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A Flexible Lunar Architecture for Exploration (FLARE) supporting NASA's Artemis Program

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ABSTRACT

The Flexible Lunar Architecture for Exploration (FLARE) is a concept to deliver four crew to the lunar surface for a minimum of seven days and then return them safely to Earth. FLARE can be implemented whenever the component vehicles are operational. FLARE was developed as an alternative to NASA's Human Landing System (HLS) reference architecture from the Design Analysis Cycle (DAC) #2 created in 2019. The DAC2 guidelines required utilization of the Gateway vehicle in a Near- Rectilinear Halo Orbit (NRHO). Instead, FLARE chooses a Low Lunar Frozen Polar Orbit (LLFPO) for lunar rendezvous of components, and an optional Gateway vehicle. The LLFPO provides a stable orbit that overflies the south pole every 2 h, ensuring easy access to the lunar surface for surface aborts with a much lower propellant requirement than NRHO. The minimum FLARE concept uses one Space Launch System (SLS) launch, one Orion, one European Service Module (ESM), and one human lander (launched on commercial vehicle(s)). FLARE adds the SpaceTug, based upon the mature and successful ULA "Common" Centaur Upper Stage vehicle, with modifications to create an Earth-Moon transfer vehicle. In the FLARE baseline mission, the SpaceTug provides propulsion needed to return the Orion + ESM from LLFPO to Earth. The SpaceTug also provides propulsion to deliver the separate human lander components – the Descent Element (DE) and the Ascent Element (AE) - from Low Earth Orbit (LEO) to LLFPO. The SLS Block 1 then launches the Orion + ESM and completes a rendezvous with the mated DE + AE components in LLFPO. FLARE offers optional phases beyond the baseline mission. The SpaceTug can deliver components of the planned Gateway, including the Power and Propulsion Element (PPE) and the Habitable and Logistics Outpost (HALO), to LLFPO. FLARE provides an option to deliver precursor equipment to the lunar surface to enhance and extend the human mission. With these components, including an inflatable habitation module and airlock, individual crew mobility vehicle(s), an In-Situ Resource Utilization (ISRU) demonstration, and science and technology experiments, the crew can explore and conduct science on the lunar surface for up to 14 days.

1. Introduction

The Flexible Lunar Architecture for Exploration (FLARE) is a practical methodology to deliver four crew to the lunar surface for 7–14 days then safely return them to Earth. FLARE can be implemented whenever the component vehicles are ready for launch. FLARE was developed as an alternative to NASA's Human Landing System (HLS) reference architecture from the Design Analysis Cycle (DAC) #2 [1]. Their goal was to develop an architecture delivering two crew to the lunar surface by 2024 using only a commercial human lander, the SLS Block 1, the Orion, the European Service Module (ESM), and Gateway in a Near Rectilinear Halo Orbit (NRHO). The DAC2 team selected NRHO for Orion due to documented propulsion limitations of the ESM [2]. The Orion + ESM alone has insufficient propellant to deliver the crew to any circular Low Lunar Orbit (LLO) and safely return them to Earth. Although outside of their documented constraints, DAC2 discussed the requirement for a "Transfer Element" (TE) as an additional vehicle to move assets from NRHO to LLO and back. The HLS team was unsuccessful at closing an architecture within their constraints due to the large mass needed (both propellant and human logistics supplies) for transfer between the lunar surface and NRHO [1]. The FLARE expands upon the TE concept by modifying a mature, upper-stage vehicle into a "SpaceTug" to transfer mass between Low Earth Orbit (LEO) and LLO.

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2. FLARE concept

FLARE is supported with a technical analysis of multiple factors, including mass and change in velocity (ΔV) calculations including crew, cargo, and propulsion systems. FLARE develops a plan for launch and mating of necessary components in Earth and lunar orbit. FLARE provides a reference design for the SpaceTug and a human lander, including both the pressurized Ascent Element (AE) and a "common" Descent Element (DE) capable of delivering either crew or cargo to the lunar surface. Payload volumetric evaluations are considered within existing launch vehicle fairings, and also for crew logistics on the lunar surface (within both the lander and in an optional inflatable habitation module). A lunar surface concept of operations is presented for Extra-Vehicular Activity (EVA) traverses and crewed exploration activities with a 7–14 day campaign. FLARE also provides an optional reference concept for an individual crew mobility device, called the "Lunar ATV" (LATV), to support extended surface traverses for early human surface missions. Similar to the SpaceTug function for orbital transfers, the LATV provides lunar surface transfers for science and crew.

The FLARE utilizes launches on various mature, Commercial Launch Vehicles (CLV) to lift uncrewed components to LEO. The FLARE launch schedule requires a nine-week period that integrates ULA, SpaceX, and NASA launch pad availability with predicted boil-off rates for vehicle cryogenic propellants. **The key to FLARE is the SpaceTug (based on an existing upper-stage vehicle) that is launched on a CLV.** FLARE utilizes a Low Lunar Frozen Polar Orbit (LLFPO) with inclination of 86.5° at an altitude of 100 km over the lunar surface. SpaceTugs transfer assets, such as the human lander, between Low Earth Orbit (LEO) and LLFPO. A single SLS Block 1 lifts crew in the Orion + ESM to Trans-Lunar Injection (TLI), and the ESM provides the propulsion to insert the Orion + ESM in LLFPO. A dedicated "Return SpaceTug" (RST), delivered to LLFPO before the crew launches, provides the necessary propulsion for crew return to Earth.

FLARE does not require the Gateway components of Power & Propulsion Element (PPE) [3] **or the Habitation and Logistics Outpost (HALO)** [4]**;** however, these elements are available as optional phases in FLARE. FLARE also provides another optional phase for a precursor cargo mission, launched on a SpaceX Falcon Heavy (FH). The precursor mission lands directly on the lunar surface with an inflatable habitation module, LATV(s), a science trailer, and technology demonstrations and experiments.

3. FLARE phases

FLARE sequence naming convention is XXY, where the XX identifies the vehicle element (see Appendix A Acronyms), and Y is an incrementing counter for those vehicle elements. For SpaceTugs, the naming convention is repeated, e.g. Tug1DE1 (which identifies the first SpaceTug that pushes the first Descent Element from LEO to LLFPO). Generic vehicle discussion uses their acronym only without a counter (Y). When discussing the launch vehicle for each element (see Table 2 and Table 3), the designated CLV is listed first, e.g. A5_AE1 (identifies the ULA Atlas V that lifts the AE1 to LEO).

The FLARE sequence is divided into five Phases A-E (with each subdivided into subphases a-b):

A. Deliver equipment to create the Gateway in LLFPO (Optional)
B. Deliver lunar surface precursor equipment (Optional)
C. Deliver vehicles to LLFPO for crewed mission support (Required)
D. Deliver crew to LLFPO, then lunar surface, then to LLFPO (Required)
E. Return crew to Earth (Required)

A summary of the FLARE mass budget and number of vehicles is provided in Table 1.

Table 1
FLARE mass budget and number of vehicles needed.

Item	Dry (mt)	Prop (mt)	Tot (mt)	ISP	Required	Optional
SpaceTug	2.75	20.05	22.80	450.50	5	2
PPE (Gateway)	8.00	0.00	8.00	n/a	0	1
HALO (Gateway)	8.00	0.00	8.00	n/a	0	1
Common DE	2.50	9.00	11.50	450.50	1	1
AE (crewed)	4.00	4.40	8.40	320.00	1	0
Precursor Surface	4.50	0.00	4.50	n/a	0	1
Crew (0.1 mt each)	0.40	0.00	0.40	n/a	4	0
EVATools (Descent)	0.10	0.00	0.10	n/a	1	0
Samples (Ascent)	0.10	0.00	0.10	n/a	1	0
Orion CM (crewed)	9.30	1.10	10.40	320.00	1	0
ESM	6.90	8.60	15.50	320.00	1	0

3.1. Minimum required FLARE phases

The minimum sequence of steps for the FLARE crewed mission (to support four crew on the lunar surface for 7 days living in the AE) is shown in Table 2, with a graphical description of required phases provided in Figs. 1–5. These maneuver burn and margin calculations are provided in Appendix B Tables B4-B10.

3.2. Optional FLARE phases

The sequence of steps for Optional FLARE Phases is shown in Table 3, with a graphical description of optional phases provided in Figs. 6–9. These maneuver burn and margin calculations are provided in Appendix B Tables B1-B3.

Optional Phase A assembles Gateway in LLFPO. NASA has existing contracts for commercial launch of the PPE and HALO, former known as "Mini-Hab" (MH), elements in NRHO [3,5]. NASA then decided on a single launch of these two integrated elements in 2023 [6]. With FLARE moving the Gateway elements from NRHO to LLFPO, vehicle thermal and power systems may need to be modified (based on the assumed stack attitude timeline in the lower lunar orbit). The PPE could provide a valuable communications relay for lunar surface operations, and HALO could provide unpressurized docking adaptors for SpaceTugs. Without Phase A, Orion + ESM must dock directly to the AE1 of the human lander (composed of AE1+DE2). This is exactly the Apollo program approach, and requires compatible docking adaptors on each vehicle (which are different from the requirements for each vehicle docking to Gateway). Within FLARE, each Gateway element is launched separately to LEO on an Atlas V (A5). Each SpaceTug is launched to LEO on a SpaceX Falcon (F9). Thus, four additional CLV launches are required to construct Gateway in LLFPO. Each SpaceTug conducts an autonomous Rendezvous, Proximity Operations, and Docking (RPOD) with the Gateway element in LEO and then transfers it to LLFPO.

Optional Phase B provides precursor hardware to the lunar surface, including an inflatable surface habitat and inflatable airlock (to support four crew living on the lunar surface up to 14 days living in the habitat). The precursor hardware also includes human logistics and EVA support, individual human mobility vehicle(s), science and technology experiments and deployed communication/navigation satellite(s). Phase B is necessary for developing early lunar surface sustainability as a human "field station". Phase B requires one Falcon Heavy (FH) launch to deliver a "common" Descent Element (named DE1) and 4.5 mt cargo payload of precursor hardware to TLI. The DE1 then deploys communications & navigation satellite(s), performs the lunar descent, and delivers the payload directly to the lunar surface.

Table 2
FLARE minimum required phases (C, D, E).

