Space Launch System Exploration Upper Stage Development & Missions

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1 author:

Ben Donahue
The Boeing Company

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The NASA SLS Development and Mission Opportunities

Benjamin Donahue, Sheldon Sigmon, Darby Cooper
Boeing Exploration Launch Systems, Huntsville, Alabama, 35806

Abstract

The NASA Space Launch System (SLS) will provide a game-changing capability for the exploration of other worlds, beginning with the Block 1 configuration that utilizes the Interim Cryogenic Propulsion Stage (ICPS). In 2023, the new NASA Exploration Upper Stage (EUS) will evolve the SLS to a significantly higher performance level. The EUS optimizes the powerful SLS Core and Booster Stages, and will provide the capability of achieving greater human exploration, operations and science objectives for 2020-2040 era Beyond Earth Orbit (BEO) missions, including crewed Cis-lunar missions in the mid-2020s, crewed Lunar Surface missions in the late 2020's and crewed Mars missions in the mid-2030s. In this report the Block 1 and 1B configurations and the EUS will be described along with some missions that take advantage of the larger, higher performing EUS upper stage.

I. The NASA Space Launch System

SLS development consists of a series of increasingly capable launch vehicles to incrementally expand BEO exploration from cis-lunar space and then to Mars. The SLS Block 1 configuration (Fig. 1) utilizes the Interim Cryogenic Propulsion Stage (ICPS) and the Block 1B features the large Exploration Upper Stage (EUS). Payload capabilities to Trans-Lunar injection (TLI) are shown in Fig. 2 for four launch vehicles. The Block-1 ICPS is a derivative of the Delta-IV upper stage; the Block 1B EUS is in development now and its TLI capability will range between 39 and 43 mt. The later, 2030's era SLS Block 2 features a TLI capability of 53 mt. Only the SLS can deliver the 27.5 mt Orion Crew Vehicle to the Moon; it delivers significantly more payload to LEO and BEO destinations than any other existing or planned launch systems. The large throw mass of the Block 1B provides a game-changing capability for the exploration of other worlds. By enabling larger margins in the design of exploration platforms and the ability to send multiple copies of atmospheric and surface probes, higher resolution spatial and temporal data can be collected in a single mission. Mission risk can be reduced by increasing the redundancy of each individual system and the architecture by using multiple copies of the same systems.

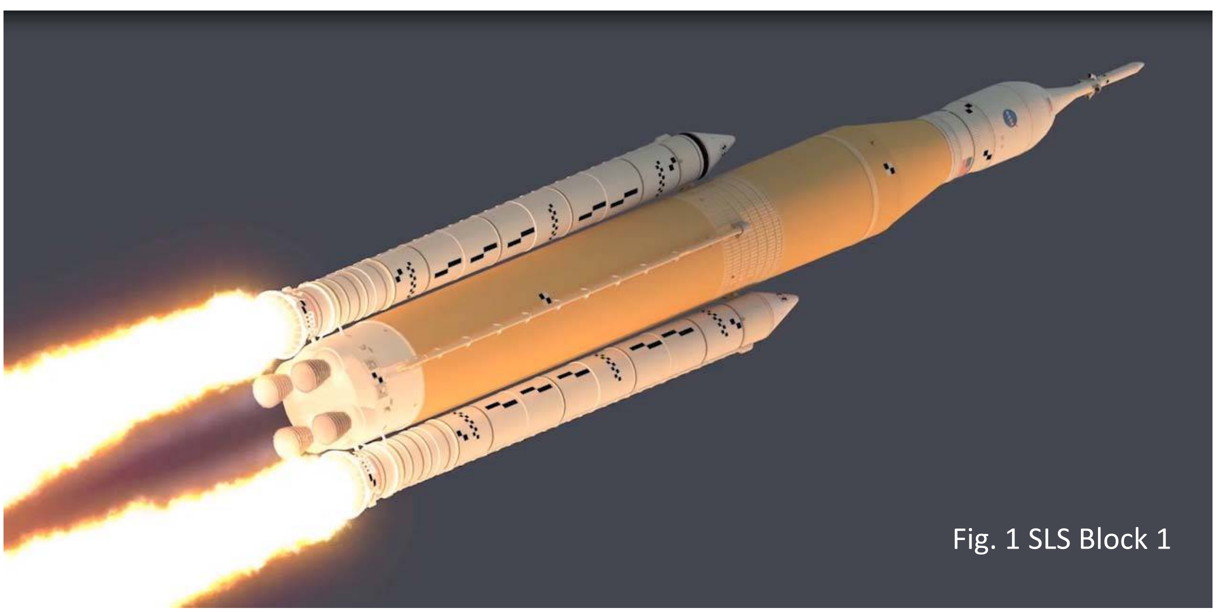


Fig. 1 SLS Block 1

II. SLS Launch Performance

Figure 2 illustrates Launch Vehicle performance to Trans-lunar injection (TLI) (Moon Lift Capability). The SLS lifts significantly more than any other launch vehicle. In Fig 3 Block 1B injection mass, in metric tons (mt), is plotted vs injection energy (C3 km2/s2). TLI performance varies from 39 to 43 mt. Fig. 3 Image courtesy NASA (Ref. 1).

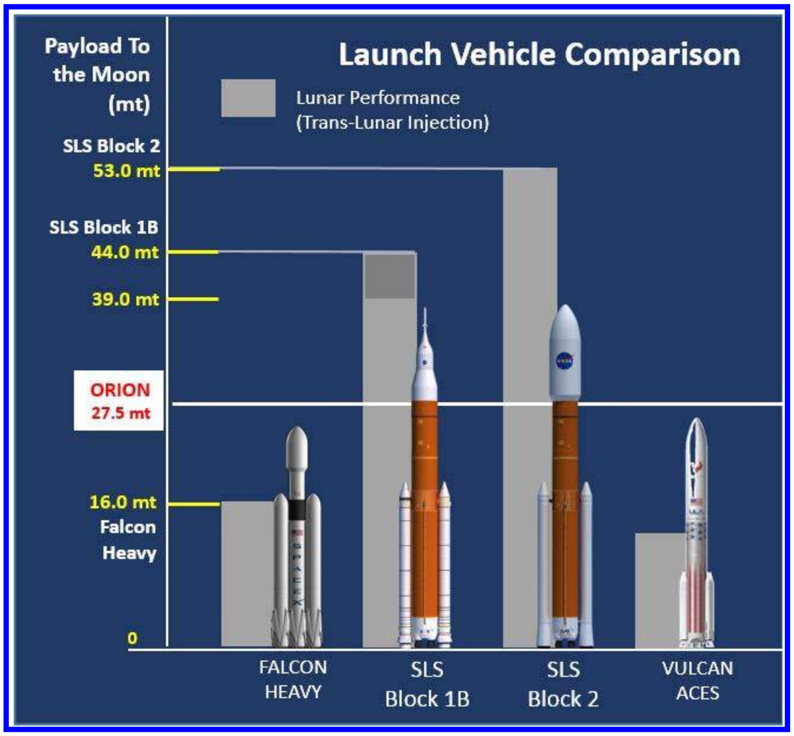


Fig. 2 Launch Vehicle Comparison: Lunar Performance

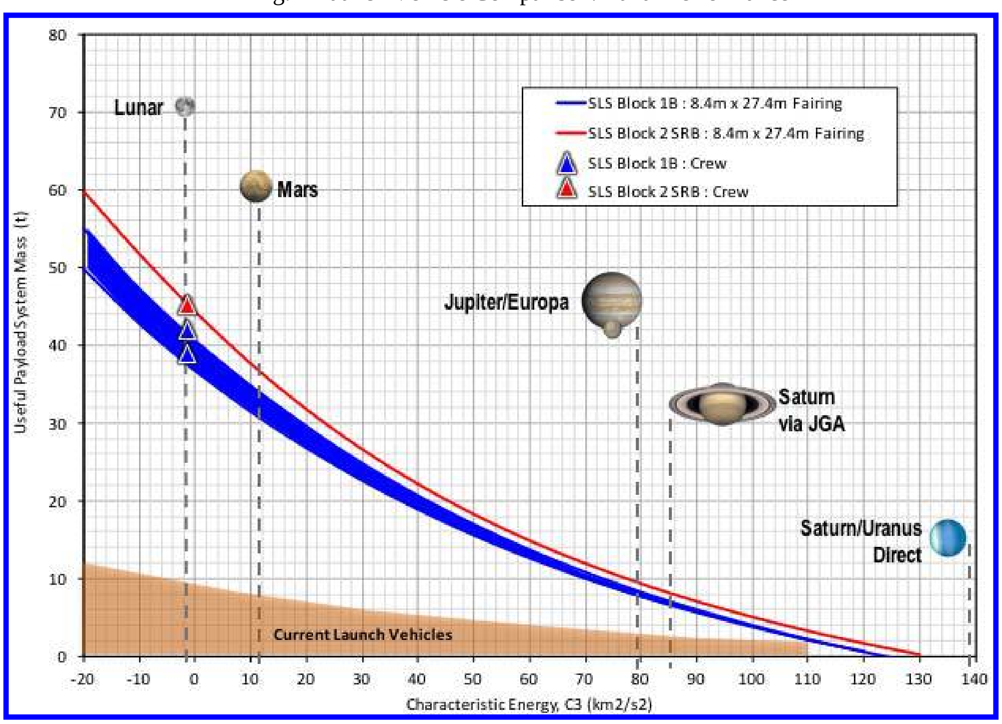


Fig. 3 SLS Payload Capability (mt) vs Injection Energy (C3) (km2/s2) (Ref. 1)

III. SLS Mobile Launch Platform

Congress has funded a second Mobile Launch Platform (MLP) (Fig. 4). This new SLS launch site architecture maintains two MLPs, establishing a standalone cargo launch capability and effectively deconflicting the SLS Block 1 and Block 1B launch schedules. This independent cargo launch capability enables launch of the Europa Clipper mission as early as 2022 and provides the infrastructure necessary to support ongoing science and exploration cargo missions. A second MLP ensures that Block 1 cargo and Orion test missions are decoupled from the next generation SLS Block 1B crewed launches (Fig. 5). With two MLP the SLS program can expand its options for added missions and has more flexibility for a faster launch cadence.

