi-rendezvous mission, basically designing a spacecraft "around" such a mission profile.

After re-calculating the trajectory and through a detailed design of a spacecraft assembly, a mission involving several landers has been set-up. The results for this mission, labeled the Spacecraft for Evaluation of Suitability of Asteroids for Manned Exploration (SESAME) [7], are presented in this paper.

Based on the idea of designing a precursor for a human asteroid rendezvous mission, the following tasks and questions have been formulated as guideline for the study:

- Is it possible to apply existing low-thrust trajectories for multi-rendezvous/ flyby asteroid missions (e.g. from GTOC) to an actual spacecraft system design for an asteroid precursor mission?
- What are the advantages of a multi-target precursor mission, which scientific goals can be gained by such a mission in difference to a dedicated single target precursor? What is a suitable science payload?

- What design properties does the spacecraft for such a mission (based on a GTOC IV trajectory) have with special regard to a possible lander, the propulsion subsystem, the power subsystem and the communication subsystem – are there other significant subsystems that are required for such a mission?
- What are special design considerations originating from the asteroid environment (e.g. wrt orbit stability, dust, illumination characteristics)?

II. REQUIREMENTS AND SCIENCE PAYLOAD

The major objective for SESAME is collecting information necessary to determine a suitable NEA target for a human rendezvous mission. To support this goal a number of requirements have been set up, described in the following subsections along with the scientific instruments to meet the science questions raised by a precursor mission.

Suitability is to be regarded based on three major criteria:

- Accessibility: Is the target accessible from an orbit mechanics view (e.g. in a flight time not exceeding health limitations) and from a proximity operations point of view (e.g. a small rotation period prevents these).
- Safety: Proximity operations should not risk the crew (e.g. due to the presence of satellite asteroids).
- Scientific Interest: It must be scientifically relevant enough to justify a human crewed mission

II.I Mission Requirements

The requirements are relevant in the categories described before, i.e. the mission shall be able to determine accessibility and safety relevant issues and also answer first scientific questions about the possible rendezvous targets, also required to effectively plan a human mission to that small body. Some requirements fall into more than one category, e.g. the ability to investigate the surface topography is relevant for the planning of a future mission but generally is also of scientific value. Mostly the requirements are payload related and dictate the scientific abilities of the spacecraft.

The list in Table 1 only shows the primary requirements, necessary for a full success. Minor ones can be found in [7].

Requirement No.	Description
1	The spacecraft shall have the ability to assess environmentally related hazards of the target (e.g. presence of a satellite body or debris)
2	The spacecraft shall determine size and shape of a target body
3	The spacecraft shall measure the gravity field of a target body
4	The spacecraft shall be able to determine the rotational motion of a target body (i.e. rotation rate and rotational axis)
5	The spacecraft shall determine regolith characteristics, composition and density of a target body (for determination of surface disturbance responses)
6	The spacecraft shall be able to determine the internal structure of a target body
7	The spacecraft shall be able to determine the topography (surface shape and features) of a target body
8	The position and velocity of a target body between precursor rendezvous and crew rendezvous shall be known with enough accuracy to ensure crew mission safety
9	The spacecraft shall be able to transmit all science data to Earth

Table 1: SESAME's mission requirements.

II.II Payload

The payload for SESAME consists of two major groups, one for the carrier vehicle, which transfers to each target body of the mission and the payload for the lander vehicle(s), which individually investigate one target body. The payload for the carrier vehicle is more global in nature and the primary mission objectives are mostly subjects for the carrier vehicle.

Both payload groups are shortly summarized in Tables 2 and 3. The carrier vehicle has a total payload mass of 33 kg at a power demand of about 100 W.

The lander vehicle's payload has a mass of 4.3 kg at a power demand of around 53 W. Its payload is mostly derived from the Pico Autonomous Near-Earth Asteroid In Situ Characterizer (PANIC) lander concept [8], the remaining instruments are derived from DLR's Philae lander on Rosetta.