Seq	Phase	Vehicle	Payload	Description	ΔV (km/s)	Margin (mt)
1	C1a: AE1 to LEO	A5_AE1	AE1	Launch AE1 to LEO		
2	C1a: AE1 to LEO	F9_Tug1AE1	Tug1AE1	Launch Tug1AE1 to LEO		
3	C1a: AE1 to LEO			Assemble AE1 to Tug1AE1 in LEO		
4	C1b: AE1 to LLFPO	Tug1AE1	AE1	Push AE1 w/TugAE1 to LLFPO	4.23	0.80
5	C2a: RST to LEO	F9_Tug1RST		Launch Tug1RST to LEO		
6	C2a: RST to LEO	F9_RST		Launch RST to LEO		
7	C2a: RST to LEO			Assemble RST to Tug1RST in LEO		
8	C2b: RST to LLFPO	Tug1RST	RST	Push RST w/Tug1RST to TLI, D/O Earth	2.40	0.15
9	C2b: RST to LLFPO	RST	RST	Push RST w/RST to LLFPO	1.88	12.15
10	C3a: DE2 to LEO	F9_Tug1DE2	Tug1DE2	Launch Tug1DE2 to LEO		
11	C3a: DE2 to LEO	A5_DE2	DE2	Launch DE2 to LEO		
12	C3a: DE2 to LEO	F9_Tug2DE2	Tug2DE2	Launch Tug2DE2 to LEO		
13	C3a: DE2 to LEO			Assemble DE2, Tug1DE2, Tug2DE2 in LEO		
14	C3b: DE2 to LLFPO	Tug1DE2	DE2	Push DE2+Tug2DE2 w/Tug1DE2 to TLI	1.80	0.15
15	C3b: DE2 to LLFPO	Tug2DE2	DE2	Push DE2 w/Tug2DE2 to LLFPO. Mate AE1 with DE2 in LLFPO	2.48	4.05
16	C4a: Crew to TLI	SLS	OrionESM	SLS to TLI (4 crew in Orion + ESM)		
17	C4b: Crew to LLFPO	2 wks after DE2	OrionESM	Push w/ESM to LLFPO	1.00	1.50
18	D1a: Refuel demo			Transfer LOX/LH2 between vehicle(s)*		
19	D1b: Crew to surface	DE2	AE1	Crew (4) push w/DE2 to LS	2.25	0.26
20	D2: Surface Campaign			Crew(4) on surface of Moon 7d		
21	D3a: Crew from surface	AE1	AE1	Crew (4) push w/AE1 to LLFPO	2.01	0.18
22	E1a: Crew in LLFPO	RST	OrionESM	Assemble RST to Orion + ESM, Push w/RST to TEI, RST deorbit Earth	1.13	0.14
23	E1b: Crew to Earth	ESM	Orion	Push to EI with Orion + ESM, all descend Earth	0.14	0.70

*stretch goal based upon SpaceTug capability.

Table 3
FLARE optional phases (A, B).

Seq	Phase	Vehicle	Payload	Description	ΔV (km/s)	Margin (mt)
1	A1a: PPE to LEO	A5_PPE	PPE	Launch PPE to LEO		
2	A1a: PPE to LEO	F9_Tug1PPE	Tug1PPE	Launch Tug1PPE to LEO		
3	A1a: PPE to LEO			Assemble PPE & Tug1PPE in LEO		
4	A1b: PPE to LLFPO	Tug1AE1	PPE	Push PPE w/Tug1PPE to LLFPO	4.23	1.05
5	A2a: HALO to LEO	F9_Tug1HALO	Tug1HALO	Launch HALO to LEO		
6	A2a: HALO to LEO	A5_HALO	HALO	Launch Tug1HALO to LEO		
7	A2a: HALO to LEO			Assemble HALO to Tug1HALO at LEO		
8	A2b: HALO to LLFPO	Tug1HALO	HALO	Push HALO w/Tug1HALO to LLFPO (GW)	4.27	0.95
9	B1a: DE1+PC1 to TLI	FH_DE1+PC1	DE1+PC1	Launch DE1+PC1 to TLI		
10	B1b: DE1+PC1 to LS	DE1	0.00	Push DE1+PC1 to lunar surface	2.90	1.18

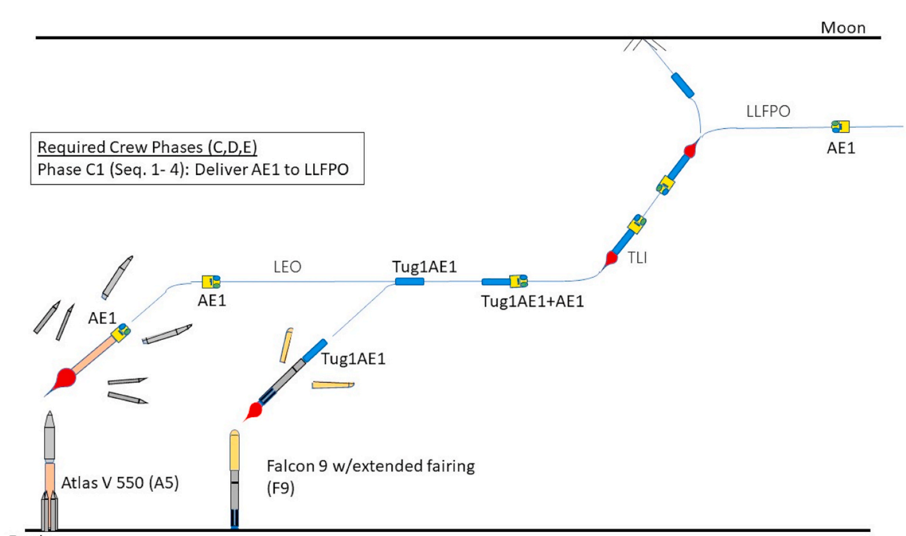

Fig. 1. Required Phase C1 – delivery of human lander AE1 to LLFPO

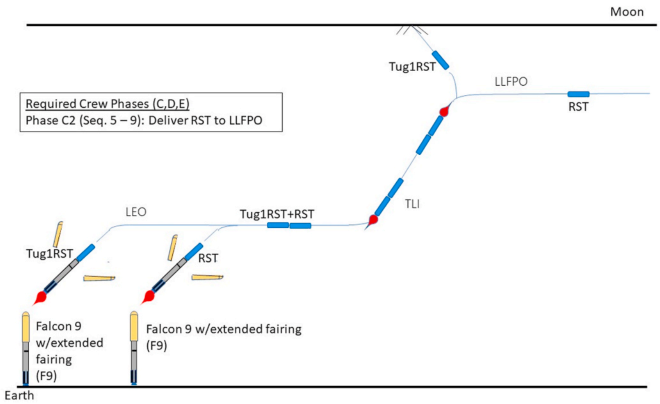

Fig. 2. Required Phase C2 – delivery of RST to LLFPO

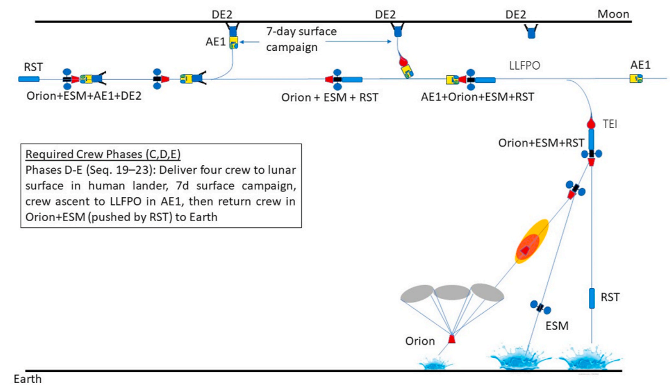

Fig. 5. Required Phases D, E – Crew to/from Lunar Surface (7d), then Return to Earth.

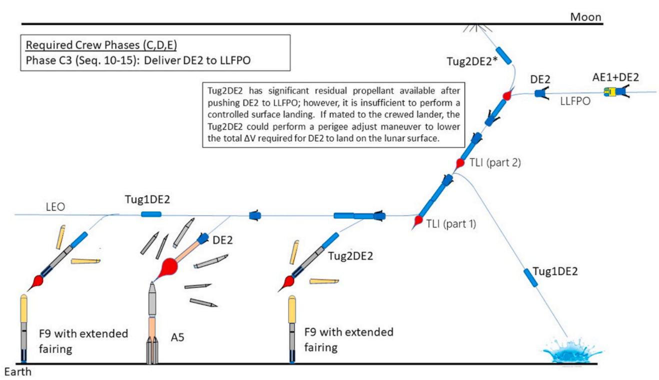

Fig. 3. Required Phase C3 – delivery of human lander DE2 to LLFPO

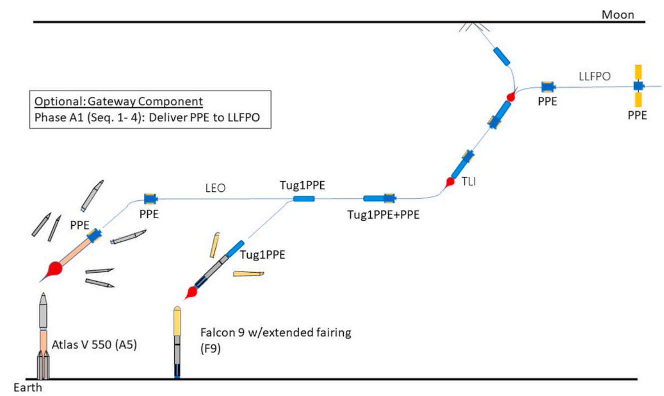

Fig. 6. Optional Phase A1 – delivery of PPE to LLFPO

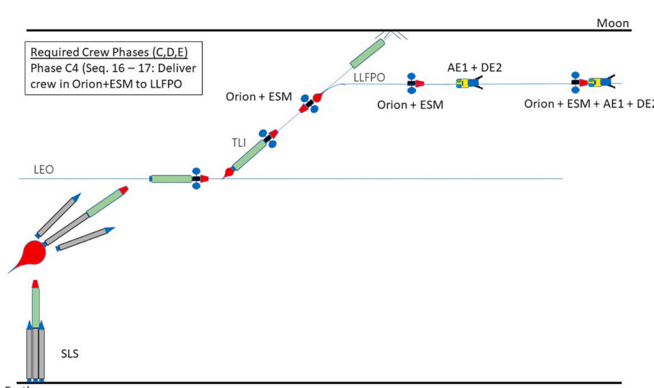

Fig. 4. Required Phase C4 – delivery of crew in Orion + ESM to LLFPO

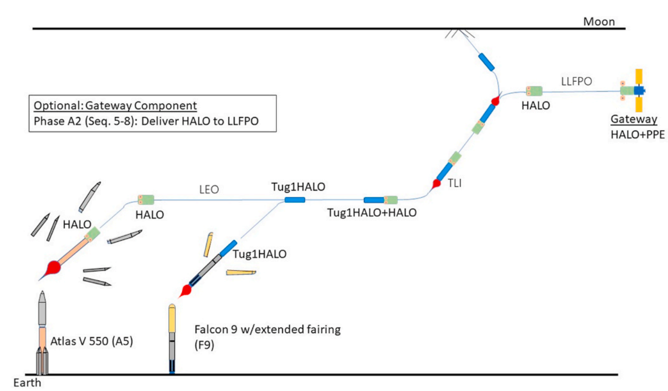

Fig. 7. Optional Phase A2 – delivery of HALO to LLFPO

4. FLARE considerations

4.1. Launch and mission schedule

Despite the coronavirus global pandemic in 2020 closing NASA centers [7], NASA continues to press towards a human lunar mission in 2024, even developing a framework of principles for lunar development with the "Artemis Accords" [8]. There is concern, however, that the Congressionally required budget to support this launch date will not be provided to NASA [9]. FLARE supports the Artemis Program with a launch sequence applicable whenever the vehicles are operational. The minimum crewed mission (FLARE Phases C, D, E) to the lunar surface requires the launch of five Falcon 9 (F9) for SpaceTugs, two Atlas V (A5) for human lander components (AE and DE), and one SLS for crew inside Orion + ESM. All these launches occur within a 9-week period (see Fig. 10). The launch timing of these phases is critical since the SpaceTug and lander DE vehicles use LOX/LH_2 as propellant that boils away in orbit. FLARE calculates the required timing and sequence to provide necessary margin in ΔV calculations, including the loss of LOX/LH_2 from "boil-off" in orbit (see Appendix B Tables B4-B10). The SLS is the final launch in this 9-week period, and its launch window must include considerations for lunar polar plane rotation in LLFPO to minimize ΔV for TEI return, and also lunar surface lighting during the crew lunar surface campaign. In the minimum mission, all four crew spend seven days on the lunar surface living in the AE1 (see Fig. 5). Adding optional Phase B, all four crew would live for 14 days in the inflatable habitat provided by the precursor cargo mission (see Fig. 9). Phase B requires

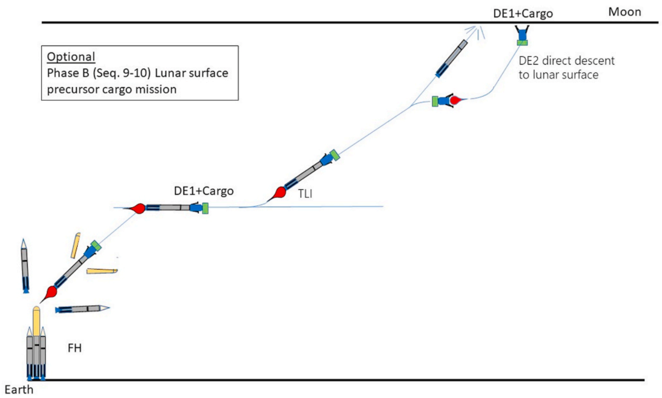

Fig. 8. Optional Phase B – delivery of precursor cargo to the lunar surface.