Fig. 4 SL Block 1 and MLP

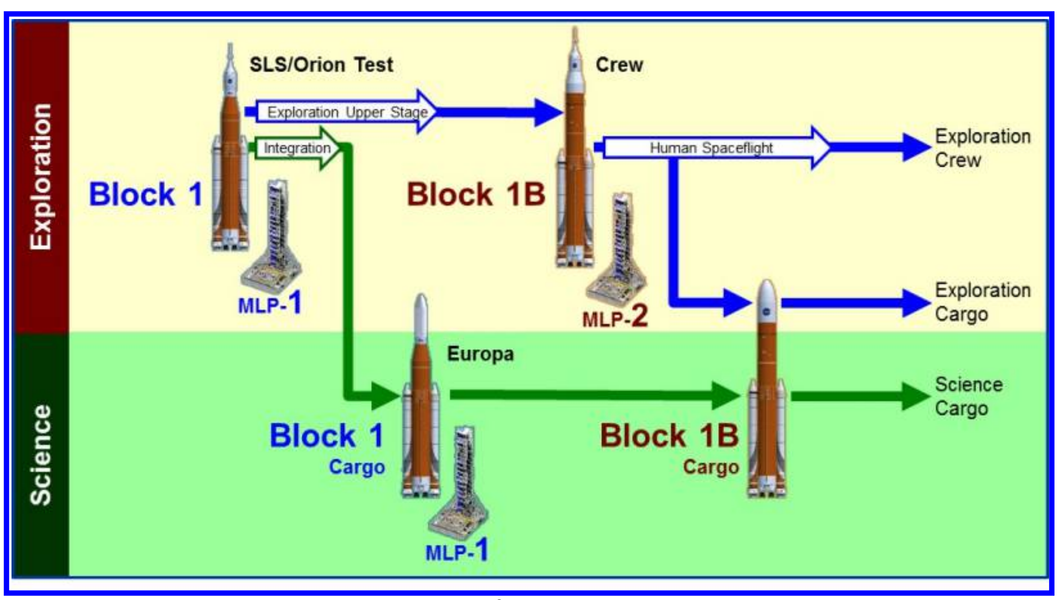


Fig. 5 Mobile Launch Platform (MPL) and SLS Launch Sequencing

IV. SLS Block 1B Adaptors and Fairings

The Block 1B utilizes the Exploration Upper Stage (EUS) which is the next step in the development of the SLS. The Block 1B's Universal Spacecraft Adaptor (USA) (Fig. 6) will allow 10 mt co-manifested payloads to fly with the Orion. These payloads can include habitats; logistics, power & propulsion modules; robotic landers, science packages, mission extension kits and other elements. The USA supports the Orion providing 286 m3 of internal volume. A Payload Attach Fitting (PAF) supports the co-manifested payload and attaches to the EUS forward skirt. The Block 1B's 8.4 m diameter Fairing contains about 3 times the volume of a 5 m diam EELV fairing. The 8.4 m fairing is capable of holding large diameter space telescopes, crewed landers, long duration crew habitats and in-space transfer stages. Fig. 7-A shows two 8.4 m fairings; the short fairing (left) (19.1 m length) (for outer planet spacecraft and other smaller payloads) and the long fairing (right) (27.4 m length). The Evolved Block 2 could fly with a 10 m fairing (Fig. 7-B), containing 6 times the volume of the 5 m fairing. Table 1 (Ref. 1) lists volumes for the 5, 8.4 and 10 m fairings.

Fig. 6. Universal Spacecraft Adapter (USA)

Fig. 7-A. 8.4 m Diameter Fairings

Fig. 7-B. 5.0, 8.4 and 10.0 m Diam Fairings

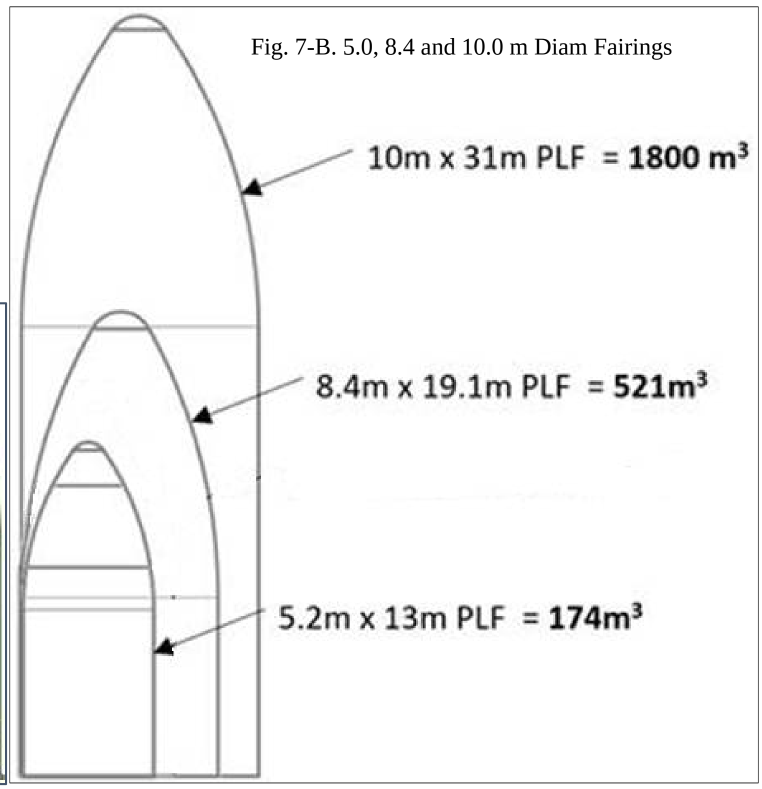


Table 1. Fairing and USA Data from NASA SLS Payload Planners Guide (Ref 1)

Enclosure	5.4m PLF	5.1m PLF	8.4m USA	8.4m USA PLF	8.4m PLF, Short	8.4m PLF, Long	10m PLF
Type	5m PPL	5m PPL	8.4m CPL	8.4m PPL	8.4m PPL	8.4m PPL	10m PPL
Length	55.8 ft	62.7 ft	32.8 ft	47.2 ft	62.7 ft	90 ft	90 ft
	17.0 m	19.1 m	10.0 m	14.4 m	19.1 m	27.4 m	27.4 m
Diameter	17.7 ft	16.7 ft	27.6 ft	27.6 ft	27.6 ft	27.6 ft	32.8 ft
	5.4 m	5.1 m	8.4 m	8.4 m	8.4 m	8.4 m	10.0 m
Internal Diameter	15.1 ft	15.1 ft	24.6 ft	24.6 ft	24.6 ft	24.6 ft	29.9 ft
	4.6 m	4.6 m	7.5 m	7.5 m	7.5 m	7.5 m	9.1 m
Available Volume	7,740 ft³	9,030 ft³	10,100 ft³	11,260 ft³	18,970 ft³	31,950 ft³	46,610 ft³
	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

V. SLS Core Stage Development Status

The SLS Block 1 is progressing toward a FY 2020 launch. Major assembles are in manufacture and test at NASA Michoud Assembly Facility (MAF) and a complete Core stage will enter testing at NASA Stennis Space Center (SSC) in 2019. Descriptions of the major elements of the Core stage are shown in Fig. 8-9 and addressed in the following pages.

Fig. 8 SLS Core Stage Major Elements

Fig. 9 Montage of SLS Flight Hardware

SLS Core Stage Development Status

The Core Stage liquid hydrogen (LH2) tank measures more than 130 feet tall, comprises almost two-thirds of the Core Stage and holds 537,000 gallons of LH2 at minus 423 degrees Fahrenheit. Propellant, flowing from the LH2 and LO2 tanks, will supply the four RS-25 engines which will provide over 1,670,000 lbs of sea-level thrust at Liftoff. The friction stir welded, 2219 Aluminum Core LH2 tank is shown at MAF in Fig. 10, primed and ready for spray on Thermal Protection System (TPS) foam insulation. The similarly constructed Liquid Oxygen tank (LO2) is shown in Fig. 11.

Fig. 10 SLS Core Stage Liquid Hydrogen Tank at NASA MAF

Fig. 11 SLS Core Stage Liquid Oxygen Tank at NASA MAF

SLS Core Development Status

The Core Stage Forward Skirt (FS) (Fig. 12) houses the mission flight computers; the Redundant Inertial Navigation Unit (RINU) is on the forward LO2 done. The avionics contained within the FS communicate with the rest of the vehicle via an external systems tunnel, which runs the length of the vehicle. The FS connects to the Upper stage via an internal interface panel. The FS integrates with the Launch Vehicle Spacecraft Adaptor (LVSA) on the Block 1 ICPS configuration, and integrates with the LO2 tank on the Core. The FS is made of 2219 Aluminum iso-grid barrel panels with forward and aft flanges. Additionally the FS has an umbilical plate interface for ground and environmental systems.

Fig. 12 SLS Core Stage Forward Skirt at NASA MAF

All four RS-25 engines (Fig. 13) that will power SLS for its first flight are ready to be integrated with the core stage at MAF before a Green Run test (Section VI). For SLS, the RS-25s have been upgraded with new controllers and nozzle insulation to perform under SLS environments.

