

Design Implications for Aerocapture Systems Placing Flagship-class Uranus Orbiters

Andrew Gomez-Delrio¹ and Soumyo Dutta²
NASA Langley Research Center, Hampton, VA 23681, USA

Aerocapture has the capability to deliver spacecraft to Ice Giant orbits while increasing on-orbit payload mass. Due to the aerodynamic forces and heat loads encountered during an aerocapture maneuver, the spacecraft must be contained by an entry vehicle, such as a typical rigid aeroshell used for planetary entry. This paper provides a preliminary design concept for an aerocapture system capable of delivering a flagship-class orbiter and planetary probe to Uranus orbit using heritage flight systems. The constraints of packaging large subsystems (radioisotope power systems, high gain antennas) and their effect on final aerocapture system design will be discussed. Mass properties will be presented for the system, with mass risks evaluated by margin policy implemented using standard mass properties control.

I. Nomenclature

<i>CBE</i>	=	mass current best estimate
<i>MEV</i>	=	mass maximum expected value
<i>MGA</i>	=	mass growth allowance
<i>MPV</i>	=	mass maximum permissible value

II. Introduction

The Ice Giant outer planets, which consist of Neptune and Uranus, are currently strong candidates for NASA's next flagship-class mission, with Uranus of particular interest [1]. A Uranus Orbiter and Probe (UOP) design concept has previously been presented as part of a mission concept study for a flagship-class Uranus mission [2]. With a time of flight to Uranus of 13 years, the spacecraft is baselined for launch opportunities starting in early 2030's. Upon arrival, a propulsive maneuver is performed to decelerate the spacecraft from high interplanetary speeds and into the desired final science orbit.

Aerocapture maneuvers utilize trajectories within a planet's atmosphere to provide aerodynamic drag and decelerate a spacecraft, achieving a captured orbit upon atmospheric exit [3]. Aerocapture orbit insertion reduces propellant and propulsion system demands when compared to fully-propulsive maneuvers [4]. In addition, an aerocapture maneuver can reduce interplanetary transit time by 2-5 years (15-30%) and overall flagship-class mission duration and cost can be reduced [4]. With entry vehicle systems providing a heritage basis for delivering flagship systems, such as the Mars Science Laboratory (Curiosity) and Mars 2020 (Perseverance), flight proven technologies can be leveraged to provide an aerocapture solution for a flagship-class mission.

A. Uranus Orbiter and Probe

The Uranus Orbiter and Probe spacecraft design was detailed in Ref. [2]. As shown in Figure 1, the system includes an instrument payload for the science tour of the orbiter, in addition to a planetary probe with heatshield designed for a Uranus entry trajectory. A 3.1 m high gain antenna (HGA) is utilized to satisfy communication requirements between Uranus and Earth. Since solar power generation cannot meet power requirements at Uranus' solar distance, three Radioisotope Thermoelectric Generators (RTG's) are required. A large primary structure houses two large propellant tanks sized to store approximately 4500 kg of propellant, with over 2000 kg allocated for a propulsive maneuver to achieve the desired Uranus science orbit. With a 2750 kg dry mass, the spacecraft wet mass at launch is 7235 kg.

¹ Aerospace Engineer, AIAA Member

² Aerospace Engineer, AIAA Associate Fellow

Spacecraft subsystems from previous and near-term missions, such as Cassini and Europa Clipper, are leveraged to provide a Uranus orbiter with minimal development required.