It shall be noted that these values already include margins between 5 and 20% according to ESA ECSS

Instrument/ Parameter	Multi-band Imager	Wide Angle Camera	Laser Alti-meter	X-ray Spectro.	NIR* Spectro.	Magneto-meter	Radio Tomo-graphy	Radio Science	Total
# of unit(s)	1	2	2	1	1	1	1	-	9
Mission Requirement	1,2,4,5,7	1,2,4,5,7	3,5,7	5	5	secondary	5,6	3,6,8	
Reference	Hayabusa	Hayabusa	Hayabusa	Hayabusa	Hayabusa	NEAR	NEASR	-	-
Margin [%]	10	5	20	10	10	10	20	-	-
Mass [kg]	6.7	1.9	8.9	3.6	1.7	1.7	8.9	-	33.4
Average Power [W]	17.9	Included in multi-band	26.4	16.5	11	1.7	28.8	-	102.3

Table 2: Summary of SESAME's carrier payload. *NIR means Near –Infrared

Instrument/ Parameter	Alpha Particle X-Ray Spectro.	NIR-Spectro.	Micro-scope Imager	Stereo Camera	Rosetta Lander Imaging System	Dust Impact Monitor	Beacon	Low Gain Antenna	Total
Topic of Analysis	elementa abun-dances	Surface Tempera-ture, space weather-ing, minera-logy	Grain size distribu-tion and space weather-ing	Imaging of surface	Descent and close-up camera	Surface distur-bances	Tracing for orbit deter-mination of target body	Tracing for orbit deter-mination of target body	-
Reference	PANIC	PANIC	PANIC	PANIC	PHILAE	PHILAE	Alcatel	Compass	-
Margin [%]	10	10	10	10	10	10	5	5	-
Mass [kg]	0.3	0.7	0.2	0.3	0.6	0.4	1.4	0.3	4.2
Average Power [W]	1.1	8	2.7	3.6	2.2	0.2	35	-	52.8

Table 3: Summary of SESAME's lander payload.

standard depending on the development status of the respective component. The cameras and laser altimeter are also used for navigation of the carrier spacecraft.

In addition to mere scientific instruments the lander payload also includes a package that is basically a transmitter (beacon and antenna), which will continue its transmission after the carrier left the target body to allow a very precise tracking of the asteroid to comply with the mission requirement 8.

III. SPACECRAFT LAYOUT AND BUDGETS

Currently only a very draft version of the layout exists and is presented in Fig. 1. The main part of the spacecraft is the bus, holding all major subsystems. At its nadir side, the landers are attached, one for each target, i.e. currently five are envisioned. They are released from there and positioned in a way that power generation and simultaneous contact with Earth is still possible. The solar array has a size of 36 m² with two panels.

The following Subsections describe the budgets for the lander and the carrier vehicle. More details about the subsystems follow in the next section.

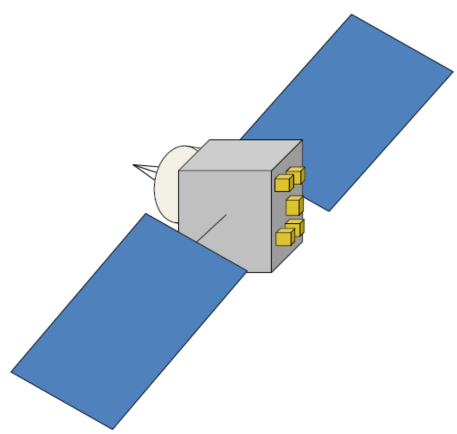

Fig. 1: Conceptual design of SESAME to show its different constituents: yellow – lander, blue – solar array, light grey – high gain antenna, grey – main bus.

III.I Carrier Mass and Power Budget

Mass Budget

The payload is currently only a small contributor to the system mass (33 kg), the major payload of the carrier is the number of landers carried along (currently 5).