Fig. 13 RS-25 Engines for SLS First Flight ready for Integration with Core Stage
(Aerojet-Rocketdyne photo)

SLS Core Stage Development: Test Hardware

The Core stage Intertank structural test article (STA) is shown arriving at NASA's Marshall Space Flight Center (MSFC) March 4, 2018 aboard the Pegasus (Fig. 14). The Intertank STA will undergo critical testing as engineers push, pull and bend the hardware with millions of pounds of force to ensure it can withstand the forces of launch and ascent. The STA is structurally identical to the flight version of the intertank, which will connect the Core LH2 and LO2 tanks, serve as the upper-connection point for the 2 Five Segment Solid Rocket Boosters (SRB) and house flight navigation avionics and electronics. The Pegasus barge, originally used during the Shuttle Program, was redesigned to accommodate the SLS's 212-ft Core Stage. The 310-ft Pegasus will ferry flight Core stages from MAF to other NASA centers for tests and launch.

The Intertank is a 2219 Aluminum cylindrical structure with flanges on each end. The intertank houses instrumentation components and provides an umbilical plate that interfaces with the ground facility arm for purge gas supply, hazardous gas detection and hydrogen gas boiloff during ground operations. The Intertank consists of mechanically joined skin, integrated stringers and machined panels of aluminum alloy; it is bolted together, not welded, with ribs on the outside for added strength, which is required because the intertank is a connection point for the SLS's twin SRBs. Additionally the Intertank houses a thrust beam which evenly distributes the thrust loads of the Boosters into the Structure of the Core.

Fig. 14 Core Stage Intertank Structural Test Article and the Pegasus Arrive At NASA MSFC for Testing

SLS Core Stage Development: Test Hardware

The engine section (ES) will house the rocket's four RS-25 engines and be an aft attachment point for the two solid rocket boosters. In Fig 15, test engineers install test hardware for the Engine Section Structural Test Article (STA) into a newly constructed 50-foot structural test stand at NASA MSFC. Engine Section testing is now complete. In the stand, electrolically controlled hydraulic cylinders pushed, pulled, twisted and bent the test article with millions of pounds of force. Over 3,000 channels of data for each test case were recorded and analyzed to verify the capabilities of the test article and validate that SLS Core stage design and analysis models prediction are correct. The Engine Section is a steel / aluminum cylindrical structure. It consists of friction stir welded barrel panels and all aluminum stiffening rings. Internal MPS elements (pressurization systems and Thrust Vector Control (TVC) systems are mounted internally). RS-25 engines are supported with a square I-beam design. The entire Engine Section is integrated at MAF with the RS-25s, before shipment to NASA SSC.

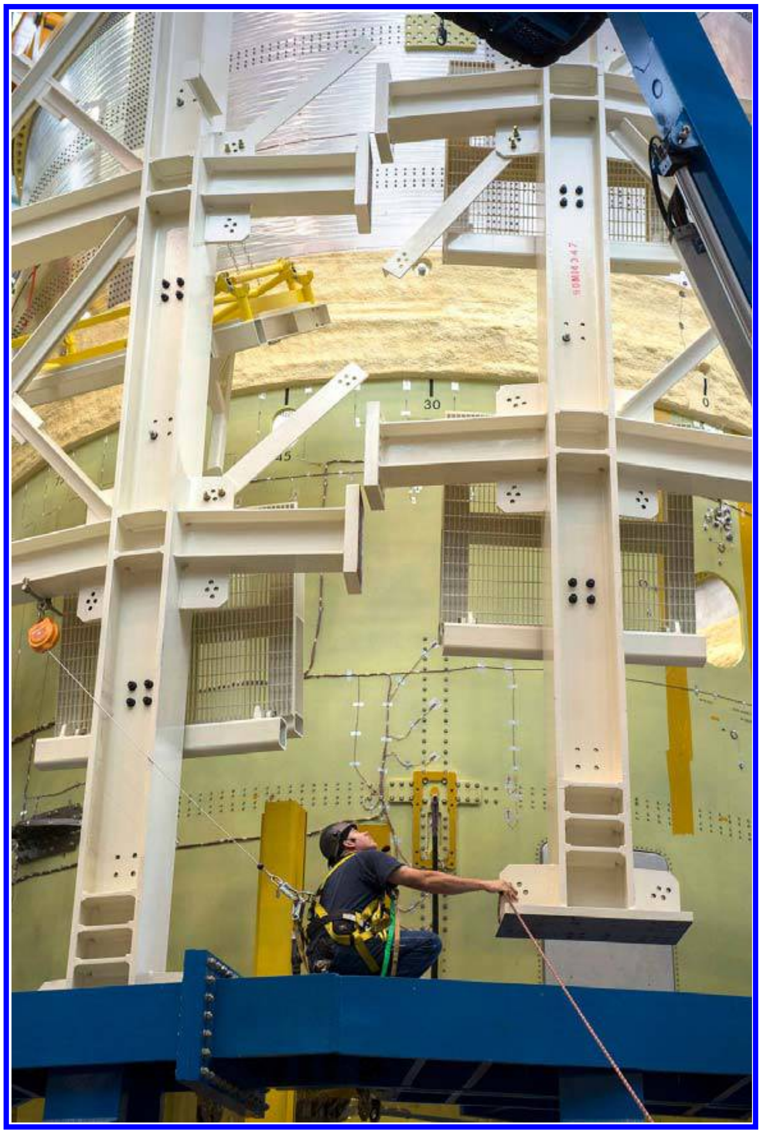


Fig. 15 SLS Core Stage Engine Section at NASA MSFC Test Facility

VI. SLS Core Stage Green Run Test

The SLS personnel will perform a Green run test of the Core Stage at NASA SSC to validate the Main Propulsion System (MPS), structural integration, and avionics functionality. The Green Run is currently scheduled for the second half of 2019. The assembled Core Stage will be shipped from MAF via the Pegasus to NASA SSC, where it will be lifted into the B-2 Test Stand. The same test stand was used in the 1960’s to test the Apollo Saturn-V First Stage. Once test systems are integrated and verified the Green Run will commence with a modal test, followed by a avionics power-on test. Once these systems are verified, the vehicle will be loaded with LO2 and LH2 propellants and Helium (He) pressurant, followed by MPS leak checks and Thrust Vector Control (TVC) checks. The wet dress rehearsal will follow, which is a simulated countdown. Next comes the main test countdown and main engine hot firing. This will be the first time RS-25 engines will be fired in a 4 engine cluster configuration. Thousands of sensors will acquire data for subsequent analysis and verification. The Green Run will provide a full duration, full thrust Core Stage run, and validate all MPS, engine and avionics systems at sea-level conditions. The test will provide valuable experience in countdown and operating procedures for the full up configuration, providing confidence in procedures designed for launch. Post test data analysis will be used to verify MPS simulation predictions, and allow refinements in preparation for flight. Once the Green Run is finished, the Stage will be refurbished and prepared for shipment by the Pegasus to NASA Kennedy Space Center (KSC). In Fig 16 the Green Run sequence of activities is given. The repurposed RS-25 engine and new engine controller has been undergoing single engine tests at a separate test stand at SSC to prove the new unit and its control logic. Drag-on instrumentation (most removed post test) will provide additional data on subsystems to address closure of detailed verification objectives (DVOs).

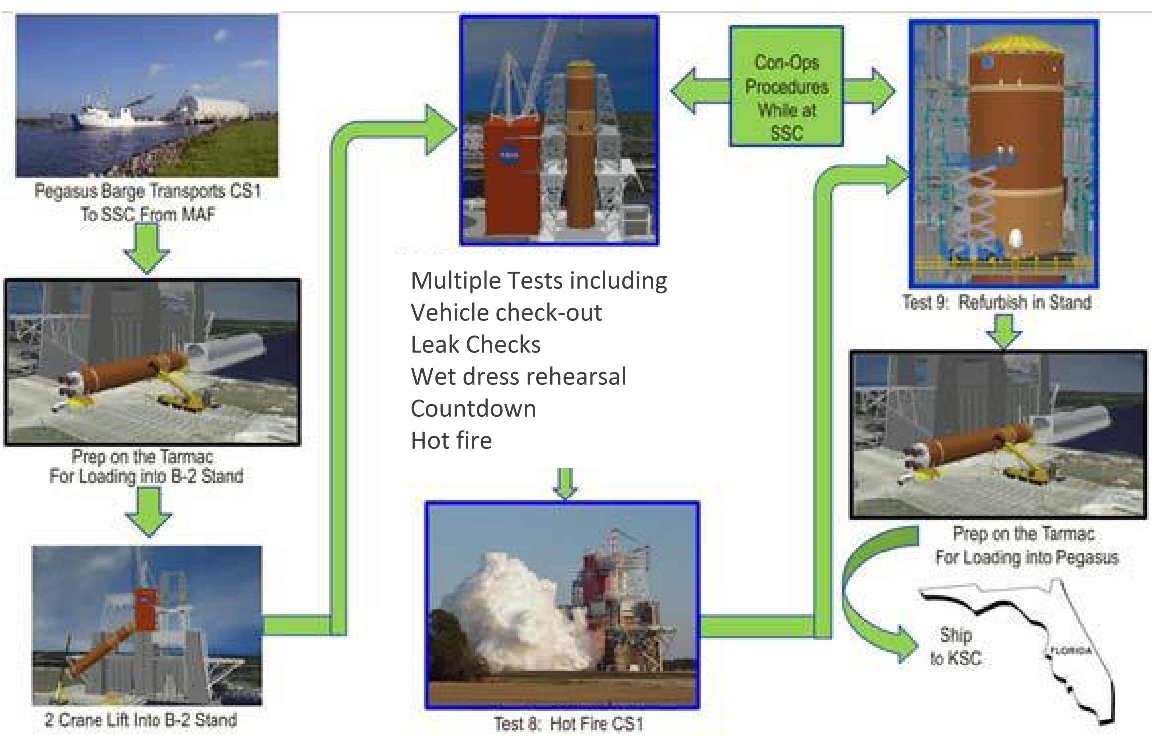


Fig. 16 SLS Core Stage Green Run Activity Flow

VII. SLS Candidate Missions and Payloads

The SLS's launch capability will enable many exciting new exploration missions and enhance many others. In Table 2 four categories of exploration missions are given with multiple destinations and/or payloads listed under each category. Several of these missions or payloads are enabled by the SLS's very large diameter payload fairing, and several more would be enhanced by the SLS's ability to inject sizable payloads on fast, direct (non-swingby) trajectories to outer planet destinations. (Fig. 17)