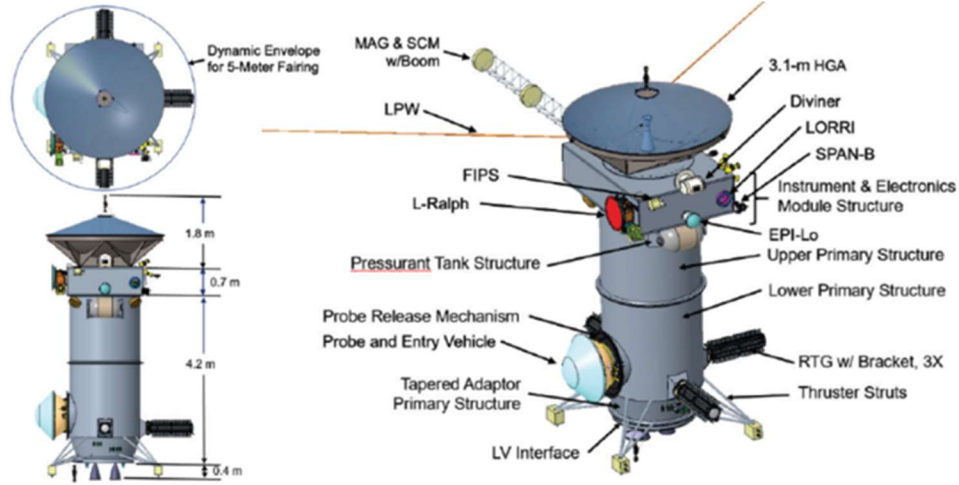


Figure 1: Spacecraft for Uranus Orbiter and Probe Mission [2]

B. Mars Science Laboratory Vehicle

The Mars Science Laboratory (MSL) entry vehicle (EV) protected the Mars Curiosity rover during a critical first phase of landing on the Martian surface [5]. The 4.5-meter aeroshell with thermal protection system (TPS) protects the rover from aerothermal heating and aerodynamic loading during Mars atmospheric entry. The MSL heatshield is composed of a Phenolic Carbon Ablator (PICA) TPS material bonded to an aluminum honeycomb sandwich carrier structure reinforced with graphite composite face sheets [6]. Back shell architecture is similar to the heatshield, with a lower density TPS used (SLA-561V) since more benign thermal environments are encountered in this area [6]. Eight engines located on the back shell allow for bank angle control to guide the vehicle along its desired entry trajectory [7]. The entry system center of gravity location is offset from the aeroshell symmetry axis in order to achieve a nonzero angle of attack and control the vehicle trajectory during atmospheric flight [7].

Although the MSL vehicle was part of an entry, descent, and landing mission architecture, the vehicle design can be utilized for different applications. The MSL heatshield design has shown to be sufficient for an aerocapture maneuver at Uranus [8], [9]. As a result, a vehicle similar to the MSL EV can be used to deliver a spacecraft similar to UOP to Uranus orbit, where it can achieve the post-orbit insertion science objectives defined by UOP.

With recent developments in TPS materials, Conformal PICA (CPICA) has shown performance and manufacturability improvements over PICA [10]. To leverage the improvements made in TPS technology since MSL's mission in 2012, this derivative material is currently under development to provide future solutions for planetary entry vehicles.

III. Methodology

Both the MSL entry vehicle and UOP designs are leveraged herein to provide an aerocapture design solution for a Uranus flagship mission. With the exception of replacing the PICA TPS with CPICA, the MSL entry vehicle architecture is maintained to limit any design risks implied by deviating from a flight proven design.

The purpose of this design is to develop a packaging layout of UOP spacecraft subsystems required for outer planet missions, such as large HGA and RTGs. Using the resulting design, mass properties can be extracted to verify performance of the aerocapture vehicle. To quantify potential risks to exceeding design parameters of the aerocapture system, a mass control policy will be implemented according to standard practices.

C. Aerocapture system mass assessment

Mass margin tracking and mass risk assessment is performed in accordance with ANSI/AIAA-S120A-2015 mass properties control [11]. This margin convention requires total system mass current best estimate (CBE), maximum expected value (MEV), and maximum permissible value (MPV) to be defined, as shown in Eq. (1) and Eq. (2). Margin and total mass growth allowance (MGA) can be referenced for the appropriate program milestone/maturity to determine whether there is a minimal, medium, or high risk to exceeding system MPV mass [11]. Mass risk assessment scorecard implemented in this paper shown in Table 1, which references early program maturity (pre-phase A) from [11] to determine mass risk as a function of both margin and MGA.

$$\text{Mass margin, \%} = \frac{MPV - MEV}{CBE} \times 100\% \quad [1]$$

$$MEV = CBE * MGA \quad [2]$$

Table 1: Mass risk scorecard for MGA and margin

MGA (% of mass CBE)	Margin (% of mass CBE)	Mass risk
>15%	>15%	Low
9% < MGA ≤15%	10% < CBE ≤15%	Medium
≤9%	≤10%	High