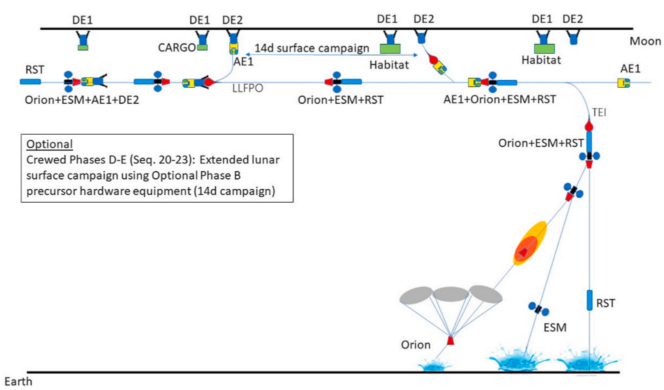

Fig. 9. Required Phases D, E using Optional Phase B – Crew to/from Lunar Surface (14d) then Return to Earth.

one additional SpaceX Falcon Heavy (FH) launch to carry a "common" DE and payload to TLI. The DE1 vehicle then lands the 4.5 mt payload on the lunar surface. Note that all optional phases (A, B) occur before the 9-week sequence leading to the launch of the crew (see Fig. 10).

The SLS launches Orion + ESM to TLI, and the ESM then provides the required 1.0 km/s ΔV for LLO insertion and RPOD to the AE1 waiting in LLFPO (or to Gateway if Option A is implemented). This consumes most of the ESM propellant. The FLARE provides a SpaceTug in LLFPO, named the Return SpaceTug (RST), with sufficient propellant to return the Orion + ESM and crew back to Earth (including boil-off margin). The RST is launched 1.5 months before SLS. After delivery to LLFPO, the RST experiences 56 days of propellant boil-off yet retains sufficient ΔV to return the crew to Earth (including a maximum 14-day surface duration with Optional Phase B). All of the necessary elements for lunar descent, ascent, and return are therefore in place before the crew launches in Orion.

Recognizing the risk of reliance on a SpaceTug in LLFPO for crew return **to Earth**, FLARE considered a possible alternative of this crewed sequence using a SpaceTug to push the Orion + ESM **to LLFPO**. This allows the Orion + ESM in LLFPO to provide the ΔV for crew return to Earth (no RST required). Preliminary analysis suggests this approach is possible but requires assembly of a large, multi-stage vehicle in LEO composed of three SpaceTugs mated to the Orion + ESM to conduct TLI and LOI (see Appendix B Tables B12-B17 for calculations using SpaceTugs from LEO). A more practical alternative is launching two full SpaceTugs in a High Earth Orbit (HEO) on commercial heavy lift rockets, where they then mate with Orion + ESM. Note the use of SpaceTugs to push Orion + ESM to the Moon means FLARE can be implemented even if the SLS Block 1 is unavailable. All of the components can be launched on commercial rockets and the SpaceTugs push the Orion + ESM to LLFPO. The ESM then returns Orion to Earth. The authors baseline the FLARE sequence using the SLS Block 1 for Orion + ESM launch, and the RST for crew return to Earth, but they demonstrate that the FLARE sequence is flexible enough to adapt to other scenarios.

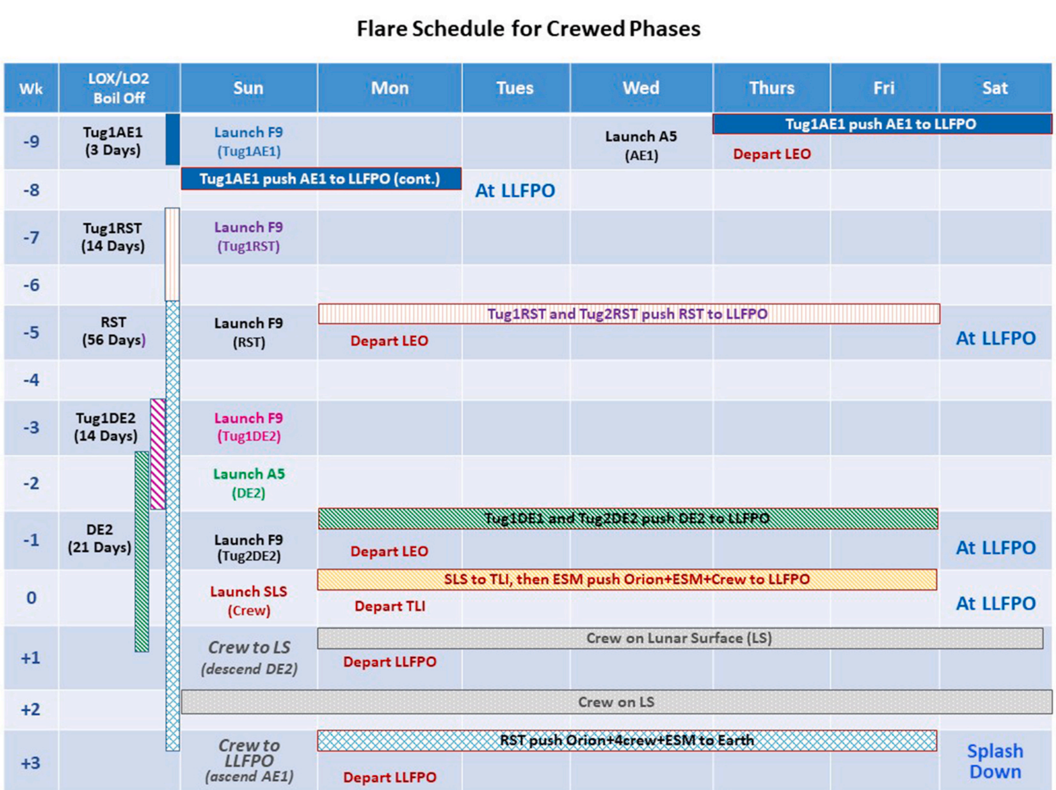

Fig. 10. Flare launch schedule for crewed phases.

4.2. NRHO mission operations limitations

The HLS DAC2 evaluation guidelines required Gateway in a NRHO [1]. The NRHO is highly elliptical lunar orbit with a period of 6–8 days. It has been well studied as a potential staging orbit for deep space exploration using the Earth-Moon-Sun L1 and L2 Lagrange Points [10]. Altitudes above the lunar surface can vary from 2,000 to 75,000 km during each NRHO period [2]. A specific NRHO with a 9:2 lunar synodic resonant, chosen for the NASA HLS DAC2, places apolune over the lunar south pole. This orbit is favored for its low orbital maintenance maneuver requirement and infrequent eclipse by the Earth and Moon [11].

In a representative 6.5-day period, southern NRHO, a spacecraft spends the bulk of every week at the far end of the orbit (relative to the Moon) with only 1–2 days near the lunar surface. For a brief time the orbiting vehicle is difficult to reach from the lunar surface (passing at high velocity nearly 2000 km above the surface) and for the majority of the week the orbiting vehicle is impossible to access (>30,000 km away). NRHO thus forces lunar human mission designers into one of two difficult choices. First, a short-duration "grab and go" mission to descend to the lunar surface from Orion + ESM in NRHO, consisting of a very brief surface exploration campaign (<12 h), and then ascend to the Orion + ESM. This short mission provides little surface science opportunity or surface infrastructure development, and requires an extremely long crew wake period (>24 h to prepare in orbit for descent, descend, explore, ascend, and dock with Orion). The second option is a weeklong mission on the lunar surface; however, the crew has no ability to rapidly abort to Orion + ESM once they are on the surface for more than 4 h (since Orion + ESM are too far away and moving rapidly further from the Moon). A few interim abort-to-orbit opportunities exist, but the crew must survive in the ascent vehicle for 2–4 days while conducting a rendezvous with the distant Orion + ESM. For the nominal mission, the Orion + ESM overflies the landing site 6.5 days after landing, and the crew can ascend and rendezvous in NRHO. Adding a surface habitation module and pre-positioned infrastructure components (e.g. crew mobility devices, EVA tools, and contingency consumables) increases safety and reduces risk for the longer surface missions while Orion + ESM is too distant for rendezvous.

4.3. NRHO science limitations

Science opportunities from the NRHO include Earth observations (outside the Earth's magnetosphere), heliophysics, fundamental physics, and microgravity or radiation studies of biological and physical systems [12,13]. The first instruments selected for Gateway observe space weather and monitor the Sun's radiation environment [14]. With an orbiting spacecraft in NRHO, however, the vehicle is so distant from the lunar surface that telescope observations from Earth exceed resolutions possible from likely equipment available viewing from a window, or externally attached to HALO. Note the current Gateway HALO concept provides little volume for internal or active, external science instrumentation and experiments. HALO has no science airlock or robotic arm to mount and remove experiments for return to Earth, although perhaps the addition of the ESA ESPRIT module would add these features after the initial Gateway vehicle arrives in orbit [12]. NASA has announced that Gateway is no longer a required component of the Artemis architecture [15], so early science may be limited to capabilities in Orion or the human lander only.

4.4. NRHO ΔV limitations

The SLS B1 vehicle with the ICPS can deliver the 26 mt Orion + ESM to TLI [16], but the 8.6 mt of propellant and oxidizer in the ESM [17] delivers a maximum total ΔV of only 1.25 km/s [2]. This ESM therefore provides insufficient delta-V (ΔV) to both insert the Orion + ESM into a 100 km circular Low Lunar Orbit (LLO), requiring a ΔV = 0.952 km/s, and return it to Earth from LLO, requiring a ΔV = 1.256 km/s [18]. It might be possible for Orion + ESM to return from LLO to Earth alone (with little excess margin); however the Orion + ESM must then be delivered to LLO by another vehicle. A summary of the relevant ΔV requirements is provided in Table 4.

The NRHO is an elegant mathematical solution to the ΔV limitations of Orion + ESM and the SLS B1. The NRHO significantly reduces the total ΔV cost for a near-lunar orbiting spacecraft, requiring a total Orion + ESM ΔV = 0.850 km/s for insertion and exit in a 21-day mission [2]. This reduction in ΔV for access to NRHO from TLI, however, forces any lunar lander to increase their ΔV lunar ascent propellant budget by 0.85 km/s to achieve the higher orbit from lunar surface [1]. This increased lunar ascent propellant mass ripples through every possible architecture with impacts on lunar descent propellant, lander dry mass, and ultimately launch mass for the components.