Table 2. SLS Candidate Missions and Payloads Matrix

Human Space Flight	Science	Astronomy	Other
Cis-Lunar Deep Space Gateway	Europa / Jupiter	Large Monolithic Optic Telescope	Space Solar Power Satellites
Lunar Surface Missions	Saturn / Titan	Lunar Far Side Observatory	Earth Observation
Asteroid /Near Earth Objects	Comet Sample Return	Telescope Servicing	Debris Mitigation
Large LEO & Deep Space Habitats	Trojan Belt Asteroids		GEO Satellite Servicing
Phobos / Deimos	Uranus		DOD / NRO
Mars Entry Descent & Landing Demo	Neptune/ Triton		National Security
Mars Sample Return	Venus Ballon Missions		
Mars Crew	Solar Explorer		
Many Possible Missions			

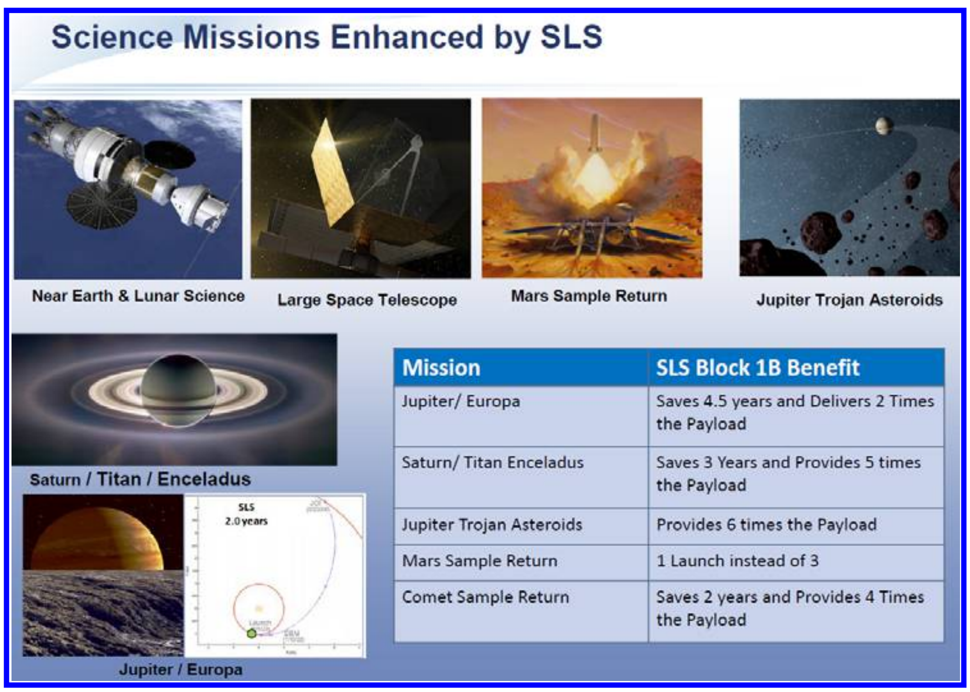


Fig. 17 Science Missions Enhanced by SLS

VIII. SLS First Orion Test Flight to the Moon

The SLS Block 1 ICPS (Fig. 18) will launch at KSC to an initial LEO. Several scenarios for the Lunar mission are available. One scenario is illustrated in Fig 19. After launch to LEO, the Orion/ ICPS combination remains in orbit for two revolutions, afterwards the ICPS engine reignites, boosting the Orion to an intermediate orbit (100 x 38,600 nmi). After separation from the ICPS and 1 revolution in the intermediate orbit, the Orion's Service Module (SM) engines are fired to boost it to the final TLI velocity (C3= -1.0 km2/s2) on its journey to the Moon. The ICPS, after separation from the Orion, fires to place itself on a disposal orbit trajectory. After a 6 day coast, the ICPS will swingby the Moon on its way to a Heliocentric disposal orbit (Fig. 20).

Fig. 18 Block 1 ICPS and Orion

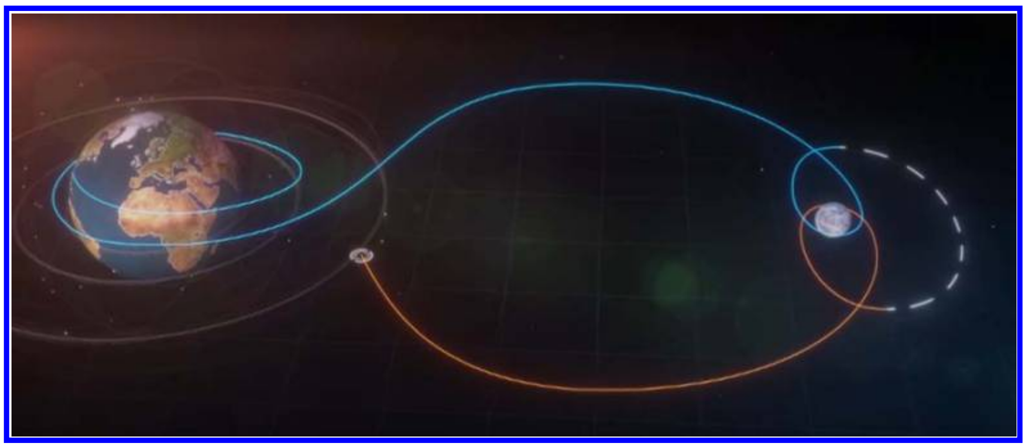


Fig. 19 Orion Trajectory to Lunar Near Rectilinear Orbit

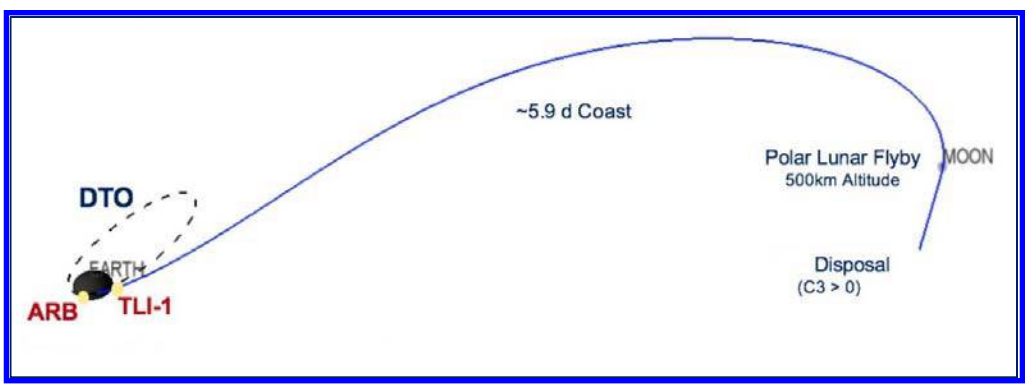


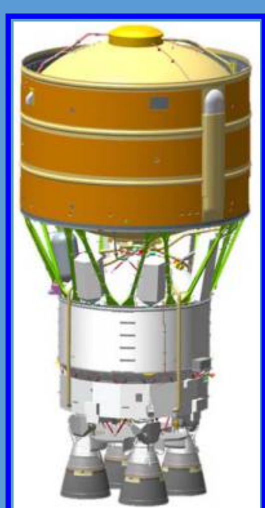
Fig. 20 Block 1 ICPS Trajectory to Disposal

IX. SLS Exploration Upper Stage (EUS)

The EUS is the next evolution in the development of the SLS prepared for a multi-mission role with accommodations for mission modification kits and variable propellant loading capability (Fig. 21). The EUS (Table 3) is a suspended stage powered by 4 RL10-C3 engines and increases the capability of SLS vs the Block 1 with the ICPS. The EUS is a multi-mission stage with accommodations for mission modification kits and variable propellant loading capability. The EUS optimizes the powerful Core and Booster Stages.

Fig. 21 Block 1B Core Stage (left), EUS, USA and Orion (right)

Table 3. SLS EUS Parameters



Element	Feature	Comment
Usable Propellant	~278,000 lb	For LEO mission, prop is off-loaded
Dry Mass	30,000-32,000 lb (31,100)	PMF = 0.890 - 0.905
Total Length	17.8 m	From the top of the LH2 tank to the engine exit plane
Engines	4 x RL10C3	460.1 Isp, Thrust=24,340 lbf
Avionics Shelf	Aft Equipment Shelf	Suspended shelf attached to LO2 tank
Thrust Structure	Trapezoid	Mounts to O2 tank dome
O2 Tank	~3,321 ft3 capacity	5.5 m dia, ~6.0 m length
Interstage Struts	Metallic Struts	~3.6 m length
H2 Tank	~10,413 ft3 capacity	8.4 m dia, ~7.5 m length
Forward Adapter	Forward Adapter	8.4 m dia, ~1.8 m length
Pressurization Sys	Helium	Mounted Externally on Midbody struts and rings
RCS	Hydrazine	Mounted on orthogrid and honey comb equip shelf

X. SLS Co-manifested Payloads and the Lunar Orbit Platform Gateway

The Block 1B's co-manifested payload capabilities will facilitate the construction of a Lunar Gateway. With each Orion launch a Platform element will be delivered. After assembly is complete missions based from the Gateway (Fig. 24) can take place. Robotic landers can sortie to and from the Platform to the surface, delivering payloads and returning samples. Various elements include habitation modules, Logistic modules, airlocks, docking modules, power and propulsion elements and robotic landers, among others (Fig. 23). Propulsion Kits, Kick Stages and/or Life Extension Kits can also be flown with the Orion, to increase its Service Module's delta-V capability or extend the in-space duration of the crew in Orion. Fig. 22 illustrates a Payload Adaptor Fitting (PAF) and co-manifested payload (left) along side a USA (right) (Ref. 3). The Gateway is assembled with multiple flights of the SLs carrying payloads with the Orion. The 10 mt co-manifest capability will provide a functional, man-tended Platform in 3 launches; once, in addition to the Orion, Power & Propulsion, Habitation, and Logistics / airlock modules are emplaced and joined together, the gateway will be operational.