Aerocapture vehicle performance and its design parameters (e.g. TPS thickness) is dependent on the mass properties during atmospheric flight [3]. To align the “design-to” system mass with the implemented mass margin policy, mass MPV is selected to be fixed for aerocapture system design. With the design process still in early stages, however, a conservative MPV is desired to manage large system design and mass uncertainty. For pre-phase A flight programs, mass growth allowances and margin are both recommended in [11] to be greater than 15% each, as shown in Table 1. Instead of the recommended 30% sum of MGA and margin, 50% was instead chosen to accommodate any potential unforeseen mask risks to the system. With CBE only required to determine MPV mass (ie. $MPV=1.5 \times CBE$), the aerocapture system mass is estimated as the sum of the MSL aeroshell and relevant UOP subsystems, a total of 2706 kg. MPV mass is derived from the CBE estimate to be 4064 kg.

D. Aerocapture payload

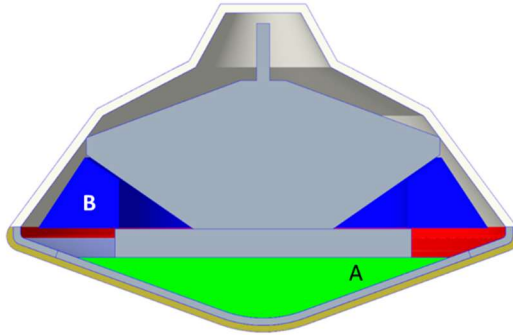
Uranus Orbiter and Probe systems are decomposed to ensure the delivery of an equivalent system capable of performing an identical science mission. UOP propulsion system dry mass is initially bookkept for purposes of obtaining early mass properties estimates, with the expectation that propellant tank and mount sizing will be reduced significantly due to aerocapture maneuver propellant savings over UOP’s fully propulsive orbit insertion [3]. UOP primary structure (428 kg), the volume of which is utilized for housing propellant tanks, are not included as a modification of structural geometry is required for aeroshell integration. All UOP subsystem mass current best estimate (CBE) and their associated mass growth allowance (MGA) presented in [2] are shown in Table 2.

Table 2: Uranus Orbiter and Probe subsystem masses

Subsystem	Description	Mass Current Best Estimate (kg)	Mass Growth Allowance	Maximum Expected Value
Science instruments	Magnetometers, narrow angle camera, thermal IR camera, ...	60.5	8%	65.3
Probe	1.25-meter entry vehicle w/ planetary descent probe	233.0	15%	268.0
RF communications	3.1 m high gain antenna	92.0	13%	104.0
Electrical power	MMRTG's, batteries, ...	250.0	11%	277.5
Avionics	IEM, ...	28.0	10%	30.8
Attitude determination and control	Star trackers, IMU, ...	73.0	5%	76.7
Thermal control	Heaters, louvers, coatings, ...	62.0	20%	74.4
Structures	RTG brackets, Probe release mechanism,	240.7	33%	319.5
Wiring harness		114.0	15%	131.1
Total reference payload mass		1123.2	26%	1412.1

E. Aeroshell geometry trade

Figure 2 illustrates a 3.1-meter HGA and electronics (grey) packaged within an MSL configuration aeroshell, the interior of which is generalized for simplicity. Two mass distributions are defined to house all remaining UOP subsystems within the remaining volume (volumes A and B). Although these volumes yield high mass densities (thus ambitious packing efficiencies), a high risk of interference between back shell interior and HGA is determined.

**Figure 2: UOP subsystem mass distribution for MSL aeroshell packaging (4.5 m)**

Although the Uranus probe is initially neglected in Figure 2 and a dense packing ratio for remaining UOP systems is assumed, very little volumetric margin is observed. As a result, a deviation in aeroshell geometry from the MSL configuration is determined to be appropriate. On one hand, back shell geometry can be modified to provide additional volume. On the other hand, overall aeroshell diameter can be increased and accomplish a similar volume increase with minimal impact to back shell geometry. A larger aeroshell diameter can eliminate some launch vehicles, as current capabilities limit payload diameter to approximately 4.6 meters [12]. However, steeper back shell angles yield a significant deviation to aeroshell geometry with unknown implications to aerothermal heating [13] and aerodynamic stability [8].