Propulsion (dry) is the largest proportion of the mass, closely followed by the power subsystem, mainly due to the solar arrays needed for powering the electrical thrusters.

Subsystems	Mass [kg]
Payload	33
Power	204.7
GNC	96.8
Communication	49.9
Data Handling	25
Thermal Control	16.4
Harness	60
Structure	150
Propulsion	222
Subtotal	850
+ System Margin (20%)	Included in subsystem values
Xenon propellant	451
Chemical propellant	125
Total	1,433

Table 5: SESAME Carrier mass budget.

Power Budget

The power budget for the carrier vehicle is listed in Tab. 4. Major drivers are the elaborate GNC, the communication and the payload. The not very distant location of the targets related to the Sun leads to a moderate power demand by the Thermal Control System.

Subsystems	Power Demand [W]
Payload	103
Guidance, Navigation and Control	101
Communication	114
Data Handling	30
Thermal Control	42
Power	12
Subtotal	402
+ System Margin (20%)	80
Propulsion	6,200
Total	6,682

Table 4: SESAME Carrier power budget.

The typical ESA standard system margin adds another 20% (not counting in the propulsion power demand, as this is electrical propulsion).

At this stage power demands for valves and comparable small components are not yet considered.

In total and under assumption on only one running electrical thrusters but on full throttle, the total power demand for the system is about 6,700 W.

III.II Lander Mass and Power Budget

Mass Budget

The lander has a total mass of about 28 kg, where one of the largest contributors is the scientific payload (4.25 kg, including the beacon). Another 500 g is added by the transmitter, antenna. The power system, mainly the secondary battery is the other major driver for the lander mass. Smaller, but still significant is the landing gear (3.4 kg).

A summary of the mass budget is given in Tab. 5.

Subsystems	Mass [kg]
Payload	4.2
Communication	0.5
Power	4.9
Data Handling	0.8
Structure	3
Mechanism (Landing Gear)	3.4
GNC	2.7
Harness	2
Thermal Control	1.5
Subtotal	23
+ System Margin (20%)	4.6
Total	27.6

Table 5: SESAME Lander mass budget.

Power Budget

Subsystems	Power Demand [W]
Payload	17.8
Communication	2.2 (35 for pulse)
Data Handling	2.5
Thermal Control	4
+ System Margin (20%)	included already
Total	26.5 (59.3)

Table 6: SESAME Lander power budget.

The power budget of the lander is dominated by the instruments and also by the pulse function of the communication subsystem, intended to improve the tracking capability of the target asteroid.

III.III Launch Mass and Summary

To determine the launch mass of the mission, the total mass of the spacecraft has to be combined with the individual masses of the five landers (27.6 kg). Therefore the total launch mass is 1571 kg, which is below the mass Soyuz Fregat can bring into orbit (1600 kg) and escape from Earth.

IV. SUBSYSTEM DESCRIPTIONS

The following sections shortly describe the individual subsystem components quickly to provide an overview over the actual design.

IV.I Data Handling System

The Data Handling system is headed by a RAD6000 processor, has one main board and has a mass memory of 8 Gb for storing science data mostly.

IV.II Electrical Power Supply System

Due to the closeness of the mission to the Sun, solar power generators are used as primary source of power consisting of Triple Junction Cells with an efficiency of 29.5%. A total size of 36 m² is used divided consisting of two wings with each 5 panels.

Li-ion batteries provide power during eclipse phases and for safe mode.

A power conditioning and distribution unit is foreseen as well.

IV.III Thermal Control System

The Thermal Control system has to keep the spacecraft in an operational state. The solar distance for the current mission time varies between 1.25 and 0.785 AU, which is not a major limitation.

The spacecraft has a radiator of approx.. 1m² size and is packed (60%) in 15 layers of MLI. No active elements are currently foreseen for thermal control.

IV.IV Guidance, Navigation and Control System

The GNC subsystem is designed to handle two major cases: the cruise phase, i.e. transfer to the target body and the proximity operations at that body.