Fig. 22 SLS Block 1B Co-manifested Payload (left) and USA (right)

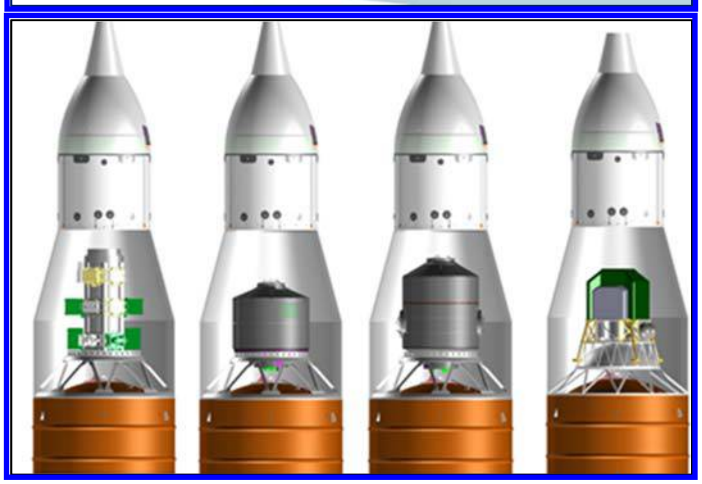


Fig. 23 SLS Block 1B Crew Mission Co-manifested Payload Options

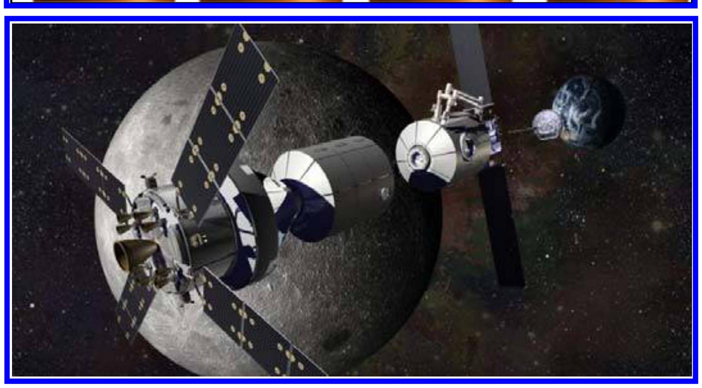


Fig. 24 Lunar Orbit Platform Gateway (LOP-G)

XI. SLS Co-manifested Robotic Lander Lunar Sample Return

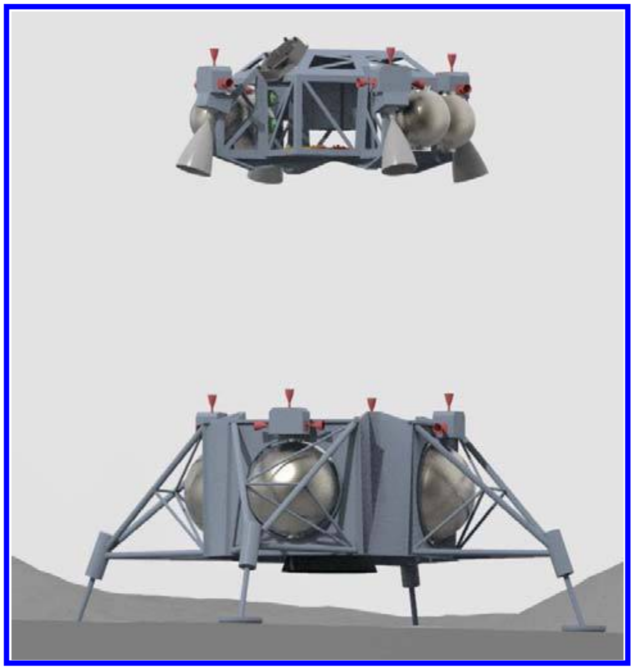


Fig. 25. Robotic Lunar Sample Return Lander

Table 4. Block 1B Co-manifested Robotic Lunar Sample Return Lander Mass

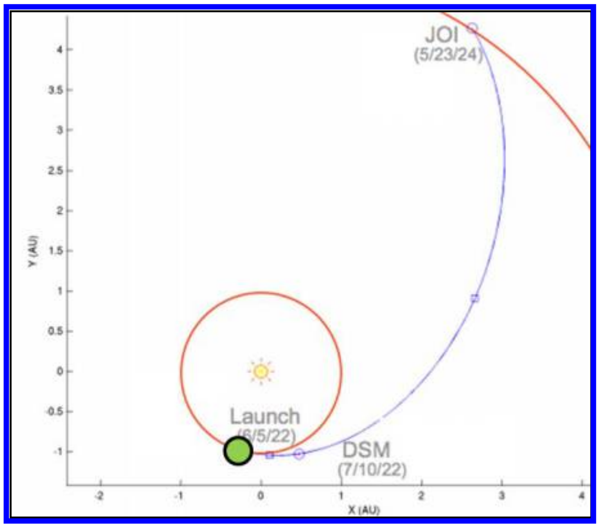
MISSION	LUNAR SAMPLE RETURN				
SLS BLK 1B CAP	39,000	KG TO TLI			
ORION	27,500	KG			
USA	4,095	KG			
PAF	800	KG			
LAUNCH MARGIN	1,178	KG			
LANDER TOT MASS	5,427	KG			
PAYLOADS	ASCENT		SURF		
	SAMPLES	3.5	SURF EQ	0	
	CANISTER	30	ROVERS	150	
	CREW	0	CONSUM	0	
	CABIN	0	SURF PWR	0	
	TOTALS	34	TOTALS	150	
STAGE INFO	ASCENT		DESCENT		
SINGLE ENG THRUST LBF	167		2,749		LBF
NUMBER OF ENGS	4		1		
PROPEL TYPE	NTO / MMH		NTO / MMH		
ISP	320.0		320.0		SEC
TANKS VOLUME M3	0.5		2.8		M3
PROP MASS FRACT	0.634		0.791		
INERT MASS	351		894		KG
PROPEL MASS	606		3,393		KG
TOT STAGE MASS	957		4,287		KG
TOT STAGE W PAYLD	990		4,437		KG

With each SLS Block 1B Orion launch a Gateway element will be delivered. Co-manifest missions with a Robotic lander can allow early Orion-to-surface sortie missions, even before the platform is complete. The lander delivers surface payloads and returns samples back to Orion or the Gateway. In Fig. 25 a Sample Return lander is pictured at Ascent Stage liftoff (rover not shown). (A mass statement is given in Table 4). A rover is used to collect samples, and is carried within the Ascent Stage structure during descent. Along with the 27.5 mt Orion and 4.1 mt USA, this storable propellant 2 Stage Lander masses about 5.4 mt, allowing a 1.2 mt launch margin. The lander is housed in the USA above the EUS.

Robotic stages based at the Gateway may be reusable and be used to validate propulsion, GN&C, thermal, imaging, telemetry and autonomy systems for later use in crewed landers operating out of the Gateway. Some landers will operate in a one-way descent only mode, carrying surface equipment, volatile extraction experiments, in-situ resource utilization (ISRU) elements, rovers and other systems.

As the sortie missions progress, the landers grow in size and capability and demonstrate more of the systems that will be utilized for larger crewed landers. Validating advanced technology will result in safer, more efficient and more productive crewed surface missions. These in turn, might build toward lunar base establishment, with the Gateway as an operational node and refueling hub for both crewed and robotic lander systems.

XII. SLS Launched Europa Clipper Mission: 2 Year Travel Time



Europa contains a subterrian water ocean, depicted in Fig. 28. A direct Earth-Jupiter transfer trajectory with a 2.0 year trip time is pictured in Fig 26. The Block 1 can inject 6.1-6.6 mt (depending on launch year) on this direct trajectory, which saves 4.5 years vs a 6.5 year Venus Earth Earth Gravity Assist (VEEGA) trajectory typical of an EELV mission. The NASA decadal survey (Ref. 2) rates the Europa mission as very high in priority.