In attempts to determine appropriate aeroshell geometry, a notional payload design is generated in Creo to represent the packaged Uranus orbiter with large HGA and three RTG's. A generalized spacecraft structure concept,

with an internal volume equal to UOP's instrument and electronics module, was generated to estimate orbiter structural mass once integrated to an entry vehicle. Two 420 N engines were modeled (ArianeGroup, Model No S400-12) to ensure their axial placement does not overly constrain the spacecraft internal volume. Two generalized aeroshell designs were generated, one with maximum diameter of 4.5 meters and modified (steeper) back shell angle to accommodate HGA (Figure 3a) and the second with 5.0-meter diameter with original MSL aeroshell cone angles (Figure 3b). Note that total aeroshell height is determined by the approximate height of HGA subreflector, and aeroshell heights are identical in both configurations. Since the design shown in Figure 3b yielded additional internal volume while maintaining the non-dimensional MSL geometry, the 5-meter aeroshell was chosen to develop the aerocapture vehicle described herein.

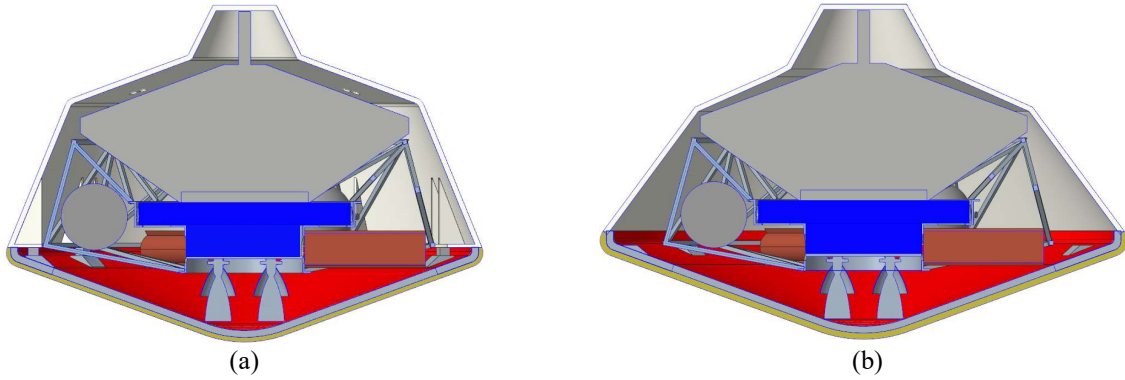


Figure 3: Aerocapture notional design for (a) 4.5 m and (b) 5.0 m configurations

IV. Results

F. Aerocapture vehicle

5.0-meter aeroshell outer mold line (OML) definition is shown in Figure 4, the result of a scaling MSL dimensions from its' 4.5-meter configuration. The CPICA TPS sizing result for the aerocapture 5-meter vehicle [9] is utilized, and a 6.6 cm thick TPS is modeled. With the exception of the heatshield TPS material selection, the remaining architectures are consistent with MSL aeroshell designs, including separation mechanism layout. In addition, reaction control system engines with propulsion module mass distribution are modeled, with identical MSL engine performance, layout and thrust vector orientation [7]. Aeroshell masses are summarized in Table 3. To determine aeroshell MEV mass, a conservative 25% MGA was applied to the aerocapture vehicle to mitigate any mass growth that might be encountered by payload and launch vehicle integration. 126 kg of ballast mass is attached to the back shell to achieve the desired center of mass location (offset from vehicle axis).

