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A Dual Outer Planet Mission to Uranus and Neptune Launched with A Single SLS

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Abstract – This paper describes an intriguing mission that is made possible with the newly developed NASA Space Launch System (SLS). The SLS flew successfully in 2022, and its capabilities will open up new opportunities for outer-planet exploration. With a single launch, the SLS can inject two Flagship class spacecraft on trajectories that terminate in orbit, one at Uranus 16 years after launch and the other at Neptune 18 years after launch. Shorter mission durations are possible with smaller spacecraft. Alternate methods to accomplish the mission are suggested and the chosen approach, which employs Earth and Jupiter gravity assists, is described in detail. Potential hazards that might be encountered during close approaches of Earth and Jupiter are assessed. A proposed spacecraft configuration in the SLS fairing is shown and a preliminary payload mass statement is provided.

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1. INTRODUCTION

In addition to launching heavy elements with large diameters, the SLS is also well suited to inject smaller spacecraft to extremely high velocities. Because launch costs represent a significant portion of total mission costs, it is important to achieve the greatest return for each launch. One approach to accomplishing this is to deliver separate payloads to each of two destinations with a single launch. This feat is routine for small payloads launched into Earth orbit with the Atlas V or Falcon 9 rockets, but for the significant payloads required to perform substantive science at deep space targets, a much greater launch capability is needed. The SLS can provide just such a capability. The following paragraphs describe such a mission that delivers flagship class science spacecraft to the orbits of Uranus and Neptune.

Any outer planets mission must simultaneously satisfy three challenging mission design pillars: a specific planetary alignment between Earth, destination, and any planetary gravity assists; sufficient injection velocity provided by the launch vehicle; and a robust spacecraft capable of both a Deep Space Maneuver (DSM) as well as capturing itself at the destination. Overcoming these challenges while concurrently targeting two far-flung planets is formidable, but achievable.

A constraint that complicates the design of a dual target mission is that the same hyperbolic excess velocity vector at Earth departure must be used to initiate the trajectories to both targets. This severely limits opportunities for such a mission. A mission to any of the planets beyond Jupiter is most efficiently performed using a Jupiter Gravity Assist (JGA). Adding Jupiter to the destination list alongside Uranus and Neptune complicates the necessary planetary alignment and requires clever astrodynamics to overcome.

Opportunities for efficient Earth-Jupiter direct transfers occur approximately every 13 months, the Earth-Jupiter synodic period. The minimum injection energy (C3) required for an Earth-Jupiter transfer is approximately 80-90 km^2/s^2. For the scientific payloads desired for this mission, this launch energy is excessive. A variety of methods exist to reduce the launch energy for a mission, all of which come at the expense of mission duration. Typical approaches include some combination of Earth and Venus gravity assists, coupled with one or more DSM's. For example, The Cassini mission began with two Venus gravity assists followed with Earth and Jupiter gravity assists. The Galileo mission included a Venus gravity assists followed by two Earth gravity assists. The resulting C3's for each of these missions was about 16 km^2/s^2. However, the proposed mission is already targeting three separate planets from a common launch vector and the likelihood that Venus is in a convenient location to contribute energy to the mission under the highly constrained circumstances is not great. Furthermore, including Venus in the mix introduces additional spacecraft thermal control subsystem issues because the

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trajectory must pass much closer to the sun. For these reasons, we chose to avoid including Venus in the mission itinerary and include only Earth gravity assists.

The specific method employed to reduce launch C3 is to prepend the primary mission trajectory with a nominal two-year Earth-Earth loop that leverages the departure C3 needed to initially establish the loop (25-28 km^2/s^2) with a relatively small DSM of several hundred m/s to deliver the larger orbital energy needed at Earth return to continue on to Jupiter. The larger energy is achieved because the spacecraft trajectory approaches Earth's orbit with a higher incidence angle, thereby increasing the speed relative to that of Earth. The optimum DSM location is normally near aphelion of the two-year loop, but a passage distance constraint will result in other locations, usually earlier in the two-year loop. The loop is known as a ΔVEGA loop (ΔV Earth Gravity Assist), which has been used successfully on past high energy missions. The purpose of the DSM is to re-direct the spacecraft trajectory to encounter Earth at some interval of time (~50 days) before the nominal two-year orbital period (2-ΔVEGA) or after the nominal 2-year orbital period (2+ΔVEGA). An example of the two options is shown in Figure 1. Each option connects the prepended spacecraft trajectory (violet path) with the same Earth-Jupiter trajectory (gray path) and with the correct hyperbolic excess speed for the gravity assist. Both the size of the DSM and the Earth passage distance will differ for the two options and the better of the two depends on mission characteristics.