Fig. 26. Two year Direct Earth to Jupiter Trajectory

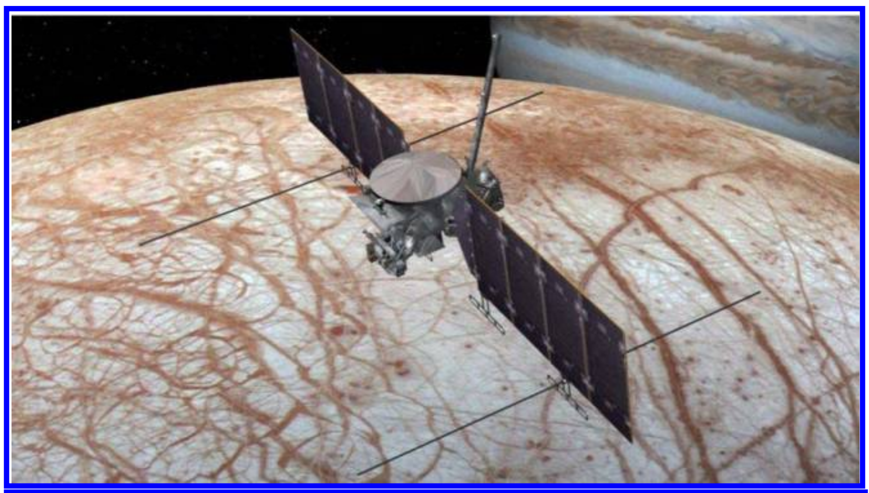


Fig. 27. Europa Orbiter

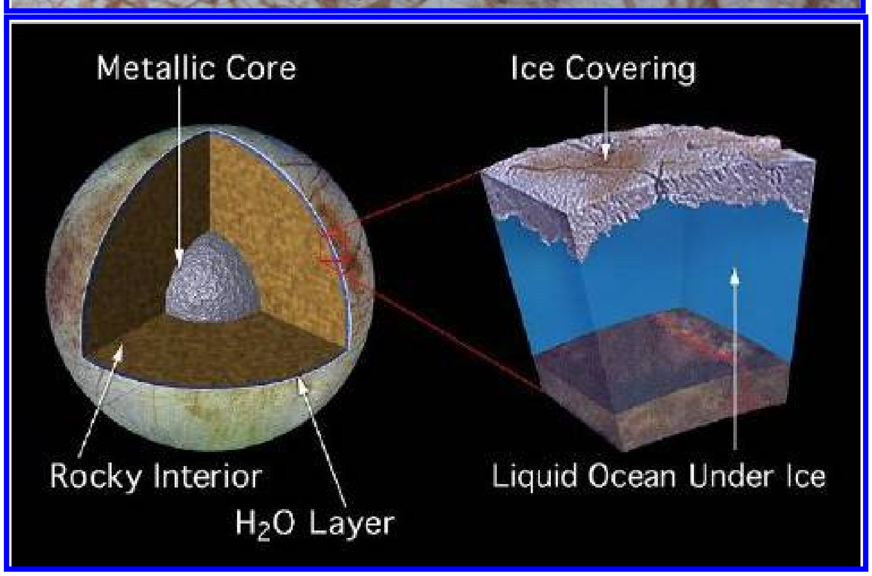


Fig. 28. Europa Interior Images Speculative

XIII. SLS Launched Titan Mission: 5 Year Travel Time

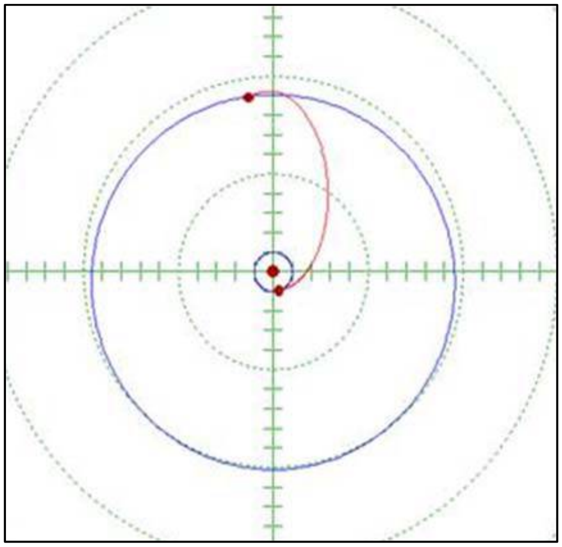


Fig. 29. Five year Direct Earth to Saturn Trajectory

The SLS Block 1B can inject 2.6 mt on a 5-year direct transfer (Fig. 29) to Saturn (C3=121 km2/s2). Titan has a diameter of ~1.5 that of Earth's moon; it's the second largest moon in the solar system and is the only moon known to have a dense atmosphere and the only object, other than Earth, for which evidence of stable bodies of surface liquid have been found (2004 Cassini-Huygens discovered hydrocarbon lakes). The atmosphere is largely composed of nitrogen and includes clouds of methane and ethane. Wind and rain create surface features that are similar to those on Earth, like shorelines and sand dunes. Composed primarily of water ice and rocky material, Titan is ~9.5 AU from the Sun. Fig. 30 includes a false-color synthetic radar map of a northern region taken by Cassini. Those regions on the surface that reflect very little radar may be hydrocarbon lakes.

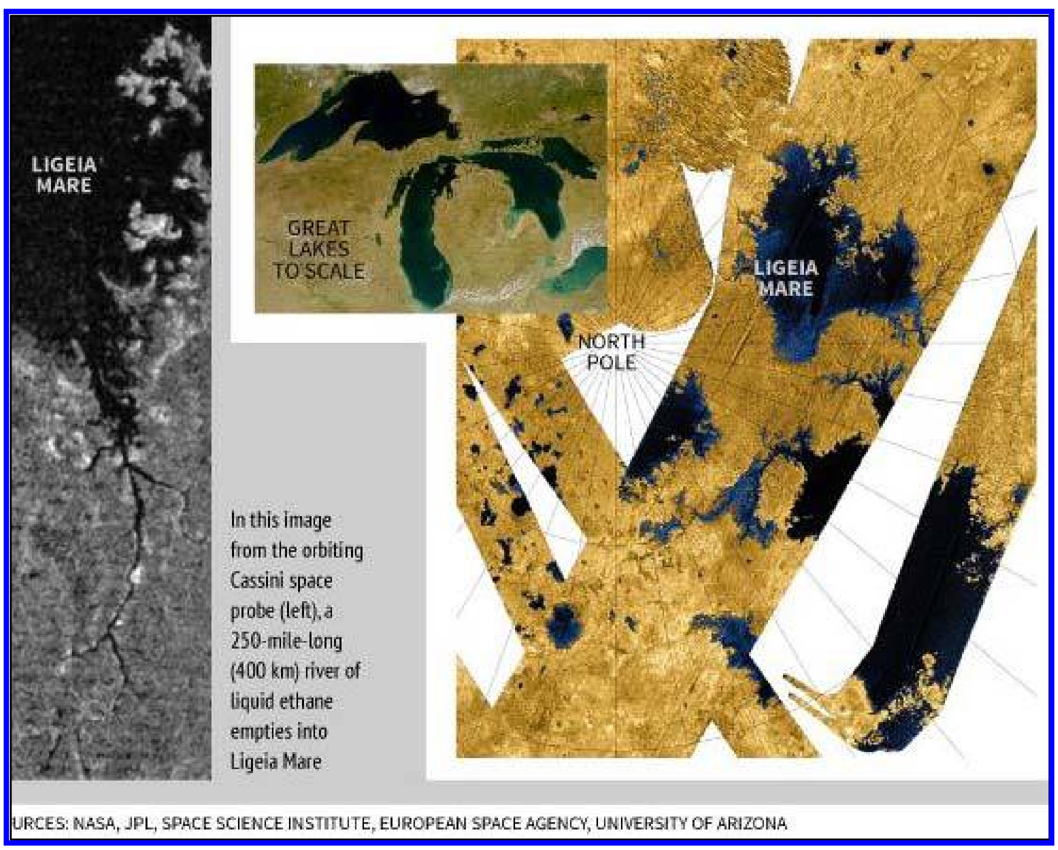


Fig. 30. Hydrocarbon Oceans on the Surface of Titan (NASA Cassini-Huygens)

XIV. SLS Launched Neptune / Triton Mission: 13 Year Travel Time

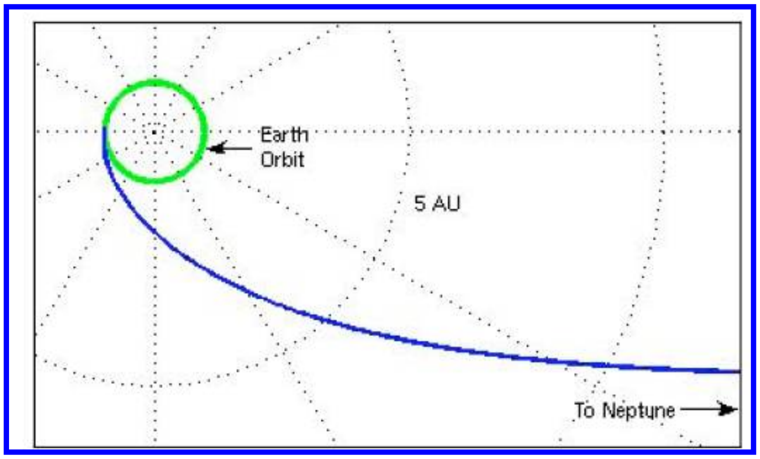


Fig. 31. Thirteen year Direct Earth to Neptune Trajectory

Triton, the largest of Neptune's eight known satellites, is different from all other icy satellites Voyager has studied. 3/4 the size of Earth's Moon, Triton (Fig. 32) circles Neptune in a tilted, circular, retrograde orbit (opposite to the direction of the planet's rotation), completing an orbit in 5.9 days. Triton shows evidence of a remarkable geologic history, and Voyager 2 images show active geyser-like eruptions spewing invisible nitrogen gas and dark dust particles several kilometers into space. In Fig. 31 a 13 year Earth-Neptune Direct trajectory is pictured.

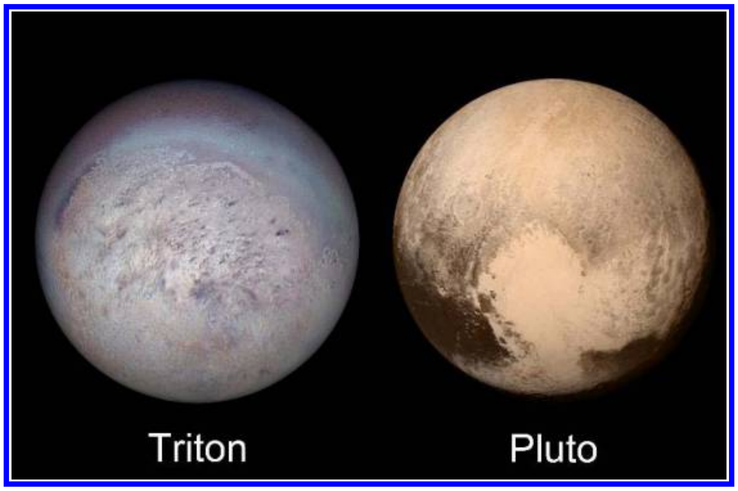


Fig. 32. Triton and Pluto

Fig. 33. Artists Impression of Triton and Neptune

XV. SLS Launched Two-Planet Saturn-Uranus Opportunity

A "once in 45 years" two-planet Saturn-Uranus trajectory opportunity will become available in 2023 (Fig. 34, Ref. 2). Departing in May 2023, this trajectory features a Saturn flyby in 2029, and a Uranus arrival in 2034. This unique mission permits sending first a probe to Saturn and then a probe-orbiter to Uranus, using an identical probe design for both planets. Total trip time to Uranus is 11.4 years. Uranus has several unique features including 1) Atmospheric dynamics due to extreme axial tilt, 2) Unusual magnetosphere geometry, 3) Unexplained energy balance and 4) Dynamically evolving rings and moons (Ref. 2). Uranus is an Ice-giant planet – ice giants have less hydrogen and helium and more "ices" (H_20, NH_3, CH_4, etc.) The majority of observed exoplanets are ice-giants, one of the great remaining unknowns of the Solar system. NASA's last encounter with Uranus was Voyager 2 in 1986. Uranus, like Saturn, has an extensive moon system (Fig. 35).