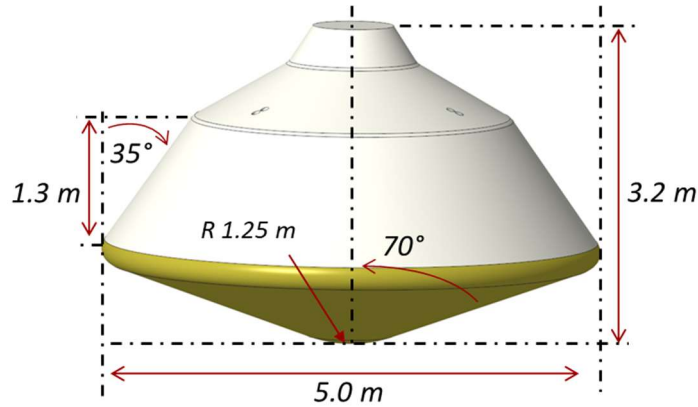


Figure 4: Aerocapture Vehicle Geometry

Table 3: Aerocapture vehicle mass summary

	Component	Mass (kg)
Forward shell	CPICA TPS	322
	Structure	225
	Sep. fittings (9)	55
Back shell	TPS + structure	375
	Propulsion system (RCS)	58
	Separation mechanisms (9)	29
	Mass ballast	126
Total aerocapture system vehicle mass CBE, kg		1190
Total aerocapture system vehicle mass MEV (25% MGA), kg		1487.5

G. Delivered payload

The notional orbiter design shown in Figure 3 was refined for integration to the 5-meter aerocapture vehicle, with the result shown in Figure 5. A structural concept was refined to define aeroshell interfaces without risking HGA interference. An aluminum truss structural design is assumed, with honeycomb panel closeouts to provide an internal volume for orbiter electronics, avionics, and attitude determination and control systems. Three RTG's are modeled and integrated to the spacecraft structure. The orbiter is modeled in Creo and shown Figure 5, with Uranus probe attached (yellow). The probe is modeled with the geometry and mass properties equivalent to that of the UOP probe design. Final aerocapture payload masses are shown in Table 4, with their MGAs defined in accordance with [2].

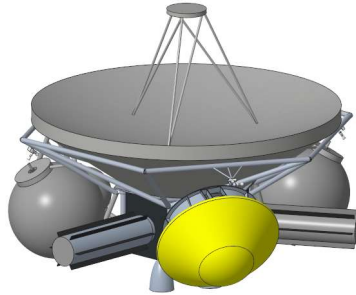
**Figure 5: Aerocapture payload**

Table 4: Aerocapture payload subsystem masses

	Mass Current Best Estimate (CBE), kg	Mass Growth Allowance (MGA)	Mass Maximum Expected Value (MEV), kg
Probe	233	15%	268.0
Propulsion systems	192.4	12%	215.5
Structures	290	25%	362.5
RF Communications	92	13%	104.0
UOP Science Payload	60.5	8%	65.3
Avionics	28	10%	30.8
Electrical Power (including NG-RTGs)	250	11%	277.5
ADCS	73	5%	76.7
Thermal control	62	20%	74.4
Harness	114	15%	131.1
Total payload dry mass	1394.9	15.1%	1605.7

ΔV requirement for the 5-meter aerocapture vehicle is determined to be 175 m/s by Deshmukh et. al [14]. In addition, 681 m/s ΔV is required by UOP [2] as the operational budget. Using the specific impulse of the payload main engines (318 s), the simple rocket equation is used to calculate propellant mass fraction. Assuming dry mass as the MEV mass of the aerocapture payload (1605.7 kg) shown in Table 4, total propellant storage requirement is calculated as 505 kg. Note that the propulsive maneuvers for aerocapture occur after atmospheric exit [3], therefore the aeroshell mass can be excluded for propellant mass calculations. To accommodate propulsive requirements of the payload after the aerocapture maneuver is performed, two large spherical propellant tanks were fitted to the spacecraft structure between the available aeroshell internal volume. A high-level orbiter propulsion system overview is shown in Figure 6.

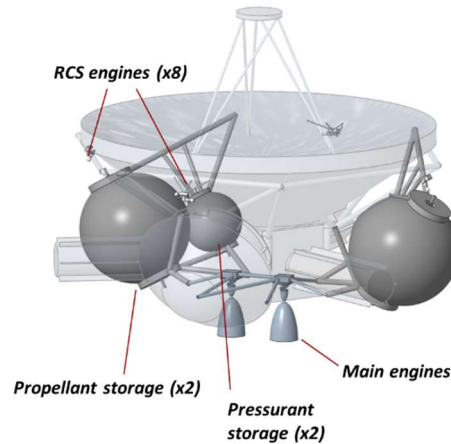


Figure 6: Orbiter propulsion system main components

H. Aerocapture system mass properties

Total aerocapture system mass is summarized in Table 5 with mass properties shown in Table 6. Mass properties are reported in the coordinate system shown in Figure 7. Nondimensional center of gravity location (normalized by vehicle diameter) allow for comparison against the equivalent MSL values previously shown [7]. Dry system mass margins are calculated according to Eq. (1) and shown in Table 7. Although total system dry mass margin is at the edge of the low and medium risk ranges, an overall low mass threat risk is determined due to the large MGA (19.7%) associated with the system.