The FLARE human lunar lander reference design AE upsizes the two-crew Apollo Lunar Module (LM) ascent stage from 2.4 mt dry mass [19] to 4.0 mt to accommodate four crew for seven days. For this 4.0 mt AE, the LLO orbit requires only 60% of the ascent and descent propellant compared to NRHO. The NRHO increased propellant requirement drives larger AE and DE vehicles (see Appendix B Table B11 for a comparison study of lander mass for NRHO and LLO where the SpaceTug, not the DE, performs the lunar orbit insertion maneuver). The NRHO integrated human lander (AE and DE) for this 4.0 mt AE would have a launch mass of 29.6 mt. One SLS Block 1, capable of lifting 26 mt to TLI [20], could not lift this integrated NRHO lunar lander. To reduce the lander weight on NASA's currently planned Artemis-3 mission, the HLS plans to only send two of the four astronauts from Orion to the lunar surface and back to NRHO. Two other crew will remain aboard Gateway in NRHO [21]. With FLARE, however, an integrated LLO human lander has a launch mass of 20.4 mt (including 0.5 mt of AE payload). This lander could be launched on a dedicated SLS Block 1 to TLI; however, FLARE assumes only one SLS launch/year dedicated to the Orion + ESM (limited by NASA budget and SLS production rate). The FLARE LLO human lander components (AE1 and DE2) are thus launched separately on CLVs (see Figs. 1 and 3). SpaceTugs then transfer the lander components to lunar orbit and provide the RPOD for mating. Future CLVs may lift an integrated human lander, or perhaps the Orion + ESM, directly to TLI - but the lander or ESM must then perform the required lunar orbit insertion burn and RPOD.

Table 4
Summary of ΔV requirements.

From	To	ΔV (km/s)	Xfer (days)	Reference
LEO (407 km)	TLI	3.276	<1	Mueller (2012)*
**TLI	Lunar Surface	2.900	5	Copernicus SW **
TLI	LLO (100 km)	0.952	5	Mueller (2012)*
TLI	NRHO	0.450	5	DAC2
NRHO	LLO (100 km)	0.740	<1	DAC2
LLO (100 km)	Lunar Surface	2.180	<1	Mueller (2012)*
RPOD	Any	0.045	<1	DAC2
LLO 2.5° Plane Change	Lunar Surface	0.071	0	Calculated ***
Lunar Surface	LLO (100 km)	1.968	<1	Mueller (2012)*
LLO (100 km)	TEI	1.256	<1	Mueller (2012)*
LLO (100 km)	NRHO	0.850	<1	DAC2
NRHO	TEI	0.450	<1	DAC2
TEI	Splashdown	0.011	5	Mueller (2012)*

* Includes 5% reserve.
** NASA software simulation, COPERNICUS program.
*** See Appendix B for Calculations.

4.5. Elliptical coplanar posigrade orbit (ECPO)

During the spring of 2020, HLS continued to study possible orbits that Orion + ESM can achieve using the SLS Block 1 launch vehicle. The annual NASA management presentations to the NASA Advisory Council (NAC) provide limited details on this orbit. The ECPO is an elliptical orbit that varies from 4500 km to 6500 km at apolune to a 100 km perilune, with an approximate 9 h orbital period. The orbit requires more maneuvers to enter and exit lunar orbit than a NRHO, and it greatly delays the Earth return window for a missed TEI maneuver [22]. This orbit likely requires less propellant mass for transfers to/from the lunar surface than NRHO, but extensive lander design analysis of this orbit is not available yet.

4.6. Low Lunar Frozen Polar Orbit (LLFPO)

Lessons learned from Apollo teach that vehicles in LLOs do not all have the same propellant budget to maintain their orbit. For example, the Apollo 16 mission released an orbital scientific satellite to study charged particles and magnetics fields around the Moon. The vehicle crashed into the moon after only 35 days due to unknown subsurface gravity mass concentrations ("mascons") that altered the satellite orbit with each revolution, thus causing it to deorbit much sooner than planned. Subsequent lunar missions have mapped the locations of these mascons and identified their gravity impact on lunar orbits [23]. A few special LLOs are less affected by these mascons. These "frozen orbits" have "constant mean eccentricity, mean inclination and mean argument of perigee" [24], and provide multi-year stability requiring no corrective maneuvers [2]. Frozen orbits ensure a constant altitude while minimizing the station keeping propellant budget, and are thus favored for orbiting reconnaissance spacecraft [25]. **To support a lunar south polar landing site, FLARE selects a specific Low Lunar Frozen Polar Orbit (LLFPO) with I = 86.5° and e = 0.153** [25].

With a period of approximately 2 h, this orbit provides frequent overflight of the near-polar landing site by the orbital vehicle. The entire lunar surface is available for observation and mission access at some point during a lunar month. From this inclination, a small 2.5° plane change during descent provides access to likely landing sites on flat areas near the Persistently Illuminated Regions (PIRs) of the south pole [26,27]. The ΔV cost for this 2.5° plane change maneuver is calculated, based upon orbital velocity for a 100 km lunar altitude, to be 71 m/s (see Appendix B Table B19). This is additional to the descent propellant requirement for 2.180 km/s [18]. During ascent (when the vehicle velocity is low) the 2.5° plane change is not budgeted with additional propellant to the required 1.968 km/s [18] for ascent to the 100 km altitude. A graphical comparison of NRHO and LLFPO and associated ΔV requirements is shown in Fig. 11.

4.7. Earth communications and lunar surface navigation

The average Earth visibility from possible landing sites at the lunar south pole vary from 30% to 70% during a typical month, and no likely site has 100% coverage [26]. This limitation can easily be included as a constraint in launch window development for Moon rendezvous, with the short mission of 7–14 days planned to occur when communications with Earth is in direct line-of-site. For a sustained, long-term human presence at the Lunar South Pole, the Earth visibility becomes problematic. To ensure continuous Earth communications, additional equipment needs to be placed either on lunar surface topographic features (e.g. atop a tall nearby mountain such as the rim of Malapert crater) or in lunar orbit. The surface solution could be implemented using a CLPS lander to deploy a communications tower. FLARE, however, chooses to place a satellite in orbit for continuous communications between Earth and the lunar landing site. Co-manifested with various elements of FLARE (possibly with PPE, HALO, DE1, AE, or DE2), the satellite(s) could be deployed after the payload stack achieves sufficient

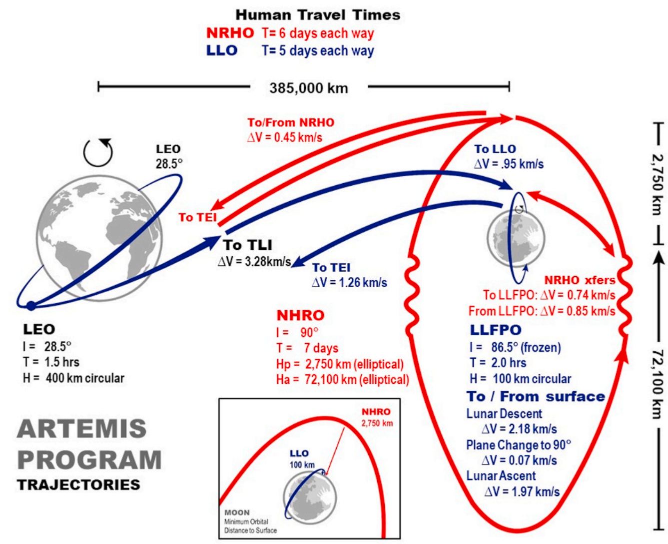

Fig. 11. Comparison of lunar orbits and change-in-velocity (ΔV) transfers.

ΔV for TLI.

5. SpaceTug

The FLARE SpaceTug is based upon a successful, mature flight-proven upper-stage developed by the United Launch Alliance (ULA). The "Common" Centaur (evolved from the Centaur-III) uses a standard RL-10 engine powered by Liquid Oxygen (LOX) and Liquid Hydrogen (LH_2) to deliver payloads to LEO atop an Atlas launch vehicle [28]. The FLARE goal is to create a SpaceTug vehicle that is capable of autonomous RPOD in Earth and Lunar orbit. It can then transfer payloads, reboost platforms, and potentially store and transfer propellants to other vehicles.

The SpaceTug requires the ULA Integrated Vehicle Fluids (IVF) technology, developed for the new ULA Advanced Cryogenic Evolved Stage (ACES) [29], to re-pressurize the system and provide power to the vehicle [30]. This also removes the need for hydrazine or helium as tank pressurizers. Additional electrical power is provided to the SpaceTug with solar arrays affixed on each side, which cover Multi-Layer-Insulation (MLI) blankets to reduce solar heating into the propellant tanks. A new Tug Adaptor (TA), replacing the Common Centaur Payload Adaptor atop the LH_2 tank, provides the electronics, batteries, and re-pressurization system components for the SpaceTug. The TA also houses the retractable docking struts and umbilical connections for mating the SpaceTug to other vehicles (including other SpaceTugs). SpaceTug configurations are demonstrated in Fig. 12.

The dry mass for the SpaceTug is 2.75 mt (adding 0.5 mt for the above modifications to the 2.25 mt "Common" Centaur dry mass) with a propellant load of 20.05 mt (slightly reduced from the Centaur-III to keep the total mass within the expected SpaceX F9 28.5° LEO capability) [28]. The SpaceTug can be stacked together to become a 2-stage vehicle for pushing heavy payloads from LEO to TLI, NRHO, or LLO (see Fig. 12). A single SpaceTug has the ability to deliver 15.5 mt from LEO to TLI, which is nearly as much as SpaceX Falcon Heavy. A double SpaceTug can deliver 32.9 mt from LEO to TLI, which is significantly more than a SLS B1 with ICPS. The double SpaceTug can deliver 21.2 mt from LEO to LLO (see Appendix B Tables B12-B17 for SpaceTug transfer calculations).

Each SpaceTug is launched on a SpaceX Falcon 9 (F9) CLV. The F9 Block 5 delivers 22.8 mt to a 28.5° LEO orbit [31] for an unspecified altitude. Prior SpaceX documentation reveals approximately a 5% reduction in payload delivery with a F9 Block 2 between a circular 200 km orbit (delivery = 10.454 mt) and a circular 400 km (delivery =

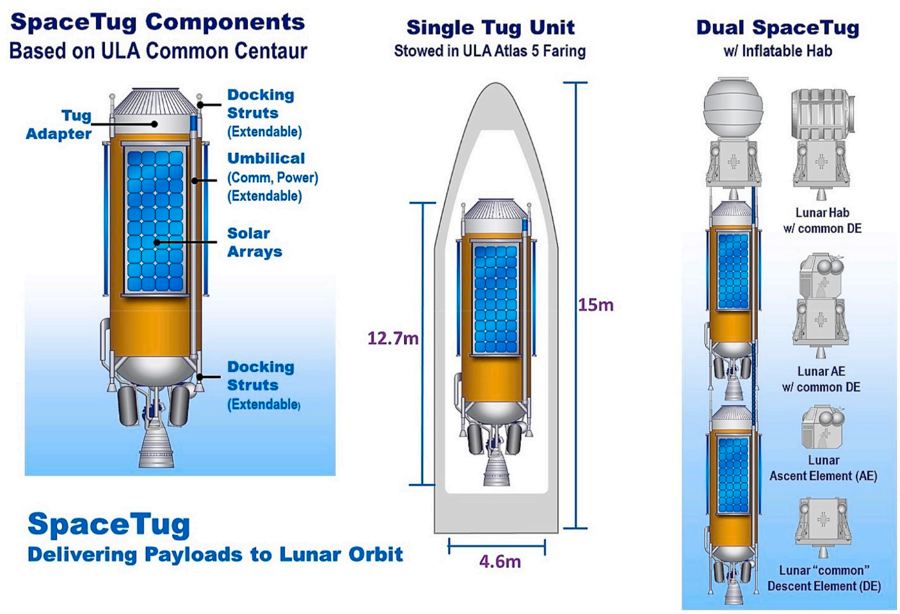

Fig. 12. SpaceTug Configurations.