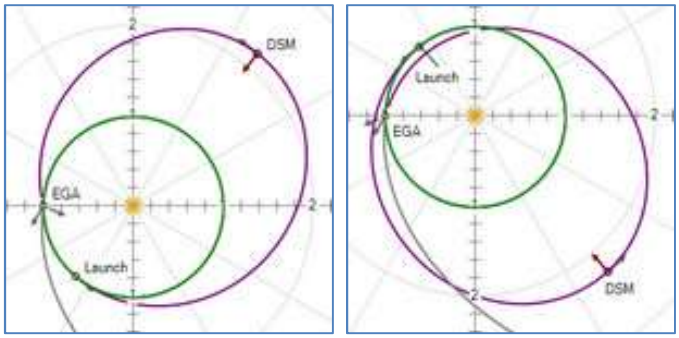


Figure 1 - 2-DVEGA (left) and 2+DVEGA (right)

2. Space Launch System

The SLS is a family of launch vehicles; the Block 1 configuration, launched in November 2022 used the ICPS upper stage. The Block-1B will evolve to a new, higher performing Exploration Upper Stage (EUS). The EUS, presently in development, is four times heavier than the ICPS. It utilizes 4 RL10 engines and carries 112.5-mt of propellant. The Block-1B is able to inject 22.0-mt to the energy described for this mission (C3=28.0 km2/s2).

3. Results

The previously discussed heliocentric mission design methodology was implemented in the Mission Analysis Environment (MAnE™) suite of software tools which are