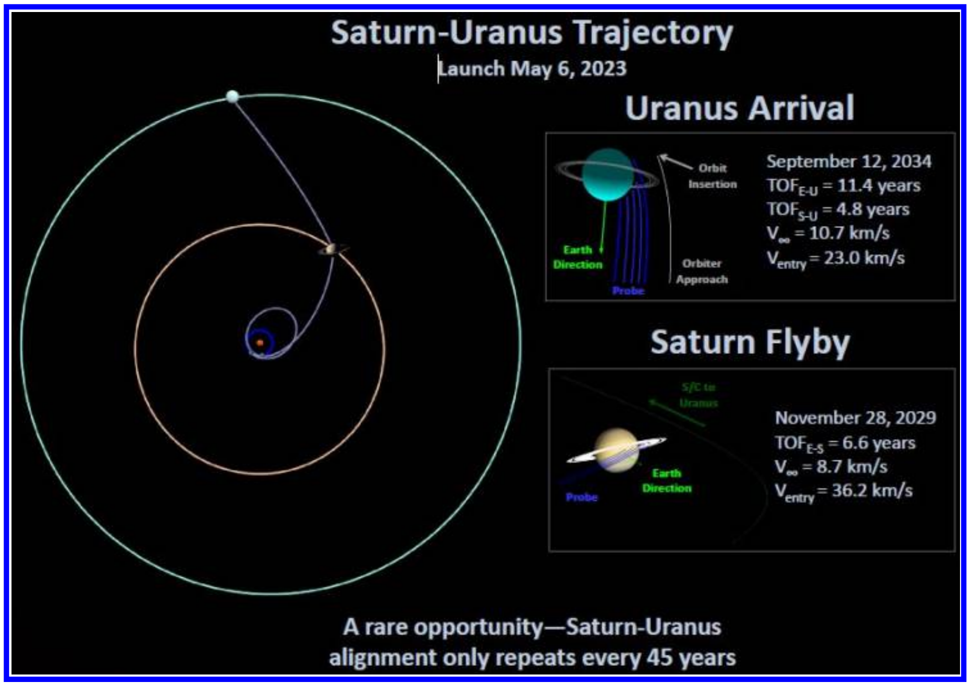


Fig. 34. Two Planet Saturn-Uranus Trajectory (Ref. 2)

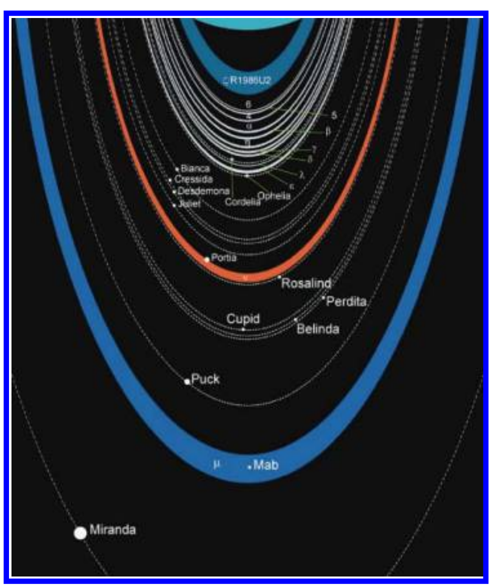


Fig. 35. Uranus' Extensive Moon System

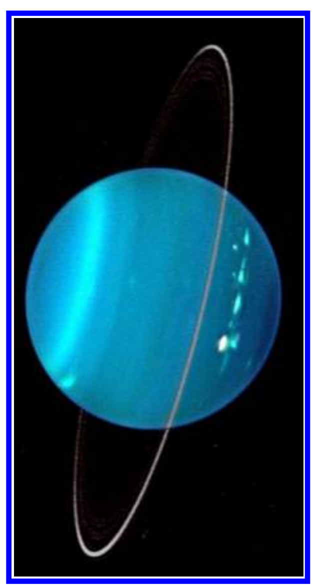


Fig. 36. Uranus' Unique Tilt

XVI. SLS Launched 8 m diameter Monolithic Optic Space Telescope

A 8 m diameter telescope concept, operating in Sun Earth Libration Point 2 (SEL2), is pictured in Fig. 37. The Block 1B can inject 38.0 mt to SEL2. Without SLS, multiple launches, complex folded optics, and/or on-orbit assembly would be the only alternatives for deploying space telescopes larger than ~7 m. This telescope would have 10 times the resolution of JWST and up to 300 times the sensitivity of the Hubble. A monolithic aperture is better than a segmented aperture; the JWST is using a segmented deployed mirror architecture only because it is the only way to launch a 6.5m aperture observatory with a 4.5 m diameter rocket. A monolithic mirror can achieve diffraction limited performance at a shorter wavelength than a segmented mirror with much less difficulty, complexity, cost and risk. An image representing a future servicing mission, including the Orion, launched by the SLS, is given in Fig. 38.

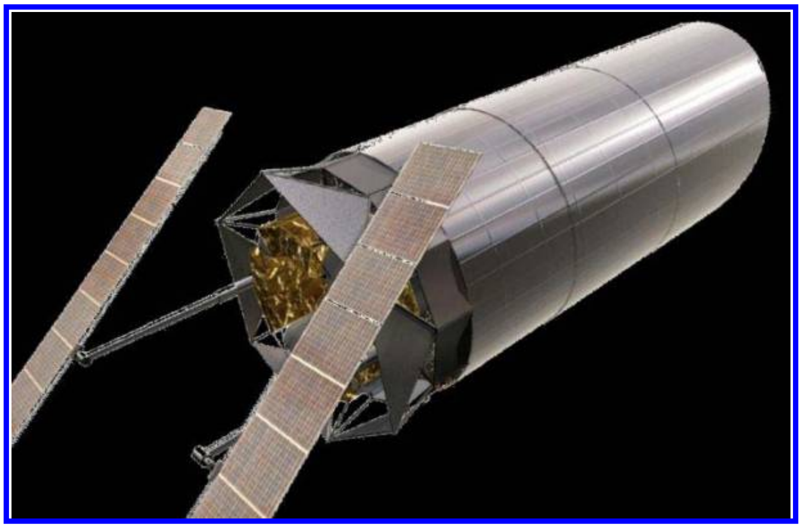


Fig. 37 Large Diameter Monolithic Optic Space Telescope Concept

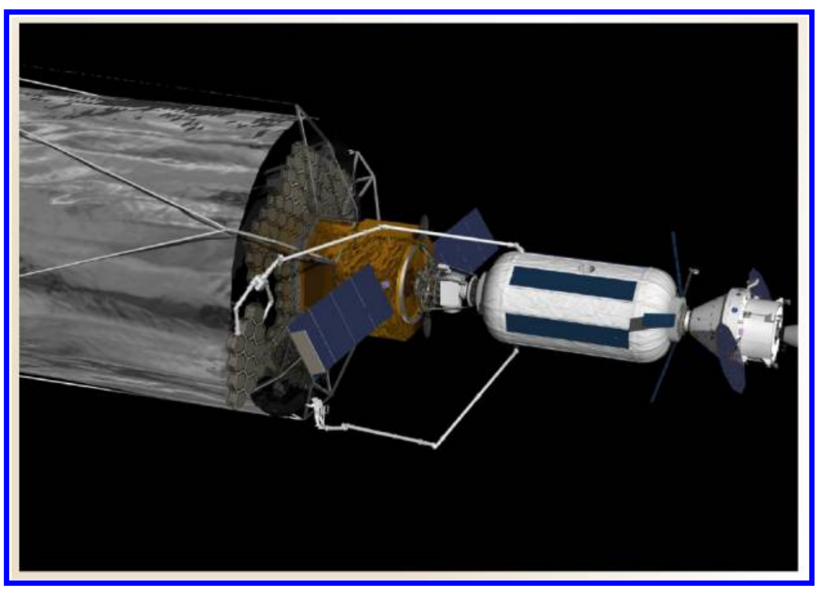


Fig. 38 Telescope Servicing Mission concept

XVII. SLS Launched Large Space Habitats

A very large single launch, long duration Crewed habitat concept, is pictured in Fig. 40. The Single launch Habitat concept offers several advantages vs a habitat made up of multiple elements launched separately and then assembled in space. The first advantage the large, single launch Habitat offers is that it enables an integrated and fully provisioned hab to be delivered to space in a single launch. The benefits of a single launch are lower cost and the habitat can be assembled and checked out on the ground rather than having multiple launches with on-orbit assembly, integration and checkout by astronauts in the weightless environment. Several studies estimate the mass for a 500 day Mars Habitat outfitting to be approximately 31-35 mt. An Inflatable concept, shown in its stowed configuration on the EUS, is shown in Fig. 39.