It has 8 thrusters with 10 N of thrust force (+8 more for redundancy). There are twos tar trackers and 8 sun sensors as well as an inertial measurement unit and 4 reaction wheels. The cameras and LIDAR which are part of the payload can also be used for navigation purposes.

Currently MON/ MMH is considered as chemical propellant for the GNC, which also contains one tank.

IV.V Propulsion

Evaluating various thrusters, the RIT-22 thrusters have been selected for this spacecraft design [9]. Three thrusters are taken along, 2 are needed for the whole throughput of the fuel and one is included for redundancy. Each thrusters also contains a power processing unit. Also a tank is included for the design and gimbals.

IV.VI Communication

SESAME uses KA band for the communication with Earth during normal operations applying a dish antenna of 1 m size. For back-up an X-band transceiver and two low gain antennas are included as well. Communication between lander and carrier is achieved with S-band and UHF. In this case a 20 cm parabolic antenna is used.

IV.VII Lander

The lander is designed to operate five years after being dropped on the target body to allow sufficient time for the preparations and arrival of the human crew (the idea is to determine the target's trajectory as precise as possible until the crewed vehicle arrives). It is powered by two sets of Li-ion batteries (one for descent/ measuring and one for beacon operations). During beacon operation the battery is recharged with a small solar panel of 20 cm x 20 cm.

For attitude control during descent the lander is equipped with a flywheel. Due to lack of knowledge regarding surface structure and strength no harpoon or anchoring mechanism for the connecting the lander with the asteroid surface is selected, but a set of small thrusters is used to remove any velocity after impact on the surface. A landing gear will also absorb energy from the impact to prevent damage.

VI. MISSION ANALYSIS

VI.I Launcher

To have moderate launch costs Soyuz Fregat has been selected as a launcher for the mission and also used for the subsequent mission analysis. A launch mass of 1600 kg has therefore been assumed as maximum.

VI.II Target Selection and GTOC Evaluation

Regarding the mission analysis one basic initial idea had been to reuse trajectories calculated for the Global Trajectory Optimization Competition (GTOC) [6]. Reason for this was that the trajectories calculated for GTOC 5 have involved multiple rendezvous with NEAs already and we were curious to know whether or not the competition results would also prove useful for an actual mission. Making use of the vast optimization skills gathered in this competition seemed prudent.

As a first step the targets of the various participants of the competition have been investigated upon compliance with various lists of targets for human missions [10-15]. Overall the list of possible targets consisted of about 200 small bodies.

After only an insufficient number of matches between GTOC trajectory targets and this list has been found, the authors decided to redo the mission analysis and optimize a new trajectory with the new targets as rendezvous partners.

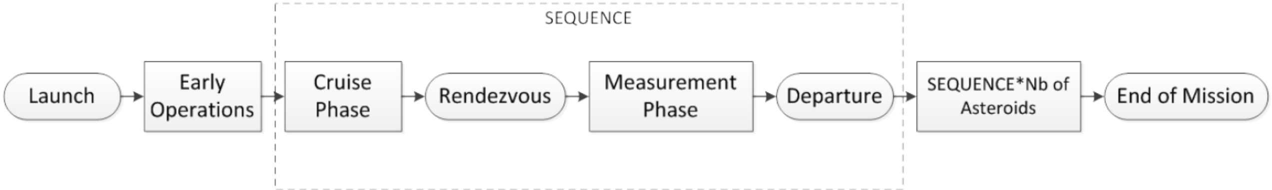

Fig. 2: Mission sequence for SESAME.

VI.II Trajectory Optimization

The trajectory optimization has been conducted in the following steps:

1. Creation of a database of feasible transfers using impulse based approximation
2. A beam search of this database
3. Compute propagation of best branches
4. Creating the output of the trajectories and verification of the results

At the beginning it is assumed that the probe is at Earth's position and applying an optimal (in terms of minimal Δv) Lambert transfer (with variations of launch and flight time). Step by step the transfer is extended, each step's results are stored for the later beam search. Pairs where a possible trajectory is not likely are sorted out with the help of the Q-law by Petropoulos [15].