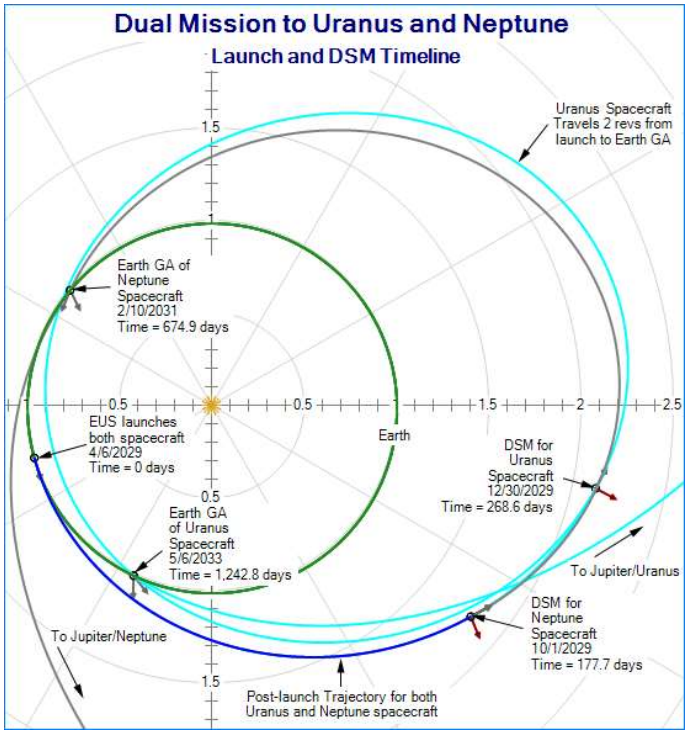


Figure 2 – Uranus/Neptune Spacecraft ΔVEGA and DSM Trajectories

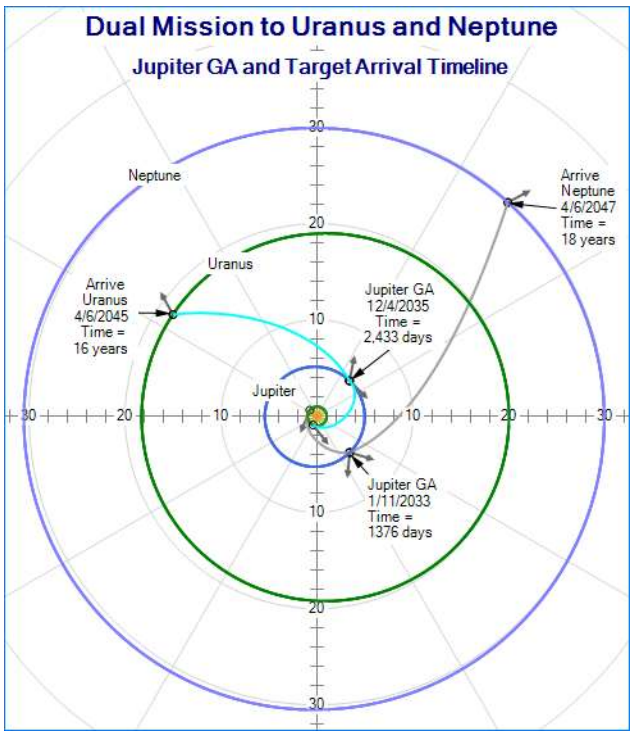


Figure 3 – Post EGA Spacecraft Trajectories

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Parameter	Uranus Spacecraft	Neptune Spacecraft
Launch Date	4/6/2029	4/6/2029
Launch C3 (km^2/s^2)	28.028	28.028
DSM Date	12/30/2029	10/1/2029
DSM Δv (m/s)	839	1,775
Earth GA Date	5/26/2033	2/10/2031
Earth Pass. Alt. (km)	1,436	300
Jupiter GA Date	12/4/2035	1/11/2033
Jupiter Pass. Dist. (radii)	7.0158	16.0761
Target Arrival Date	4/6/2045	4/6/2047
Capture Orbit (radii)	1.5x60	1.5x60
Capture Δv (m/s)	1,197	1,563
Total Δv (m/s)	2,036	3,338

Table 1 – Trajectory Characteristics

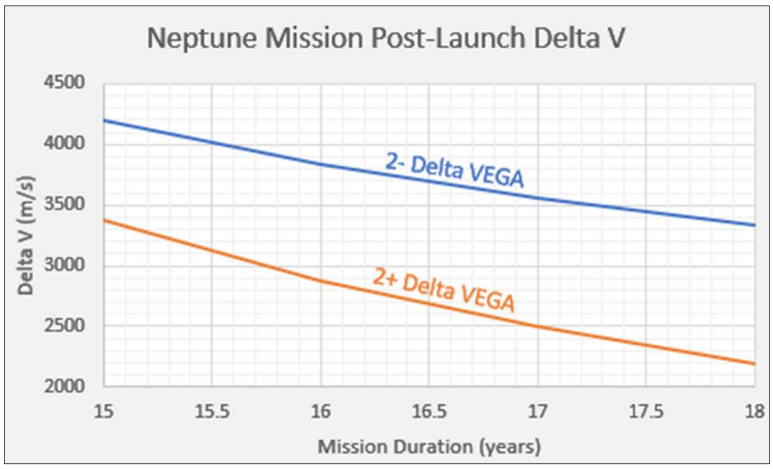


Figure 4 – ΔVEGA Trade Study for Neptune Spacecraft

used for the optimization and graphical visualization of multi-leg interplanetary missions. The result is a rare trajectory displayed in Figures 2 and 3 that demonstrate a mission using a ΔVEGA loop, along with a JGA, to reach Uranus and Neptune with two payloads in a single launch. In each figure, the black arrows at departure and/or arrival of each event represent the directions of the hyperbolic excess velocity vectors or the heliocentric velocity vectors at the DSM (these overlay each other at the scale shown).

The red arrow at the DSM is the direction of the Δv maneuver. Note that display of the ΔVEGA loops have been suppressed in Figure 2 to focus on the outer planet trajectories. Table 1 provides further specific details about the heliocentric trajectory for each spacecraft after launch vehicle separation. After the launch opportunity in April 2029, the next favorable planetary alignment for dual launch to the Ice Giants in the manner shown above will not occur for several decades.

4. DISCUSSION

As part of the iterative mission design process, the 2- and 2+ΔVEGA options were traded against total post launch Δv (DSM plus capture) and mission duration in Figure 4. As the graph shows, the 2+ΔVEGA option is clearly superior for the Neptune bound spacecraft from a Δv, and therefore mass, perspective.