9.953 mt) orbit [32]. The F9 Block 5 is thus expected to incur the same Performance Fraction (PF) for similar LEO altitudes. It is unknown how much reserve SpaceX maintains for the F9 Block 5 capacity. FLARE thus assumes the F9 Block 5 can deliver 22.8 mt to a 400 km circular orbit of 28.5° inclination. The 2019 Falcon 9 fairing is 5.2 m in outer diameter and 13.2 m high overall and it can accommodate payloads of 4.6 m diameter and 11 m tall (barrel volume). A longer fairing is needed for the SpaceX Falcon 9 (F9) to hold the 12.7 m tall SpaceTug derived from the ULA Common Centaur. There are longer fairings available, developed by RUAG for the ULA Atlas V, that have previously been discussed to support Department of Defense (DoD) payloads [33]. The SpaceX launch pad also needs modification to allow cryogenic refill of the SpaceTug immediately prior to launch.

The boil-off of cryogenic LOX/LH_2 must be minimized in vehicles on-orbit. This cryogenic propellant is chosen for both the SpaceTug and the human lander reference design. The original Titan/Centaur was designed to support an 8-h mission with a boil-off of 2%/day [34]. ULA has developed and patented numerous concepts to store propellant on-orbit [35,36]. A new design, using the Centaur upper-stage as a secondary tank, called the CRYogenic Orbital Test (CRYOTE) concept, conceives of up to 1 year of storage of cryogenic propellants on-orbit [37,38]. Building on CRYOTE tests, the next ULA concept uses a "Drop Tank" which waits in LEO for "days, weeks, or even months" to refill a Centaur upper-stage launched on a subsequent mission [34]. The Drop Tank remains attached to its depleted upper-stage and spins slowly (1°/sec) to provide centrifugal acceleration that settles the cryogenic fluids. The Drop Tank design includes features to minimize boil-off with insulated blankets, lightweight materials, a vacuum insulated common bulkhead and low conductivity struts. This expected boil-off is under 0.1%/day of the total propellant load. Studies with "Zero-Boil-Off" systems show that spacecraft with proper insulation, and/or small cryo-cooling systems, could provide for months of liquid hydrogen storage without evaporation [39,40]. **FLARE assumes the LOX/LH_2 boil-off rate of 0.5%/day (based on a full cryogenic propellant tank) for both the SpaceTug and the reference Descent Element (DE) of the human lander.**

Although FLARE does not require on-orbit fluid transfer of LOX/LH_2, it provides an opportunity to demonstrate this capability in LLFPO (see Sequence #18 in Table 2). NASA has investigated cryogenic fuel transfer and fuel depot concepts using the Centaur, with a goal of reducing boil-off to 0.1%/day [41]. A SpaceTug in LLFPO could transfer residual propellant to the human lander DE or another SpaceTug. NASA considered the transfer of LOX between vehicles in Earth Orbit Rendezvous (EOR) as one of four architectures for lunar exploration. The Apollo plan required transfer of oxidizer from a tanker S-IVB upper-stage to a Trans-Lunar Injection Stage containing the CM, SM, Lunar Touchdown Module and Lunar Braking Module in order to achieve TLI [42]. A challenge of cryogenic fluid transfer on-orbit is evolved gas release during the fill process. "No-vent-fill" designs, tested by NASA in the 1990s [43,44], use cold liquid thermodynamic properties to condense the vapor in the tank [34]. In 2019, a company built an experiment "Furphy" on the ISS that successfully demonstrated the transfer of water on-orbit [45]. NASA and Yetispace have conducted tests on Earth demonstrating successful liquid nitrogen transfer under flight-like conditions [46]. The Shuttle program demonstrated an astronaut-controlled remote transfer of hydrazine between two tanks mounted in the payload bay of STS-41G [47]. Originally planned as a water transfer, Astronaut Dave Leestma convinced NASA management to allow the transfer of toxic hydrazine in order to better simulate refueling of spacecraft on-orbit [48]. Similarly, the Apollo 14 crew demonstrated liquid transfer of an inert fluorochemical, perfluorotributylamine, from one container to another using a hand pump operated by an astronaut [49]. The propellant transfer technology gap must be closed to support long-term harvesting of planetary resources to fuel space vehicles. SpaceTug on-orbit transfer of LOX/LH_2 enables future Mars exploration and lunar commercialization.

The FLARE minimum required crew phases requires five SpaceTugs, of which three (Tug1AE1, Tug1RST, Tug2DE2) ultimately crash on the lunar surface. Adding an auxiliary Electric Propulsion (EP) system to the SpaceTugs would allow the depleted LOX/LH_2 vehicles to remain on-orbit and be repurposed as communications, navigation, or science experiment satellites. NASA has employed EP on three science missions: Deep Space 1, Dawn, and Space Technology 7 [50]. Current NASA Solar Electric Propulsion (SEP) plans employ a 100 kw system with a 13.3 kw Hall thruster system for the PPE [51]. Development of a low power SEP, perhaps using the SpaceTug solar arrays (anticipated power in the 100 s W range), has previously been studied [52].

The SpaceTug could provide both the storage system and the transportation engine to deliver In-Situ Resource Utilization (ISRU) harvested LOX/LH_2 propellants from the lunar surface to LLO (or beyond). Numerous lunar fuel depot studies have been published detailing the necessary technology and infrastructure requirements for this capability

[53–56]. ULA has developed a concept to transform an ACES upper stage into a horizontal lander. Called XEUS, it provides a novel design to deliver crew and cargo to the lunar surface [30]. For FLARE, a vertical "lander" SpaceTug is estimated to require an additional 1.25 mt (dry mass = 4.0 mt) of hardware modifications for structural components (legs, pads, tanks) and surface refilling equipment. FLARE calculations predict that a modified SpaceTug could achieve LLFPO with ~11.25 mt of propellant remaining from a full (~20 mt) propellant load on the lunar surface (see Appendix B Table B18). Orbiting SpaceTugs could then refill other orbiting vehicles.

An additional future concept for future SpaceTug design is aerocapture. Rather than have a propellant depleted vehicle de-orbit to Earth after pushing components from LEO towards TLI, necessary decelleration could be performed with an inflatable shield that would slow the SpaceTug again to LEO velocities for RPOD with ISS [55]. The SpaceTug could then be refilled and reused from LEO.

6. Lunar landers

FLARE provides a reference 2-stage human lander design concept, consistent in mass and volume with numerous previous lunar lander designs [57]. The components are a "Common" Descent Element (DE) for either cargo or crew delivery to the lunar surface from LLFPO, and a pressurized Ascent Element (AE) for crew transfer to and from the lunar surface. The AE also provides a lunar surface residence for four crew up to seven days. The FLARE concept can accommodate any commercial lander for humans or cargo that falls within the mass, diameter, and height constraints of available CLVs launching elements to LEO. The sequence would follow a similar configuration for the FLARE reference human lander components, such as AE1 (See Fig. 1) using one SpaceTug for transfer from LEO to LLFPO, or DE2 (See Fig. 3) using two SpaceTugs for the transfer to LLFPO.

6.1. Commercial Lunar Payload Services (CLPS)

In April 2020, NASA announced three commercial contracts for development of a human lunar lander supporting the Artemis Program [58], although these landers are not required to follow HLS's DAC2 requirement for rendezvous with Gateway in a NRHO [15]. NASA had previously selected nine companies in 2018 to provide unmanned landers for lunar exploration with the Commercial Lunar Payload Services (CLPS) program [59]. An additional 12 NASA payloads and experiments were selected in early 2019 [60], with another 12 selected in summer 2019 [61]. Beginning in 2021 with payloads of at least 10 kg, the CLPS landers are expected to grow to support future payloads of up to 500 kg or larger [59]. The CLPS landers can then provide preliminary science or human infrastructure equipment for human surface missions.

6.2. Human Ascent Element (AE)

Selecting the AE mass is the key design driver for FLARE. The AE supports the crew as they descend from LLFPO, land, live on the lunar surface for 7 days, then return to Orion (or Gateway) in lunar orbit. The AE has a dry mass of 4.0 mt (upscaled from the 2-man Apollo vehicle), and thus requires ascent propellant of 4.4 mt to achieve LLFPO (see Appendix B Table B9). The AE volume is 8.4 m^3 (or approximately 2 m^3 per person) and has a pentagon-shaped outer mold line (see Fig. 13). The AE is designed such that the crewmembers stand during descent and ascent. The AE supports crew use of either full xEMU spacesuits or Orion Launch and Entry Suits (LES). The crew has sufficient volume to don/doff their xEMU and/or LES two-at-a-time, and also supports crew sleep periods (by use of hammocks) inside the pressurized volume. The AE is designed to carry 100 kg of lunar surface and crew biological samples (allocated as payload). The AE dry mass includes 20 kg of additional science supporting equipment such as containment boxes within the pressurized volume.

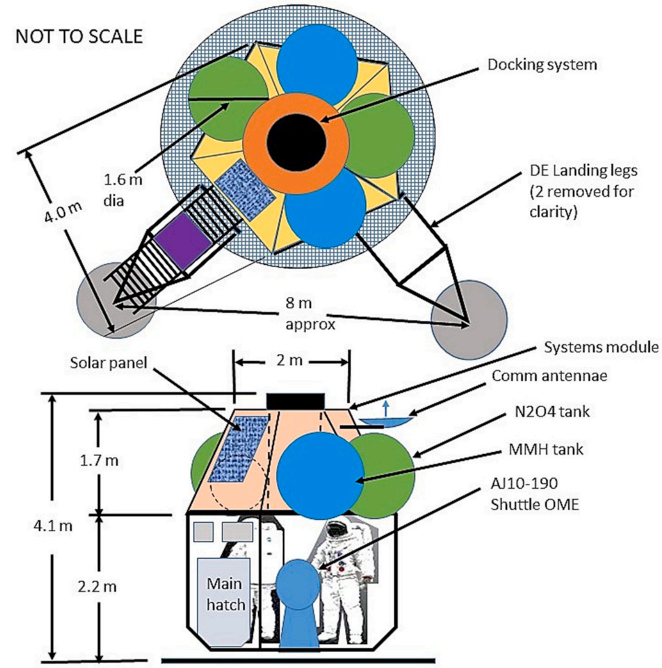

Fig. 13. Ae conceptual design.

The AE propulsion system uses a single Shuttle AJ10-190 Orbital Maneuvering Engine (OME) internal to the crew volume, similar to the Apollo-era Lunar Excursion Module (LEM). The AE uses the hypergolic bipropellants of Dinitrogen Tetroxide (N_2O_4) and Mono-Methyl Hydrazine (MMH) fed under helium pressurization and ignited in the OME with an oxidizer-to-fuel ratio of 1.65:1. It has two N_2O_4 and two MMH Composite Overwrap Pressure Vessels (COPV) tanks, both with 5% ullage volume and the same 1.6 m diameter. If loaded to full capacity, the four tanks contain approximately 1734 kg of MMH and 2667 kg N_2O_4 for a total ΔV capability of 2.3 km/s for ascent to the 86.5° inclination 100 km LLFPO. The AE also has four Reaction Control System (RCS) thruster quads fed from the same propellant and pressurant tanks as the OME.