Each leg's launch window starts with the previous leg's arrival times.

Property	Value
Departure Date from Earth	21 March 2023
Arrival @ 5^{th} target	26 February 2030
Total Δv	16.6 km/s
Wet mass	1,571 kg
Xenon fuel mass	451 kg
Bi-propellant mass	125 kg

Table 7: Mission parameters of SESAME

Target Body Data	Absolute Magnitude (H)	Orbit Condition Code (OCC)	Observation opportunities prior launch
$2001QJ_{142}$	23.4	6	2012
$2000SG_{344}$	24.8	2	none
$2009OS_{5}$	23.6	5	2014-2020
2007YF	24.8	5	2021
$1999AO_{10}$	23.9	6	2018, 2026

Table 8: Mission target data

Each transfer between targets is divided into a number of segments with a thrust impulse in each segment's middle. Between the impulses Keplerian orbit mechanics are assumed. The number of segments is varied between 2, 3, 4 and 6, in sequence, until a transfer is regarded unfeasible or 6 segments are reached.

For each leg a Δv is determined and those are added for each trajectory. From this, via the rocket equation, the spacecraft mass can be determined. If the mass exceeds the mission limits, it is regarded unfeasible.

Once the database is generated it is subject to a beam search [16]. This method allows investigating several paths in parallel. Heuristic methods are used to drop solutions which are not usable. [16]

Each beam receives an evaluation based on the trajectory's flight time, the mass and the number of asteroids visited.

In the case of this search, a maximum number of 1600 is selected for the search breadth.

The branches with the best evaluation are then propagated more thoroughly. Each segment of the respective trajectory is divided once more into subsegements, where one is a thrust phase, the other one a coast phase. Using an evolutionary algorithm called Covariance Matrix Adaption Evolution Strategy (CMA-ES) [17], the trajectories are optimized regarding fuel mass. Input parameters are the thrust angles and durations as well as the launch date and the overall flight duration. The initial values are the values obtained for an impulsive transfer, stored in the database.

The spacecraft parameters used for the optimization can be found in Tab. 7, along with the launch and arrival dates.

VI.III Mission Sequence and Summary

The sequence for SESAME's mission can be found in Fig. 2.

It consists of the launch and early operation phases after which the cruise to the first target is conducted, ending with the rendezvous. This is succeeded by the scientific measurements, including the lander operation, before the spacecraft departs. Beginning with the cruise phase until the departure of the spacecraft, this part of the sequence is repeated until all targets have been reached.

Body	Arrival Date	Departure Date	Time of Flight (d)	Duration of Stay (d)
Earth	-	21 Mar 2023	316	-
2001QJ$_{142}$	31 Jan 2024	22 Jul 2024	444	173
2000SG$_{344}$	9 Oct 2025	8 Mar 2026	384	150
2009OS$_{5}$	27 Mar 2027	28 Sep 2027	445	185
2007YF	16 Dec 2028	19 May 2029	283	154
1999AO$_{10}$	26 Feb 2030	4 Sep 2030	314	190

Table 9: Trajectory data.

Depending on the status of the spacecraft subsequent to the measurements of the last target the mission ends.

With the current mission analysis 5 targets can be reached by SESAME. Some values can be found in Tab. 8, which describe the target bodies. The absolute magnitudes are not very large yet also not very small. Sizes derived from these are mostly about 50-100 m. For several targets the orbit condition code is large (9 being the maximum possible), i.e. their orbit data is rather unknown, however for most targets observations are planned before the precursor mission launches.

A summary of the mission is found in Tab. 9. It can be seen that for three instances the flight duration exceeds one year, whereas the dwelling time at the respective targets always is significantly larger than 100 days, the minimum being 150 days, i.e. approx.. 5 months.