In order for the two missions to be accomplished with a single launch, the optimum launch dates for the two missions launched separately should be approximately the same, within a few weeks. Due to the relative configuration of the planets, the better Earth passage date for the Uranus spacecraft is about two years later than that for the Neptune spacecraft. Fortunately, however, this difference can be accommodated by requiring the Uranus spacecraft to remain in its two-year ΔVEGA loop for an extra revolution. Beyond this, one wants the optimum launch dates to occur in the same month. For Uranus, the optimum launch date is early April for the 2+ΔVEGA loop and mid-July for the 2-ΔVEGA loop. For Neptune, the corresponding optimum launch dates are early January and early April. Clearly, this dictates that the Neptune launch be compromised by choosing the costlier 2-ΔVEGA loop in order to achieve matching launch dates for the two trajectories. The data in Table 1 results for optimizing the Neptune trajectory for the 2-ΔVEGA loop and accepting a small penalty in the DSM Δv of the Uranus trajectory to force the same launch date as that of the Neptune trajectory.

The operational plan is to separate the two spacecraft after launch and before the time of the DSM established for the Neptune spacecraft. The Uranus spacecraft would execute a separate DSM at a location that is optimized for its own Jupiter arrival date. Although this essentially doubles the total DSM Δv for the mission compared to that needed if the two-spacecraft remained as a single stack through a single DSM, the initial mass of each spacecraft would be halved, leaving the total propellant expended relatively unchanged.

A trajectory design issue relating to the Earth and Jupiter gravity assists is the required close approach altitude relative to known environmental hazards such as Low Earth Orbit (LEO) satellites or space debris, natural radiation belts, and moons. The likelihood of LEO satellites or space debris impacting the transiting spacecraft during the brief approach window is extremely small, but nevertheless that statistical chance should be considered in the final trajectory. Fortunately for the mission design, the Uranus spacecraft flies by Earth at an altitude of 1,436 km which is well outside the operating range of typical LEO satellites and debris built up over decades of space activity. The Neptune mission by contrast only has a 300 km flyby altitude which is well inside LEO. At this altitude satellites are not commonly operating for extended durations and any members are highly likely on their way to entering Earth's atmosphere. Similarly, persistent debris in this region has been removed due to atmospheric drag and is

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not an impact concern. While artificial objects are a concern for Earth only, radiation belts are present at both gravity-assist planets.

Earth is host to the Van Allen Radiation Belts which exist between approximately 1,000-12,000 km altitude for the inner belt and 13,000-38,000 km for the outer belt [2]. Jupiter is also well known to have strong radiation belts within its magnetic field especially in the region <5 Rj [1]. The Io Plasma Torus is another major source of radiation exposure existing between approximately 5.2-10 Rj [3].

The Uranus spacecraft just grazes the inner Van Allen belt however since it passes closer to Jupiter on the JGA at 7 Rj, it should be well shielded to cover both encounters after adding in the impact of the Io Torus as well. The Neptune vehicle will fare better with a more distant JGA of 16 Rj. Due to the close approach speeds of both spacecraft during the gravity assists, the total dose exposure is expected to be relatively small because of the short duration in the most intense radiation zones.

The final hazard to consider is the Jovian moons of Io, Europa, Ganymede, and Callisto. They orbit the planet at distances of 6 Rj, 9.6 Rj, 15.3 Rj, and 27 Rj. Both vehicles fit between the moon's orbits with relative ease and neither flyby comprises a significant concern. For the Neptune spacecraft JGA in 2033, the closest moons on the approach side of Jupiter are well inside of the spacecraft's close approach distance. For the closer Uranus JGA in late 2035 of 7 Rj, all moons are on the far side of the planet from close approach.

Another mission design component is the initial capture orbit at the destinations. For this paper, we have chosen both spacecraft to capture into a 1.5 x 60 planetary radii orbit upon arrival. This orbit strikes a balance between minimizing the capture Δv and transitioning to any potentially desirable science orbit. The high apoapsis allows for inexpensive plane change corrections,