Table 5: Aerocapture system mass summary

	CBE, kg	MEV, kg	Total MGA, %CBE	MPV, kg	Margin, %CBE
Aerocapture total dry system	2584.9	3093.2	19.7%	3480	15.0%
Propellant	505	505	-	505	-
Aerocapture system entry mass	3089.9	3598.2	16.4 %	4064	15.0%

Table 6: Aerocapture system properties

Total system mass, kg	3168.8
Aeroshell diameter (D), mm	5000
Center of gravity location (x, y, z), mm	(1041, 108.2, 0.5)
Non-dimensional center of gravity location (x/D, y/D, z/D), mm	(0.208, 0.021, 0)
Moment of inertias, (Ixx, Iyy, Izz), kg*m²	(6970, 4893, 4907)

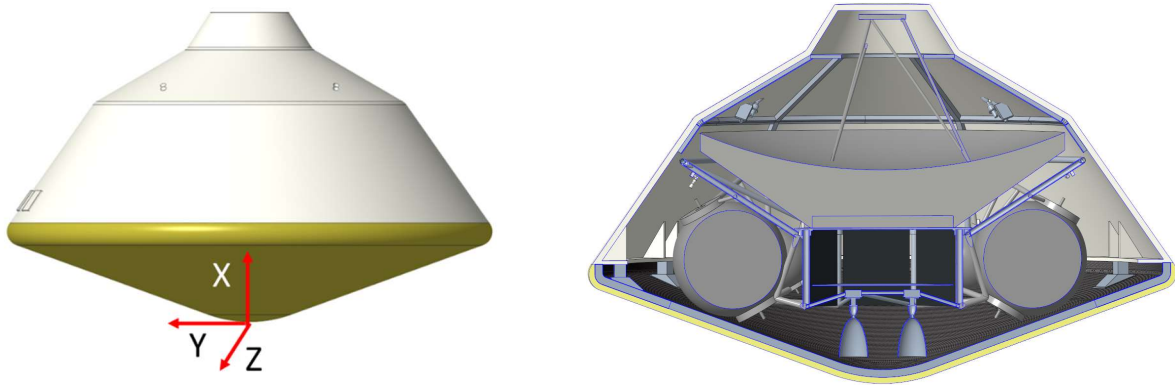


Figure 7: Aerocapture vehicle (a) coordinate system and (b) cross section with integrated payload

Table 7: Aerocapture system mass risk assessment (MPV=4064 kg)

	Value	Mass risk
Total dry system margin	381.8 kg (15%)	Medium
Mass growth allowance (MGA)	542.6 kg (19.7%)	Low
Margin + MGA	924.4 kg (34.7%)	Low