The total duration of the mission is 2,724 days (ca. 7.5 years). The total flight time is 1,872 days, the total time spent at the target bodies is 852. Considering the flight time (which can still be used for scientific measurements) this means a ratio of 31% of the whole mission time is spent at the target bodies.

The trajectory results from the optimization have been checked with the help of the software Satellite Tool Kit, which only produced differences of the target body orbits in a range of some kilometres and differences in velocities of some meters per second.

VII. OPEN ISSUES

The authors present a preliminary design of the SESAME spacecraft and mission, which treats the feasibility of such an undertaking. Obviously the whole design needs refinement and further considerations.

Especially the trajectory has been calculated with a rough model, more thorough investigation is needed to find a final trajectory. Considering that over time more possible targets might be found, refining the trajectory will be an on-going process during the further design of a SESAME-like mission.

Furthermore the targets right now have been selected mostly for accessibility reasons, no per-se scientific evaluation has been conducted. From the list of likely targets certain small bodies might be more interesting from a science point of view than others, this would have to be evaluated by scientists of the planetology domain.

Another major issue is the accommodation and complete configuration. Right now only a rough draft of the configuration has been set-up, mostly driven by considerations regarding communication (with Earth and between Carrier and Lander) and the dropping of the lander as well as engine and solar array pointing. There is no internal accommodation of components and consequently there is a lot of room for optimization.

Considering the timeframe of the precursor mission, i.e. a launch in 2023 and final mission in 2030, this is rather late for a precursor preparing a human mission to a NEA. Nonetheless the authors regard this a realistic time frame as the current development for crewed spaceflight systems is rather slow. The first crewed mission with the new Orion vehicle will be launched in 2019 to 2021, targeted at a lunar orbit [18]. This will also mark the second flight of a Space Launch System (SLS). The first flight of SLS will be in 2017 with an unscrewed vehicle [18]. There are currently no further testflights planned in the time between 2017 and 2021, which the authors find peculiar, especially in comparison with e.g. the Apollo programme, where several unscrewed testflights preceded the crewed launch of Apollo 7.

There are no detailed plans yet for a NEA mission timeline, but considering that a NEA mission would still require significant enhancement of launcher and capsule (where currently no funding is appointed for) and also regarding the speed at which launches for SLS take place (four years between two launches), the authors regard it unlikely that SLS and Orion would be ready for an asteroid mission before 2030.

SLS development and mission road map is the most recent and most detailed one for a launcher of this size. With respect to previous studies [3] about a human mission to a NEA it is unlikely that a NEA mission can be conducted without an SLS-like vehicle (or even larger), i.e. if SLS is not ready before 2030 for a NEA mission the authors do not assume another mission (e.g. solely European) will be available at an earlier point of time.

Therefore the late timeframe for the precursor mission, while only an example with the currently given asteroid data, does not make the proposed mission obsolete.

A more detailed description of all mission aspects, including e.g. a communication scheme, can be found in [7].

VIII. CONCLUSION

This paper shows the feasibility of a multi-rendezvous solar electric propulsion precursor mission preparing a crewed rendezvous with a NEA. It has been shown that with a moderate launch mass, capable to be handled by e.g. Soyuz Fregat, a mission reaching five targets can be achieved. The multi-rendezvous approach prevents the risk of "showstoppers" and allows to gather scientific data aside from only preparing a crewed mission – all data gathered about an asteroid finally not selected for a crewed mission is on top. Generally the multi-rendezvous approach increases the success rate of finding a suitable target at a moderate effort (compared to launching individual missions to each target).

The design includes a preliminary instrument suite for both the carrier and lander vehicles, likely to be modified for actual implementation.

While the design parameters of GTOC have not been violated, the actual trajectories did not prove useful for a precursor mission due to non-compliance with human asteroid target lists. However the methods used by GTOC haven been employed for finding a new trajectory.

Overall SESAME or another precursor mission will likely be unavoidable in case of a human NEA rendezvous. This paper shows how such a precursor mission could work successfully.