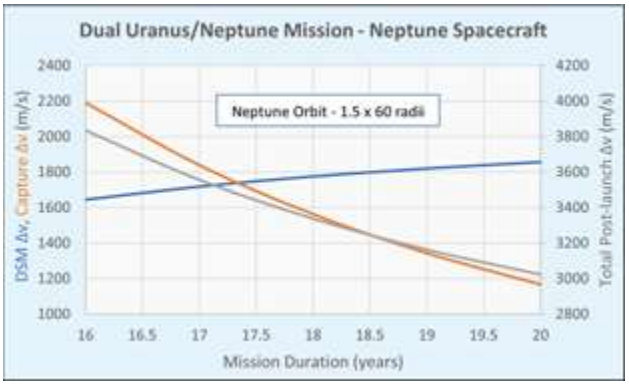


Figure 5 – DSM Δv and Capture Δv vs Mission Duration

identification of navigational hazards not possible with remote imaging prior to launch, and time outside of any high dose radiation environments if they are discovered. The low periapsis conversely gives close flybys of the upper atmospheres and cloud formations along with a

SLS Launched Dual Uranus Neptune Mission

SLS Payload (kg)		
Uranus Spacecraft	**6,503**	
Dry mass (kg)	2,435	
Dry Mass growth	438	
Propellant Load	3,456	
Propellant reserve	173	
Neptune Spacecraft	**6,759**	
Dry mass (kg)	2,685	
Dry Mass growth	483	
Propellant Load	3,420	
Propellant reserve	171	
DSM Stage (Orion 50 XL)	**4,374**	
Inert Mass with adaptor	484	
Propellant	3,705	
Propellant reserve	185	
Total System Mass (kg)	**17,636**	
Payload Attach Fittings	973	
Launch Margin	**3,391**	18.2%
Launch Vehicle Capability	**22,000**	

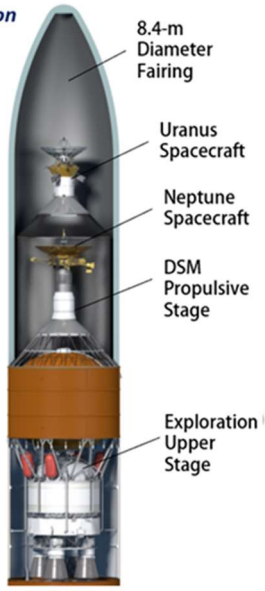


Figure 6 – Spacecraft Launch Configuration and Mass Statement

favorable orbital energy position to lower the apoapsis. It is possible to change the capture orbit pending further studies and defining mission objectives. It may also be desirable to shorten the mission duration at the cost of increased capture Δv as shown in Figure 5 for the Neptune spacecraft. For a modest amount of 200 m/s, the total transit time can be reduced by approximately 1 year. As shown, changing the DSM in contrast does not have a significant impact on mission duration due to its role of aligning the Earth return after the ΔVEGA loop for a subsequent JGA.

Given the mission design considerations previously discussed along with Δv requirements for DSM and capture maneuvers, we have conceived preliminary spacecraft designs and a launch configuration on the SLS Block 1B as seen in Figure 6. The spacecraft are configured with dry mass and propellant margins of 18% and 5% respectively. An additional 18% payload growth is reserved for launch vehicle margin (3.4 mt). The spacecraft masses, with propellant, are 6.50 and 6.76 mt, respectively, (Figure 6) as compared to NASA's Flagship class Cassini spacecraft with a mass of 5.6 mt.

The Uranus spacecraft's propellant tanks contain both its DSM and capture propellant; Δv's for these (Table 1) are 839 and 1,197 m/s respectively.

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To make up for using the less efficient ΔVEGA route of the Neptune spacecraft, an Orion-50XL solid motor (Figure 7) is used with it; it provides 70 percent of the DSM Δv. The DSM and capture Δv are 1,775 and 1,563-m/s, respectively, for Neptune (Table 1).



Figure 7 – Orion 50XL Solid Propellant Upper Stage for DSM

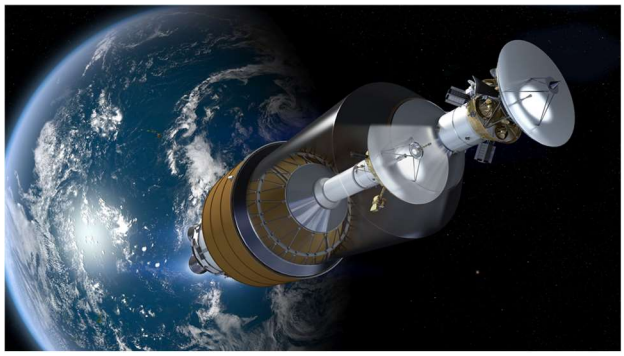


Figure 8 – Earth Departure with The EUS Upper Stage

5. URANUS MISSION

The heliocentric distance of Uranus varies between 18.3 and 20.1 AU. Its features include: unique atmospheric dynamics due to extreme axial tilt (98 degrees), effectively orbiting the Sun on its side, like a celestial barrel; unusual magnetosphere geometry; unexplained energy balance; and dynamically evolving rings and moons. The planet's aquamarine hue is caused by the presence of methane, which absorbs red light from the weak incident sunlight and reflects back colors at the blue-green end of the color spectrum. Uranus has an extensive moon system.