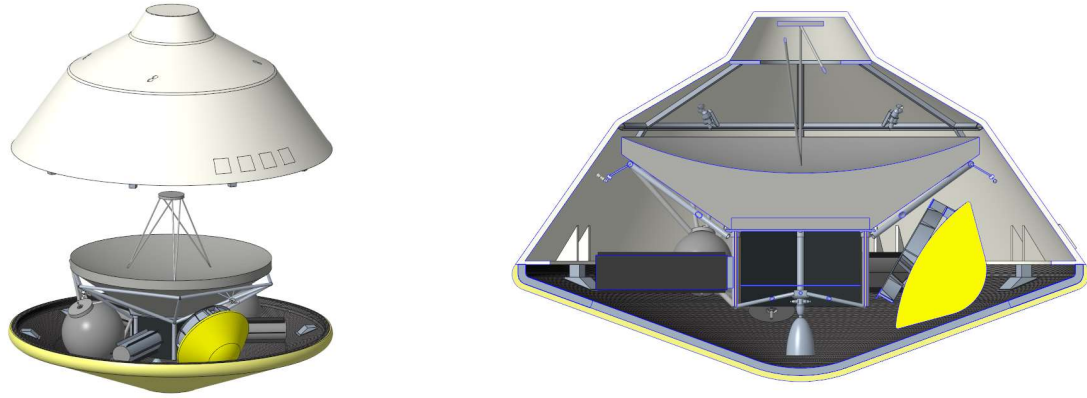


Figure 8: Aerocapture system packaging

I. Probe and accommodations

The 1.25-meter diameter Uranus probe with 45° cone angle is modeled in Creo with similar geometry as the probe presented in [2] with identical total mass (268 kg MEV). The probe model with conceptual spacecraft separation mechanism and interfaces is shown in Figure 9. To determine if an opportunity exists to accommodate additional probes, mass margin is plotted against total number of probes integrated to the aerocapture system. As shown in Figure 10, two total probes can be delivered while maintaining positive mass margin. As vehicle subsystems mature, however, future opportunities for additional probes may arise as MGA + margin remain positive for up to two probes. In terms of volume only, the current aerocapture system may accommodate 3 probes by mounting additional separation mechanisms directly to spacecraft structure.

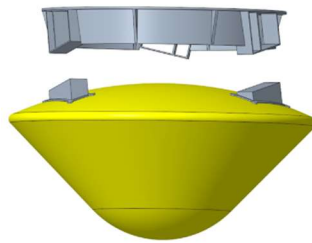


Figure 9: Uranus probe and separation mechanism

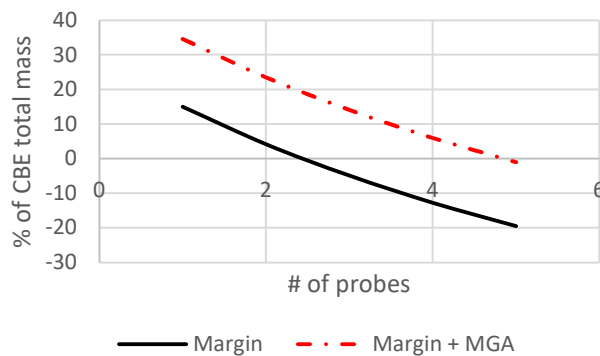


Figure 10: Margin and margin + MGA vs. number of accommodated probes

V. Conclusion

A Uranus Orbiter and Probe designed to fulfill flagship-class mission requirements was decomposed and integrated to an entry vehicle for Uranus orbit insertion using aerocapture. An orbiter conceptual design with large HGA and three RTG's was generated to represent the aerocapture vehicle payload, focusing on high mass and/or volume subsystems to verify an equivalent spacecraft can be accommodated within an aeroshell interior without any mass risk to the system. A 1.25-meter Uranus probe was also designed and integrated to the orbiter structure by separation mechanism. Aerocapture vehicles with diameters of 4.5 and 5.0 meters were presented, with the 5.0-meter configuration showing the most promise for Uranus Orbiter and Probe accommodation. Once the orbiter was integrated to the aerocapture vehicle, mass properties were generated to verify vehicle performance relative to published MSL values. Mass control policy was implemented to evaluate system mass risks for an aerocapture system using standard mass properties control methods with a 15% and 19.4% observed total dry system margin and MGA, respectively. In case of future interest of flying multiple probes to Uranus, a mass margin study was conducted for the aerocapture vehicle, showing the two probes can easily be accommodated without compromising mass margin.

Future work should include the thermal implications of packaging three RTG's within an aeroshell, as significant waste heat is generated by these systems. An appropriate thermal management system will be investigated for the orbiter to ensure the temperature of all systems are kept within their operational range during an extended interplanetary cruise.

VI. Acknowledgements

The work presented in this paper is funded by the Early Career Initiative project sponsored by the NASA Space Technology Mission Directorate. The authors would like to thank the project mentors at NASA Langley Research Center, Ames Research Center, and the Jet Propulsion Laboratory.