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REFERENCES

[1] NASA; *Summary Report of the Review of US Human Space Flight Plans Commitee*; http://www.nasa.gov/pdf/384767main_SUMMARY%20REPORT%20-%20FINAL.pdf, accessed 5 July 2012

[2] Abell, P., Korsmeyer, R., Landis, R., Jones, T., Adamo, D., Morrison, D., & Lemke, L. (2009, December); *Scientific Exploration of Near-Earth Objects Via the Orion Crew Explroation Vehicle*; Meteoritics et Planetary Science, 44, 1825-36

[3] Maiwald, V., Quantius, D., Löscher, M., Dumont, E.; *Proposition of a European Mission Architecture for Crewed Missions to NEAs*; 63rd International Astronautical Congress, 1-5 October 2012, Naples, Italy

[4] Barbee, B. W.; *Target NEO: Open Global Community NEO Workshop Report.* Washington, D.C., 2011

[5] Lunar and Planetary Institute; *The Surveyor Program*; http://www.lpi.usra.edu/lunar/missions/surveyor/; accessed 5 July 2013

[6] ESA; *GTOC 5 – Penetrators*; http://sophia.estec.esa.int/gtoc_portal/?page_id=25; accessed 5 July 2013

[7] Marchand, E.; *A Precursor for a Crewed Mission to Near-Earth Asteroids: the SESAME Concept*; Master Thesis, Technical University of Delft, Faculty of Aerospace Engineering, Chair of Space Systems Engineering, 2012

[8] Schindler, K., Thomas, C., Reddy, V., Weber, A, Gruska, S., Fasoulas, S.; *PANIC – A surface science package for the in situ characterization of a near-Earth asteroid.* Acta Astronautica, 68 (11-12), 2011, p. 1800-1810

[9] Loeb, H.W., Schartner, K., Seboldt, W., Dachwald, B.; *CONSEP Contribution to Solar-Electric Propulsion*, DLR, Bonn, 2007

[10] Landis, R.R., Korsemeyer, D.J., Abell, P.A., Adamo, D. R.; *A Piloted Orion Flight To A Near-Earth Object: A Feasibility Study;* American Institue Of Aeronautics and Astronautics, 2004

[11] Adamo, D.R., Abell, P.A., Giorgini, J.D.; *Asteroid destinations Accessible for Human Exploration: A preliminary Survey in Mid-2009*; Journal of Spacecraft And Rockets, Vol. 47, No. 6, Nov.-Dec. 2010, pp. 994-1002

[12] NASA, *Near-Earth Object Human Space Flight Accessible Targets Study*, http://neo.jpl.nasa.gov/nhats/, accessed 19 July 2013

[13] Landau, D., Strange, N.; *Near-earth asteroids accessible to human exploration with high power electric propulsion*; Workshop Caltech Space Challenge, 2011

[14] Zimmer, A.K., Messerschmid, E.; *Going Beyond: Target Selection and Mission Analysis of Human Exploration Missions to Near-Earth Asteroids*; Acta Astronautica 69, pp. 1096-1109

[15] Petropoulos, A.E.; *Refinement to the Q-law for Low-Thrust Orbit Transfers*; 15th AAS/ AIAA Spaceflight Mechanics Meeting, Copper Mountain, Colorado, January 23-27, 2005

[16] Furcy, D., Koenig, S.; *Limited Discrepancy Beam Search*; Proceedings of 19th International Joint Conference on Artificial Intelligence, pp. 125-131, 2005

[17] Hansen, N., Kern, S.; *Evaluating the CMA Evolution Strategy on Multimodal Test Functions;* 8th International Conference on Parallel Problem Solving from Nature PPSN VIII, Proceedings, pp. 282-291

[18] Bergin, C.; *Acronyms to Ascent – SLS managers create development milestone roadmap,* http://www.nasaspaceflight.com/2012/02/acronyms-ascent-sls-managers-create-developmental-milestone-roadmap/; accessed on 24 July 2013