6. NEPTUNE/TRITON MISSION

The orbit of Neptune is nearly circular with its heliocentric distance varying from 29.8 to 30.3 AU. The main axis of its powerful magnetic field is inclined about 47-degrees relative to the planet's rotation axis. Its magnetosphere undergoes variations during each rotation because of this misalignment. Neptune's atmosphere is made up mostly of hydrogen and helium with a small amount of methane.

Triton, the largest of Neptune's satellites, is different from all other icy satellites Voyager has studied. About 3/4 the size of Earth's Moon, Triton circles Neptune in a tilted, circular, retrograde orbit of 5.9 days in duration. Volcanic eruptions at low cryogenic temperatures were observed in Voyager images; several geyser-like volcanic vents spew nitrogen gas laced with extremely fine, dark particles. The particles are carried to altitudes of 2 to 8 km and then blown downwind before being deposited on Triton's surface. An extremely thin atmosphere extends as much as 800-km above Triton's surface, which is very bright, reflecting 60 to 95 percent of the sunlight that strikes it. Polar caps and wind streaks can be seen on Triton's surface. Triton appears to have the same general size, density, temperature and chemical composition as Pluto.

7. CONCLUSIONS

In this paper we have described the trajectory design and necessary considerations for a dual launch mission to the Ice Giant planets of Uranus and Neptune with a launch date occurring late in 2029. This mission successfully satisfies the 3 main pillars of outer planets trajectory design: a favorable planetary alignment, sufficient launch vehicle C3, and capable spacecraft to perform both a DSM and capture at the targeted destination.

To launch both spacecraft in a single mission requires using a 2-year ΔVEGA loop setting up an Earth return flyby and targeting a subsequent JGA to reach the required C3. A ΔVEGA loop provides advantages over using Venus as the gravity assist planet by not requiring another planetary encounter, simplifying the set of thermal design environments to consider, and its constant availability independent of launch opportunity. The Uranus mission's JGA occurs 2 years later than the Neptune spacecraft because of the planet positions. Therefore, that vehicle must remain in the ΔVEGA loop for an additional orbit for a total of 4 years between launch and Earth gravity assist.

Additionally, we have considered the impact of environmental hazards present at the flyby locations and the selection of an initial capture orbit. In this preliminary analysis we do not see evidence that debris, radiation belts, or moons require accommodation through modifications to the trajectory. The choice of a 1.5 x 60 planetary radii orbit at Uranus and Neptune was selected to strike a balance between expended Δv by the spacecraft upon arrival, initial data collection and surveying efforts, and the potential transition to follow-on science orbits.

The SLS is capable of launching the two spacecraft with its 22.0-mt capacity to a C3 of 28.0 km^2/s^2 (Fig. 8). These two spacecraft have a mass in excess of 6.5-mt each, which is tied for the heaviest interplanetary probes launched to date by any space agency. The favorable alignment for this 'dual mission' opportunity, available in 2029, will not re-occur for many decades.

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BIOGRAPHY



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Matthew Ziglar *received his B.S. and M.S. in Mechanical Engineering from Mercer University. He is currently the habitation technical integrator within Boeing Exploration Systems working on a variety of space exploration hardware from sustaining to new development of future exploration habitats and surface transportation. Additionally, Matthew works as a Structures Engineer working to develop lightweight, efficient structures and to infuse new technology into projects. Prior to this he helped sustain and upgrade the ISS structure.*



Jerry Horsewood *received a B.S. degree in Aeronautical Engineering from Purdue University in 1959 and an M.S. degree in Engineering from UCLA in 1964. His career began with Douglas Missiles and Space Systems early in the US space program and he has remained active in the space industry and in software development since. He now operates a sole proprietorship, SpaceFlightSolutions, which is devoted to continuing the development and enhancement of the MAnE™ software. He also provides consulting services for the design and optimization of complex deep-space exploration missions.*