References

- [1] "Origins, Worlds, and Life: A Decadal Strategy for Planetary Science and Astrobiology 2023-2032," Tech. rep., The National Academies Press, 2022. <https://doi.org/10.17226/26522>.
- [2] Simon, A., Nimmo, F., and Anderson, R., "Uranus Orbiter and Probe: Journey to an Ice Giants System," Tech. rep., National Aeronautics and Space Administration, 2021.
- [3] Dutta, S., Shellabarger, E., Scoggins, J.B., Gomez-Delrio, A., Lugo, R.A., Deshmukh, R.G., Chadalavada, P.S., Williams, J.D., Garland, J., Johnson, B.J., Matz, D.A., Geiser, J.K., Morgan, J., Restrepo, R., and Mages, J., "Uranus Flagship-class Orbiter and Probe Using Aerocapture," AIAA SciTech 2024, Orlando, FL.
- [4] Girija, A., "A Flagship-class Uranus Orbiter and Probe mission concept using aerocapture," *Acta Astronautica*, Vol. 202, 2022, pp. 104-118. <https://doi.org/10.1016/j.actaastro.2022.10.005>
- [5] Szalai, C., Slimko, E., Hoffman, P., "Mars Science Laboratory Heatshield Development, Implementation, and Lessons Learned", *Journal of Spacecraft and Rockets*, Vol. 51, No. 4, 2014, pp. 1167-1173.
- [6] Mahzari, M., Beck, R., Hwang, H., Monk, J., Morgan, J., Williams, J., Edquist, K. "Development and Sizing of the Mars2020 Thermal Protection System," AIAA 2022-3951. AIAA Aviation 2022 Forum. June 2022.
- [7] Schoenenberger, M., Rhode, M., Paulson, J., Van Norman, J. "Characterization of Aerodynamic Interactions with the Mars Science Laboratory Reaction Control System Using Computation and Experiment," AIAA 2013-971. 51st AIAA Aerospace Sciences Meeting including the New Horizons Forum and Aerospace Exposition. January 2013.
- [8] Shellabarger, E., Scoggins, J.B., Hinkle, A.D., Dutta, S., Deshmukh, R., Patel, M., and Agam, S., "Aerodynamic Implications of Aerocapture Systems for Uranus Orbiters," AIAA SciTech 2024, Orlando, FL.
- [9] Morgan, J., Williams, J.D., Venkatapathy, E., Gasch, M., Deshmukh, R.G., Shellabarger, E., Scoggins, J.B., Gomez-Delrio, A., Tackett, B., Dutta, S., "Thermal Protection System Design of Aerocapture Systems for Uranus Orbiters," AIAA SciTech 2024, Orlando, FL.

- [10] Milos, F., Gasch, M., Prabhu, D., "Conformal Phenolic Impregnated Carbon Ablator (C-PICA) Arcjet Testing, Ablation and Thermal Response," AIAA 2015-1448. 53rd AIAA Aerospace Sciences Meeting. January 2015.
- [11] Standard: Mass Properties Control for Space Systems (ANSI/AIAA S-120A-2015(2019)). January 2015.
- [12] SpaceX Falcon User's Guide. September 2012.
- [13] Scoggins, J.B., Hinkle, A.D., Shellabarger, E., "Aeroheating Environment of Aerocapture Systems for Uranus Orbiters," AIAA SciTech 2024, Orlando, FL..
- [14] Deshmukh, R.G., Dutta, S., Lugo, R., Restrepo, R., Mages, D., Johnson, B., Matz, D., Geiser, J., Scoggins, J.B., Shellabarger, E., Gomez-Delrio, A., Williams, J., "Performance Analysis of Aerocapture Systems for Uranus Orbiter," AIAA SciTech 2024, Orlando, FL.
- [12] Dutta, S., Shellabarger, E., Scoggins, J.B., Gomez-Delrio, A., Lugo, R.A., Deshmukh, R.G., Chadalavada, P.S., Williams, J.D., Garland, J., Johnson, B.J., Matz, D.A., Geiser, J.K., Morgan, J., Restrepo, R., and Mages, J., "Uranus Flagship-class Orbiter and Probe Using Aerocapture," AIAA SciTech 2024, Orlando, FL.