The main pressure wall material of the AE habitable volume uses the common 0.040" aluminum 6061-T6 material (slightly thicker than the Apollo LEM 0.012" thick pressure wall). Similar to the DE, it has external MLI interleaved with additional sheets of inexpensive fiberglass fabric for micro-meteoroid protection [62], and polyethylene sheets for radiation protection [63] The entire AE module itself is structurally attached to the DE-based support structure using (4) pyrotechnic bolts with ZipNuts. These pyrotechnically-modified Snap-On© ZipNuts [64] mate the 2 elements (AE and DE) on-orbit and only require a straight push to fully engage. They therefore do not require any turning of the nut to ensure proper torqueing. These also have the advantage that the greater the tension the better the split nut threads grip the bolt shaft. The bolts themselves have been modified to fragment the bolt head upon the separation firing command. The AE will fire the four pyrotechnic bolts to separate from the DE immediately prior to liftoff from the lunar surface.

The FLARE AE is designed for an atmospheric pressure of 8.2 psi to 14.7 psi. The AE is able to support multiple depress/repress cycles for EVA, including the optional Phase B extended surface mission. With the optional surface habitation module in Phase B, the crew depress the AE to vacuum, power down the AE, traverse to the inflatable habitation module, complete the surface mission, return to the AE, repress the AE, power up the AE, liftoff and ascend to the Orion + ESM. The AE has a single large Ingress/egress hatch for individual crew access in and out of the module while on the lunar surface. This allows a single crewmember to quickly traverse across the sill while only stooping slightly. A second hatch exists for crew entrance/exit on-orbit when docked to either the

Orion or Gateway. The AE has two windows, each of which is a triple pane system to protect the pressure pane from inadvertent impact from the crew or impact from micro-meteoroids in lunar orbit or on the surface. After the AE1 vehicle returns to Orion in LLFPO, it could be repurposed as an additional pressurized volume or as an airlock for Gateway (if present).

6.3. "Common" Descent Element (DE)

The FLARE "common" DE concept is shown in Fig. 14, representing the vehicle as a lunar descent stage for either a cargo platform (4.5 mt payload) or for the crewed AE (8.4 mt vehicle).

The DE uses two flight control computers for command and control when an AE is not attached. When an AE is attached, the AE flight computers control the integrated stack. By implementing the optional Phase B, the "common" DE is fully flight tested and demonstrated prior to its required service for landing the crew. The DE lightweight composite truss structure has a dry mass of 2.5 mt, which requires a total mass of 11.5 mt consisting of:

a) two LOX and two LH_2 tanks carrying a total of 9.0 mt propellant (7.714 mt LOX, 1.286 mt LH_2)
b) four helium pressurant tanks at pressure of 2.0684E7 Pascals (3000psi)
c) four small self-contained monopropellant RCS thruster pods each containing 40 kg hydrazine
d) a single gimbaled RL-10A-4-2 engine (specific impulse of 450.5 s)
e) four landing legs that are launched folded up (to fit inside the FH CLV fairing), then deploy and lock into place after the TLI burn is completed
f) an overhead composite support structure for a maximum 8.4 mt payload mass (crewed AE)
g) a base composite support structure for the Main Propulsion System (MPS) tanks, leg attachment and main engine
h) two composite thrust deflection ramps to redirect the AE module ascent plume away from the DE propellant and pressurant tanks during AE ascent.

The required ΔVs (shown in Table 4) for lunar descent and landing from LLFPO are 2.180 km/s ΔV for altitude change [18], and 0.071 km/s for the plane change (see Appendix B Table B19). Note that FLARE Sequence #19 (see Table 2) assumes the Descent Element (DE2) for crew delivery to the lunar surface provides this entire ΔV (see Appendix B Table B8). An option exists using a SpaceTug for part of this ΔV. Initially, Tug2DE2 pushes the DE2 from TLI to LLFPO and has residual propellant available when the crew arrives to LLFPO. Although not included in these calculations, the Tug2DE2 could conduct a perigee adjust maneuver for the crewed lander (AE1+DE2) to decrease the deorbit propellant needed by DE2 for the lunar landing.

The DE MPS consists of two cylindrical, dome-capped 2.75 m long LH_2 fuel tanks (1.286 mt at a density of 70.8 kg/m^3) and two cylindrical, dome capped 1.9 m long LOX oxidizer tanks (7.714 mt at a density of 1141 kg/m^3). This provides a total of 9.0 mt of propellant. Ullage volume was assumed to be 10% for both the LOX and LH_2 tanks. The DE employs a single gimbaled RL-10A-4-2 deep throttle-able engine mounted on the base thrust structure burning an oxidizer-to-fuel ratio of 6:1. As stated previously, FLARE assumes a cryogenic LOX/LH_2 boil-off rate of 0.5%/day (for both SpaceTugs and DE).

The DE has four landing legs which are launched stowed to fit within the 4.6 m dynamic envelop of the FH fairing. The legs are deployed immediately following the completion of the TLI burn. Each leg is deployed using small electrical motors and the legs lock in place when fully extended. Internal to each leg is a crushable aluminum honeycomb structure which absorbs a portion of the touchdown loads. One leg has the crew surface access ladder attached to it. The ladder is in segments to "fold" when the landing leg compresses from the touchdown loads. This prevents the buckled ladder from blocking the crew access to the surface. Approximately 2 m above the landing pad is a small open-grated "porch" which also functions as a station for dust removal from the xEMU, as well as a storage location for tools and other equipment.

DE attitude control is maintained by a combination of reaction wheels, RCS thrusters and main engine gimbaling. The DE uses a reaction wheel module mounted on the support structure that torques the DE when slower rate movements are required. When faster rates are required, the four 40 kg RCS thruster pods are also utilized. These are mounted on the four landing legs of the DE and are self-contained monopropellant (hydrazine) systems. The "up" firing thruster on the RCS thruster pod on the leg with the ladder is normally inhibited from firing, but can be commanded to fire through the open-grated porch platform. The opposite leg, opposite thruster is normally used to accomplish this "pitch down" motion. When faster rates are required, a combination of reaction wheel operation, RCS thruster firings, and main engine gimbaling are used to accomplish the movement.

The AE support structure on the DE is a composite frame attached to the base thrust structure and is intended to support the loads induced from a fully loaded AE during lunar orbit maneuvering and landing on the lunar surface. It is not flown during any precursor supply mission. The top of the AE structural support structure on the DE is located 5.0 m

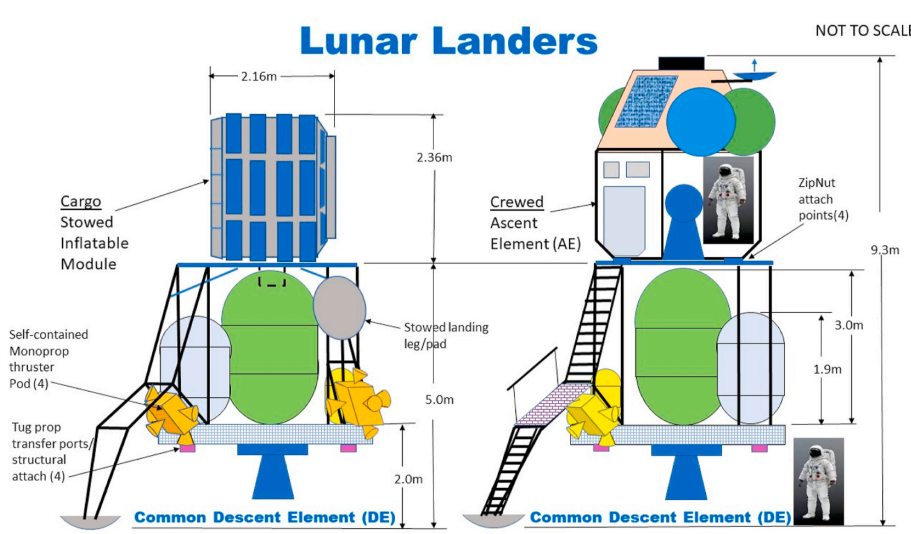

Fig. 14. "Common" DE conceptual design.

above the lunar surface and consists of an open web platform with the four attach points for the AE. Also attached to this structure are two ascent engine exhaust "chutes". These are placed to redirect the AE OME liftoff exhaust plume such that it doesn't impinge on the DE LOX/LH_2 tanks and potentially cause an explosion. These main LH_2/LOX propulsion system components are mounted inside this support structure and, with the chutes in place, are able to withstand the brief pressure and temperature spikes seen during AE liftoff.

The DE does not have a "hard" shell around its moldline. Surrounding DE propulsion components are "soft side" sheets of MLI interleaved with additional sheets of inexpensive fiberglass fabric. This provides a viable short-term Micro-Meteoroid and Orbital Debris (MMOD) shield similar to what is used to protect the SpaceX Dragon commercial cargo vehicles [62].

A FLARE stretch goal is to have the DE capable of on-orbit refueling from a SpaceTug (see Sequence Step #18 in Table 1). Mounted on the underside of the base thrust structure are four propellant transfer/structural attach points for the SpaceTug extendable docking struts. These ports provide pressurized fluid transfer as well as structural attach points for the SpaceTug. In addition, there are also two ports on the underside for power/data umbilical attachments. These redundant ports provide communication and power transfer to/from other SpaceTugs when they are attached.

7. Lunar surface precursor equipment

The FLARE "common" DE design provides a delivery vehicle for crew or cargo to the lunar surface with mission DE1 (see Table 3, Sequence #10 and Fig. 8). With the additional launch of one SpaceX Falcon Heavy to TLI, the DE1 can land 4.5 mt of precursor cargo payload mass directly to the lunar surface. This option could be executed once, or multiple times, to build a sustainable lunar surface infrastructure. The first precursor payload (landed with DE1) includes an inflatable habitation module (2.0 mt), consumables supporting four crew on a 14-day surface mission (0.35 mt), two additional xEMU space suits (0.38 mt), individual mobility vehicle(s) (0.92 mt), science experiments on a portable trailer (0.30 mt), and an In-situ Resource Utilization (ISRU) demonstration (0.40 mt) selected from NASA contractors investigating space resource collection [65]. Addition smallsats (0.15 mt) carried onboard the DE1 are deployed prior to descent to support lunar surface communications and navigation.

7.1. Inflatable habitation module

The primary component of the precursor mission is an inflatable habitation module. Based upon the proven design of the ISS Bigelow Expandable Activity Module (BEAM), the 3.6 m^3 packed volume expands to 16 m^3 when inflated [66], which exceeds the reference AE volume of 8.4 m^3. The ISS BEAM dry mass of 1.4 mt is increased to 2.0 mt for FLARE to include surface components for crew sleep stations, toilet, galley, and an inflatable airlock. The assumed internal pressure is 14.7 psi, although detailed design may lower the desired pressure to 8.2 psi to accommodate EVA preparations. Since the inflatable hab and airlock are delivered directly to the lunar surface, they do not require any orbital docking with other vehicles. Thus, the hatch diameter can be increased from the Orion hatch size.