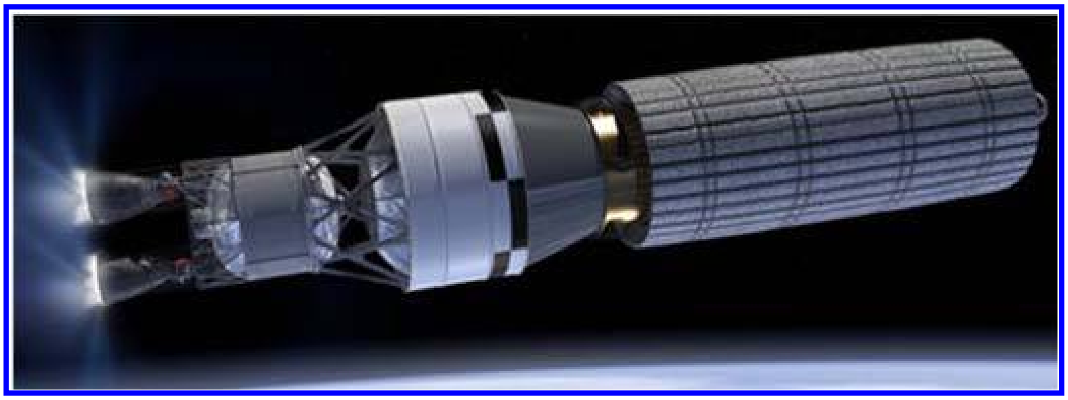


Fig. 39 Large Inflatable Crew Habitat in Stowed Configuration on SLS EUS

Fig. 40 Large Inflatable Crew Habitat Deployed

XVIII. SLS Launched Deep Space Transports

Solar Electric Propulsion (SEP) systems have two major advantages; they can make for very mass efficient transfer vehicles in Heliocentric space, and they can be reusable systems. SLS launched SEP deep-space transport systems may feature modular systems (arrays, thrusters, Power management and Distribution (PMAD)) which can be easily updated while retaining identical bus and structure elements. The benefits of Electric Propulsion (EP) for heliocentric flight to Mars include significant increases in payload or significant decreases in propellant mass vs lower Isp chemical propulsion systems. SEP's high Isp (>3500 sec, 10 times that of chemical propulsion) facilitates return to cislunar space for refueling and reuse after a Mars mission. The higher power SEP Mars Transfer Stages (MTS) for Mars cargo delivery (Fig. 41) can be scaled-up derivatives of lower power SEP stages allowing for technology maturation paths into the mid 2030's for element delivery to High Mars Orbit (HMO).

With the Block 1B's large fairing diam and length, the SEP-MTS can be integrated with a large Mars cargo payload, and both can be boosted to cislunar space by the EUS. There the MTS arrays are deployed and will, with its Mars cargo, spiral out of cislunar orbit to escape velocity, entering its heliocentric trajectory. SEP's disadvantage is low thrust; and requires Earth-Mars trip times that are significantly longer than high thrust systems. Non-crewed cargos that are not time critical may be flown with these SEP stages. A SEP-MTS may achieve the lifetime of a ComSat; on the order of 10-12 years, and will be designed for reuse on subsequent missions; returning to HEO for refueling. A 12 year operational life may allow 4 round trip missions for the Stage before it is expended. Several all EP ComSats are now operational, and the development of technologies for these commercial systems can be applied to these future SEP systems for heliocentric transfer. Also SEP systems may have other applications to exploration missions, including LEO to Moon cyclers, robotic solar system transfer stages (up to four AU) and other missions.

Fig. 41 Solar Electric Propulsion (SEP) Mars Transfer Vehicle

XIX. SLS Launched Mars Sample Return (MSR) Mission

MSR missions would benefit significantly from SLS. The SLS can launch enough mass so that the MSR's Mars Ascent Vehicle (MAV) can ascend directly from the surface to an Earth Return Trajectory, eliminating the need for a separate, dedicated Earth Return Vehicle (ERV). The MAV carries propellant for both ascent and Trans-Earth Injection (TEI). Unlike other MSR approaches, the MAV, carrying samples, would not have to rendezvous with a separate ERV in Mars orbit. This 'ascent-direct-to-Earth' approach would also eliminate the complex sample canister transfer maneuver (between the MAV and orbiting ERS) required of other MSR approaches. The SLS provides enough payload capacity to inject two direct-Earth-return MSR vehicles in a single launch. A MSR craft is shown on the Mars surface in Fig. 42 (all systems not shown). The MSR descent stage carries a rover and other equipment for multiple site sample retrieval. Once the Rover returns the sample sets they are stored in a sample canister in the ERV at the top of the MAV. The dual MSR spacecraft are boosted to TMI by a single Block 1B (Fig. 44) ; after a 10-11 month coast, the vehicles separate and aerobrake before a propulsive landing. Each MSR lands (in addition to the MAV), 195 kg of surface payloads; among these are a rover that gathers samples from multiple sites. After the surface mission, the MAV ascends to orbit (Fig. 43), and fires again to achieve Trans-Earth injection (TEI) velocity (C3=10 km2/s2). Earth return requires 9-11 months. The Block-1B can launch these two spacecraft while retaining a launch margin of about 4 mt (Mars easy year) and 2 mt (average year). Advantages to sending two MSRs include redundancy in case of failure, and sample collection from two very different regions, ensuring a diverse sample set. This single launch, dual direct Earth return MSR mission compares to other recent [Ref 2] MSR plans, which require 3 EELV launches and 9 years of mission time. SLS launch enables samples sooner, at less total cost, with a higher probability of success, while eliminating the procurement of a separately emplaced ERV.

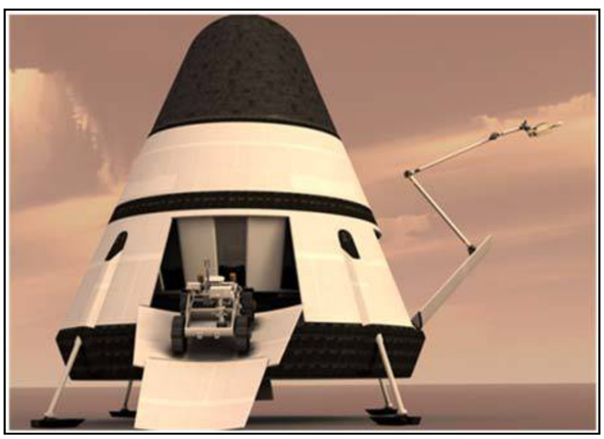


Fig. 42 SLS Launched Direct Return to Earth MSR Vehicle

Fig. 43 MSR Lift-off

Fig. 44 SLS MSR Dual Launch

XX. SLS Launched Mars Crew Lander

A Crewed Mars Lander is shown in Fig. 45 (left, aeroshell not pictured). The Descent stage and Mars Ascent Vehicle (MAV) feature LO2/CH4 (methane) propellant; for the current NASA Evolvable Mars Campaign planned for the 2030's, an In-situ Propellant Production Plant (ISPP) is pre-emplaced on the surface previous to the arrival of the Crew. ISPP allows the MAV to be launched with its oxidizer tanks empty, significantly reducing its launch mass. The ISPP separates oxygen out of the atmosphere (95% CO2). The extracted and liquefied LO2 will be used with Earth supplied methane. After crew boarding in Mars Orbit, the Lander descends, decelerating aerodynamically; after aero-descent, the aerobrake is jettisoned and descent engines provide the remaining 800 m/s of dV required for terminal descent. On the surface, the MAV's oxidizer tanks are filled with in-situ provided LO2. Cryocoolers maintain the propellants for the duration of the surface stay. Later, the MAV will ascend to orbit, and rendezvous with the Earth-Mars Transfer Stage. While on the surface the Crew work out of a separate Surface Habitat, landed with a Mars Cargo Lander, which shares a common aeroshell and descent stage with the Crewed System. The 2.5 year Crewed Mars missions require several large, single piece elements, including, along with these Mars Crew and Cargo landers, the Earth-Mars transfer stage and a transit Crew habitat (Section 17). The large payload volume and lift capability provided by the Block 1B and Block 2 Configurations would enable these missions. The MAV ascends to a 400 km circular Low Mars Orbit (LMO). Another option (not shown) has the MAV ascend to a 500 km by 1 SOL (24 hour) High Mars Orbit (HMO), which requires more ascent propellant, producing a heavier lander and a heavier Entry, Descent and Landing (ED&L) system. The lander pictured is sized for the lower orbit, to ease the ED&L requirement; it is shown on lift-off in Fig. 45 (right).

Crewed Mars Lander
Ascent Stage: 2 Stage O2/CH4 Pump Fed 365 Isp
Ascent Payload: Crew Cabin
Ascent: Low Mars Orbit (LMO) (400 km circ)
In-situ O2 Provided on Surface
Descent Stage: O2/CH4 Pump Fed 365 Isp
Payloads: Surf Exploration Sys, EVA suits, Rover, Surf Power Sys, ISRU Elements, Consumables
Aeroshell: Aero descent, Jettisoned , not shown

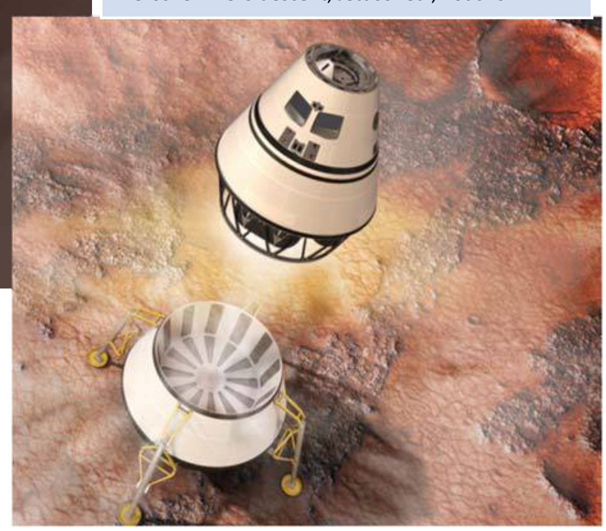


Fig. 45 SLS Launched Mars Crew Lander (LMO option, Landed left, Lift-off right)

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3. Christensen, P., May, L., NASA, *Mission Concept Study Planetary Science Decadal Survey, MSR Orbiter Mission* (Including Mars Returned Sample Handling), March 2010