An additional mass of 0.35 mt is also included in the DE1 payload for crew logistics to support a 14-day surface mission for 4 crew (water/air/food), which is 6.25 kg/person/day. Similar to the AE, the inflatable habitation module has external MLI interleaved with additional sheets of inexpensive fiberglass fabric for micro-meteoroid protection, and polyethylene sheets for radiation protection [63]. Vertical solar arrays atop the inflated habitat (located at least 8 m above the surface) provide nearly continuous power generation for the field station, despite the low polar sun angle [26].

7.2. Lunar All-Terrain Vehicle (LATV)

The FLARE surface mobility concept is modeled after the successful Antarctic Search for Meteorites (ANSMET) program. The ANSMET program, sponsored by the National Science Foundation (NSF), has recovered more than 22,000 meteorites from the ice of Antarctica since 1976 [67]. A small team of scientists is deployed each year to a scientific "field station" in the remote Antarctic mountains for 4–8 weeks of research and sample recovery. The ANSMET team conducts exploration traverses by walking and using individual mobility devices, the Ski-Doo snowmobile. The FLARE optional Phase B precursor mission has limited mass and volume that likely precludes an unpressurized 2-person rover, but could accommodate smaller 1-person vehicle(s) that fits (partially disassembled) within the deflated habitation module. NASA is currently seeking commercial designs for an unpressurized 4-wheel, 2-person "buggy"-style Lunar Terrain Vehicle (LTV) for Artemis missions [68]. Following are considerations for selecting a one-person Lunar All-Terrain Vehicle (LATV), rather than LTV for early mission crew mobility on the lunar surface.

7.2.1. Modify existing designs

Modification of an existing commercial design is likely much faster and cheaper than developing a new vehicle, so the LATV might be available before a LRV. Many vendors provide "hardened" All-Terrain Vehicle (ATV) designs for off-road, sand, snow, or ice conditions with electric propulsion systems using removable batteries (which allows for rapid replacement of depleted batteries). The LATV must survive the Moon environment, including extremely cold conditions and dusty regolith. The LATV must provide batteries that can be easily recharged. The LATV must be compatible with the lunar xEMU suit design so that astronauts can sit and ride on the vehicle comfortably. Commercial companies may contribute "in-kind" development resources for a LATV to gain publicity and market attraction for collaboration with NASA. A reference design by Doohan [69] provides an example of a commercially-available, 3-wheeled, electric vehicle with an Earth weight of ~160 kg.

7.2.2. Light weight vehicles

A typical ATV weighs from 100 to 300 kg on Earth, which is 50 kg (approximately 110 pounds) on the Moon with 1/6th Earth's gravity. A single astronaut can partially or completely lift, rotate, or push this mass on the Moon. If the LATV is stuck in soft regolith, the crew could lift the wheel or pull the vehicle towards a more solid surface.

7.2.3. UnPacking, stowage, and spare parts

The LATV can be designed to easily assemble/disassemble into large pieces, so that they could perhaps be initially unloaded from the hatch in the deflated surface habitation module (FLARE optional Phase B with DE1) or from a CLPS nearby lander. The crew then rapidly assembles the components into a functional vehicle on the lunar surface. Reversing the process allows long term storage of the vehicles in the habitat, or in a lunar surface shed that protect them from solar wind and micro-meteoroid exposure while providing warmth and power to survive lunar nights. ANSMET teams in Antarctica have often cannibalized parts from one snowmobile to keep other vehicles functional [70]. Having multiple identical vehicles on the lunar surface similarly provides an inventory of spare parts in the event of unforeseen failures.

7.2.4. Rechargeable batteries

The mobility system needs solar array(s) with recharging capability for the LATV batteries. As previously discussed, solar arrays atop the inflatable habitat could be used for recharging the LATV batteries, or a separate ground station could be installed. Multiple excess batteries should also be charged and ready for rapid replacement once an LATV has depleted the energy in its current battery. The recharging of each battery could either be done inductively with the battery remaining on the vehicle (necessary for telerobotic operations), or the battery can be removed by the crew and attached to the recharging terminal (either outside or inside a pressurized volume).

7.2.5. Telerobotic capability

Telerobotics on a LATV adds multiple capabilities to the entire mission. NASA and terrestrial industries have developed the capability to robotically drive a vehicle using a remote operator. For remote lunar operations, the vehicle must be designed to recharge the electrical battery directly with easy connectors to a charging station or through inductive field charging.

Telerobotics allows exploration of surface regions that are precluded from humans (due to slopes, temperatures, or surface roughness/softness). The astronaut could observe the LATV directly and remotely drive the vehicle in conducting operations in these more difficult terrains. The LATV could also be telerobotically operated as a rescue vehicle to recover a stranded astronaut at a distant location.

7.2.6. Traverse flexibility and crew rescue

The LATV can support lunar surface traverses up to 10 km from the landing site. Using a pair of LATVs provides redundancy and rescue capability. The pair of astronauts could work collaboratively or independently on tasks. This increases the efficiency of astronaut EVA time and enhances the flexibility in executing traverses. Each LATV is capable of carrying two xEMU-suited astronauts. A rescue LATV could be driven to a stranded astronaut by one crew, or an empty LATV could be telerobotically driven to the disabled crew location.

7.2.7. Science trailer

Similar to the SpaceTug concept for orbital transfers, the LATV provides surface transfers for science and crew. The LATV provides a simple, flexible vehicle derived from existing commercial designs at minimum cost. To minimize development cost and schedule risk, FLARE separates the surface science function on a dedicated trailer. Fixed and deployable instruments, including a robotic arm, are mounted on a chassis that provides power, communications, and computer resources. The trailer is pulled to a desired location by the LATV (driven by crew or telerobotically), left to collect data (but commanded remotely), and then later retrieved by the LATV. One science trailer (allocated mass 0.3 mt) is included in the optional precursor FLARE Phase B mission, but it could be delivered separately on a CLPS lander.

8. Lunar surface crewed concept of operations

After a successful launch and transit to the Moon, the Orion + ESM docks with assets in LLFPO, which is, at a minimum, the human lander (AE1 + DE2) and the RST. The PPE and HALO may also be present (as Gateway) but are not required. On Crew Flight Day 8 (see Fig. 10) all four Orion crew board the AE1 and seal the hatch. They then descend to the lunar surface and land at the designated site. After safely landing on the lunar surface, the operations concept depends upon the available surface assets. If Phase B is NOT implemented, the four crew will conduct limited walking EVAs alternating pair teams using the 2 xEMU suits in the AE. The surface duration may be very brief (hours) or extend up to one week before the crew returns to Orion.

The FLARE Option B extends to a 14-day surface mission with longer science traverses using the prepositioned assets. All four Orion crew descend to the lunar surface inside the pressurized AE1. Two of the crew are dressed in xEMU suits, and two are dressed in the Orion LES. Once the DE2 has landed safely on the Moon, the AE1 is depressurized and the two crew dressed in xEMU suits depart. The two crew remaining in the AE1 - dressed in the LES that can keep the astronauts alive for up to 6 days [71] - then repressurize the AE1 and wait for the return of their crewmates. The two crew in xEMU suits walk to the deflated habitation module and unload the crew mobility vehicles and two xEMU suits from inside. They then inflate the habitation module and attached inflatable airlock, then return to the AE1. The AE1 is again depressurized to allow the xEMU crew to enter, then the AE1 is pressurized and the two crew left inside doff the LES and don the xEMU. The AE1 is again depressurized and all 4 crew depart to the DE1 landing site. The crew ingress/egress the habitation module via the inflatable airlock. Science, ISRU, EVA tools, and solar array mobility equipment are unloaded and deployed. The crew sleeps in the habitat for the next 14 nights. Each day, one team conducts traverses using the LATVs, and one team remains at the habitat landing site (either inside the habitat or conducting walking EVAs). Each traverse pair includes one crewmember on a LATV, and the other one on an LATV pulling the science trailer. A third LATV (if available) can be pulled behind the non-science trailer pulling the LATV, driven telerobotically, or left behind at the habitat site (for rescue). Each team of two crew will alternate days as an LATV traverse team or local habitat team. Upon completion of the 2-week campaign, the crew will reconfigure the site for remote science operations by enabling telerobotic equipment and stowing equipment and configuring the habitat module for human absence. The crew then reverse the sequence of walking between the AE1 site and habitat site returning two xEMU suits to the habitat and bringing two xEMU suits into the AE. On Flight Day 22 (see Fig. 10), all four crew in the AE1 return from the lunar surface to Orion in LLFPO.

9. Transportation system capabilities and costs

9.1. Comparison of launch vehicle performance

A comparison of past, present, and future vehicle payload mass delivery to LEO and TLI is provided in Table 5 [20,31,32,72–77]. Mass fractions are used to equalize delivery mass to LEO and TLI across vendors. The 0.94 mass fraction for performance reduction from LEO 28.5° to LEO 51.6° is based upon published values for the SpaceX F9 Block 2 [32]. The 0.27 mass fraction for performance reduction from LEO 28.5° to TLI is based upon published values for SLS [75,78]. Note the Space Shuttle (retired) is an entirely different architecture from CLVs that does not follow the mass fraction for 51.6°, and it was not capable of delivering payloads to TLI. Also included in Table 5 is the predicted performance of a ULA SpaceTug (FLARE concept) in either a single (See Fig. 1) or double-stacked (See Fig. 3) configuration.

The FLARE LEO assumption is a 400 km circular orbit at 28.5° inclination. The chosen CLV for human lander (AE1 and DE2), or

Table 5
Comparison of launch vehicle performance.

	From KSC (ETR)	Launch to 28.5° LEO	Launch to 51.6° LEO	Launch to TLI
	Mass Fraction*	1.00	0.94*	0.27**
Vendor	Launcher	(mt)	(mt)	(mt)
SpaceX	Falcon 9 Block2	10.0	9.4	2.7
ULA	Atlas V 551	18.5	17.7	5.1
SpaceX	Falcon 9 Block5	22.8	21.4	6.2
ESA	Ariane 5	20.2	19.0	5.5
ULA	Delta IV Heavy	28.8	27.1	7.9
ULA	Vulcan Heavy	32.62	30.7	8.9
Blue Origin	New Glenn	45.0	42.3	12.3
ULA	SpaceTug (1 only)	n/a	n/a	15.5
SpaceX	Falcon Heavy	63.8	60.0	17.5
NASA	SLS Block I	95	89.3	26.1
ULA	SpaceTug (2 stacked)	n/a	n/a	32.9
NASA	SLS Block 1b	105	98.7	37.0
NASA	SLS Block II	130	122.2	45.0
NASA	Space Shuttle	27.5	16.1	n/a

* Mass fraction for 51.6° from SpaceX F9 Block 2.
** Mass fraction for TLI from NASA SLS Block 1.

optional Gateway (PPE and HALO), components is the ULA Atlas V (A5) 551 rocket, which is capable of lifting 18.5 mt to LEO [79]. The FLARE reference CLV for the SpaceTug (including the RST) delivery to LEO is the SpaceX Falcon 9 (F9) Block 5 rocket, which is capable of lifting 22.8 mt to an unspecified altitude [31]. Prior SpaceX documentation reveals approximately a 5% reduction in payload delivery with a F9 Block 2 between a circular 200 km orbit (delivery = 10.454 mt) and a circular 400 km (delivery = 9.953 mt) orbit [32]. The F9 Block 5 is thus expected to incur the same Performance Fraction (PF) for similar LEO altitudes. It is unknown how much reserve SpaceX maintains for the F9 Block 5 capacity.

9.2. Transportation system costs for FLARE

The Federal Aviation Administration (FAA) publishes an annual report on the estimated launch costs for commercial vendors [72], and frequent website announcements provide general industry details on price modifications. FLARE assumes a launch cost of each SpaceX F9 is \$62 M, each SpaceX FH is \$90 M, each ULA A5 is \$153 M, and each SLS B1 is \$2000 M [80]. In general the SLS cost overwhelms the cost of all other components. Using only one SLS launch for the crewed Orion + ESM minimizes this expense, but forces the human lander elements (AE1 and DE2) to be launched separately on CLVs to LEO and then pushed to LLFPO by SpaceTugs for autonomous docking.

The unit cost (after development) of a SpaceTug is expected to be similar to the existing ULA Common Centaur. The SpaceTug assumed build cost is \$80 M each (~1/2 the cost of an Atlas V 551 launch). The development costs of the human lander components is not included. The optional precursor mission (Phase B) costs are also not estimated, but FLARE has chosen existing components which could be modified for less cost than new development.

In summary, the transportation costs for the minimum required phases (C,D,E) is estimated at \$3,016 M (5 SpaceTugs, 5 F9, 2 A5, and 1 SLS B1). The optional Phase B transportation costs (CLV) add \$90 M for one SpaceX FH launch. The assembly of Gateway PPE and HALO components in LLFPO add \$590 M (2 SpaceTugs, 2 A5, and 2 F9) (see Appendix B Tables B20-B22 for calculations of costs).

10. Conclusion

The FLARE provides a reasonable, practical sequence to deliver four Americans to the Moon and then return them safely to Earth. The FLARE supports the Artemis Program using components currently being developed by NASA. The underlying FLARE concept is to maximize available commercial technology for the mission, and limit development of entirely new systems or vehicles. As new technology or vehicles become available, FLARE provides multiple growth opportunities for their integration. Transportation costs are minimized using existing CLVs for delivery of components to LEO, and the SpaceTug (which is a modification of mature, successful technology combined with newer, proven innovations) provides the necessary propulsion for transfer between LEO and LLFPO (which is the optimal lunar orbit for sustained lunar surface operations). A new human lander is required, but commercial contracts provided by NASA are underway for its further definition and development. FLARE provides a human lander reference design that can be replaced with any of the selected commercial landers. FLARE allows for inclusion of Gateway elements (PPE and HALO) and lunar surface precursor equipment to extend and enhance human surface operations. Crew lunar surface rovers being investigated by NASA can be added, although FLARE provides a reference concept for an individual vehicle, called the Lunar-ATV (LATV), for early human surface missions. Advanced technology demonstrations for on-orbit fluid transfer, ISRU propellant resupply, and deep space communications satellites are included in FLARE to enable enhanced exploration of cislunar space, then on to Mars.

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The authors declare that they have no known competing financial interests or personal relationships that could have appeared to influence the work reported in this paper. Both authors, as employees of NASA, created this research while supporting the Artemis Program. The authors are not participating in any selection board for contracts regarding commercial lunar landers or surface mobility rovers. This research did not receive any specific grant from funding agencies in the public, commercial, or not-for-profit sectors.

Declaration of competing interests

The authors declare that they have no known competing financial interests or personal relationships that could have appeared to influence the work reported in this paper.

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APPENDIX A. ACRONYMS

A5	Atlas V rocket built by ULA
ACES	Advanced Cryogenic Evolved Stage, the ULA new Upper Stage for Vulcan rocket
AE	Ascent Element (HLS terminology)
ANSMET	Antarctic Search for Meteorites (NSF funded program in USA)
ARES	Astromaterials Research and Exploration Science Division at NASA JSC
ATV	All-Terrain Vehicle (FLARE terminology) or Automated Transfer Vehicle (basis of ESM)
CC	Crew Capsule (HLS terminology)
CFM	Cryogenic Fluid Management (HLS terminology)
CIII	Centaur III upper-stage built by ULA for Atlas V rocket
CLV	Commercial Launch Vehicle delivering payload to LEO or TLI (HLS terminology)
CM	Apollo Command Module
COPV	Composite Overwrap Pressure Vessels, component of the FLARE AE
CV	Centaur V upper-stage built by ULA for Vulcan rocket
DAC	Design Assessment Cycle, NASA evaluation of architectures for Artemis program

(*continued on next page*)

(*continued*)

A5	Atlas V rocket built by ULA
DAE	Fused (single-stage) Descent-Ascent Element, Combined (HLS terminology)
DE	Descent Element (HLS terminology)
DoD	Department of Defense
ECPO	Elliptical Coplanar Posigrade Orbit, a possible orbit being studied by HLS for Artemis
EML1	Earth-Moon L1 point
EML2	Earth-Moon L2 point
EOR	Earth Orbit Rendezvous, a concept considered for Apollo stages before TLI
ESA	European Space Agency
ESM	European Service Module (based on Automated Transfer Vehicle, ATV)
F9	Falcon 9 built by SpaceX
FH	Falcon Heavy built by SpaceX
FLARE	Flexible Lunar Architecture for Exploration, an alternate architecture scenario for NASA's Artemis Program
HALO	Habitation and Logistics Outpost (former "Mini-Hab" Gateway component before 2019)
HEO	High Earth Orbit
HLS	Human Landing System (HLS) - NASA team developing the Artemis human lander
ICPS	Interim Cryogenic Propulsion Stage (ICPS), upper-stage for the NASA SLS Block 1
ISS	International Space Station, NASA vehicle located in Earth's LLO at 51.6° inclination
LLFPO	Low Lunar Frozen Polar Orbit, a specific LLO at 86.5° inclination
LBM	Lunar Braking Module, a concept not adopted by Apollo using EOR
LEM	Lunar Excursion Module (LEM), later Lunar Module (LM) from the NASA Apollo program
LEO	Low Earth Orbit, for FLARE defined as a 28.5° inclination, 400 km circular Earth orbit
LES	Orion Launch and Entry Suit
LLO	Low Lunar Orbit, for FLARE a 100 km circular orbit above the lunar surface
LOI	Low Orbit Insertion burn to place a vehicle in orbit around the Moon
LORAD	Lunar Operations with Reuse and Assisted Descent (HLS terminology)
LATV	Lunar All-Terrain Vehicle (FLARE terminology) for a 1-person, 3-wheel electric vehicle
LTV	Lunar Terrain Vehicle, 4-wheel, 2-person "buggy" style lunar rover
MH	Minihab module (derived from the Cygnus vehicle) component of the Gateway vehicle
MMH	Mono-Methyl Hydrazine, $CH_3(NH)NH_2$, storable rocket propellant
MMOD	Micro-Meteoroid and Orbital Debris
MPCV	Multi-Purpose Crew Vehicle, another name for NASA's Orion capsule
MPS	Main Propulsion System (HLS terminology)
NextSTEP	Next Space Technologies for Exploration Partnerships: NASA public-private partnership
NRHO	Near Rectilinear Halo Orbit (rp = 3300 km, ra = 65,000 km in HLS terminology)
NSF	National Science Foundation
OCSS	Orion Crew Survival System (OCSS), orange colored ascent/entry suits for NASA crew
OME	Orbital Maneuvering Engine, part of the ESM and lunar lander AE
PF	Performance Fraction, a method to estimate delivery of mass to certain ΔV targets
PIR	Persistently Illuminated Regions of the Moon (near the poles)
PMF	Propellant Mass Fraction (HLS terminology)
PPE	Power and Propulsion Element, component of the Gateway vehicle
PSR	Permanently Shadowed Region on the Moon (near poles)
RCS	Reaction Control System
RF	Radio Frequency
RFID	Radio Frequency Identification
RMS	Remote Manipulator System, a mechanical arm for spacecraft to move external objects
RPOD	Rendezvous, Proximity Operations and Docking phases of mating vehicles in orbit
RST	Return SpaceTug (FLARE terminology), vehicle pushing Orion + ESM to Earth from Moon
SLS	Space Launch System, a proposed family of rockets being developed by NASA
SLS B1	SLS Block 1 rocket, which delivers up to 26 mt to TLI (HLS terminology)
SM	Apollo Service Module
TA	Tug Adaptor (FLARE terminology), equipment to convert a Centaur US into a SpaceTug
TCM	Trajectory Correction Maneuver
TE	Transfer Element (HLS terminology)
TE1	TE for AE/DAE for orbit to orbit transfer or initial braking (HLS terminology)
TE2	TE for TE1 and provides orbit to orbit transfer and/or initial deorbit (HLS terminology)
TEI	Trans-Earth Injection burn to send a vehicle into the Earth's gravity field
TLI	Trans-Lunar Injection burn to send a vehicle into the Moon's gravity field
TRL	Technology Readiness Level
ULA	United Launch Alliance, rocket company formed in 2006
USA	United States of America
xEMU	Exploration Extravehicular Mobility Unit, NASA space suit for lunar surface exploration
ZBO	Zero Boil-Off, concepts to limit the loss of cryogenic LOX/LH_2 in spacecraft

APPENDIX B. CALCULATIONS

General Assumed SpaceTug Boiloff rate: 0.5%/day, 0.10 mt/day (Total Prop = 20.05 mt full).
General Assumed Descent Element (DE) Boiloff rate: 0.5%/day, 0.05 mt/day (Total Prop = 9.0 mt full).
SpaceTug Dry mass = 2.75 mt, Full Prop = 20.05 mt, Total = 22.80 mt (max lift of a F9 to LEO 28.5°)

Table B.1
Tsiolkovsky Rocket Equation Calculations for Phase A1

A1: PPE	Step1	Step2
Vehicle	Tug1PPE	n/a
Init Mass (mt)	30.80	n/a
Final Mass (mt)	11.80	n/a
Prop (mt)	19.00	n/a
Isp (s)	450.50	n/a
ΔV (km/s)	4.24	n/a
Margin (mt)	1.05	n/a
Margin (boil-off days)	10.5	n/a

To LEO: PPE is launched on an A5, Tug1PPE launched on a F9.
Vehicle Initial mass = Tug1PPE (22.8 mt) and PPE (8.0 mt).
Vehicle final mass = Tug1 PPE (3.8 mt) and PPE (8.0 mt).

Table B.2
Tsiolkovsky Rocket Equation Calculations for Phase A2

A2: HALO	Step1	Step2
Vehicle	Tug1HALO	n/a
Init Mass (mt)	30.80	n/a
Final Mass (mt)	11.70	n/a
Prop (mt)	19.10	n/a
Isp (s)	450.50	n/a
ΔV (km/s)	4.28	n/a
Margin (mt)	0.95	n/a
Margin (boil-off days)	9.5	n/a

To LEO: HALO is launched on A5, Tug1HALO launched on a F9.
Vehicle Initial mass = Tug1HALO (22.8 mt) and HALO (8.0 mt).
Vehicle final mass = Tug1 HALO (3.7 mt) and HALO (8.0 mt).
To LLFPO: SpaceTug pushes HALO and conducts RPOD with PPE.

Table B.3
Tsiolkovsky Rocket Equation Calculations for Phase B

B: Lunar Surface Precursor Equip.	Step1	Step2
Vehicle	DE1	n/a
Init Mass (mt)	15.78	n/a
Final Mass (mt)	7.88	n/a
Prop (mt)	7.90	n/a
Isp (s)	450.50	n/a
ΔV (km/s)	3.07	n/a
Margin (mt)	0.88	n/a
Margin (boil-off days)	19.4	n/a

DE1+Payload launched on a FH to TLI, DE1 lands directly on surface with payload.
DE1 Prop mass prior to lunar descent = 11.28 mt after 5 days LOX/LH_2 boil-off.
DE1 Payload mass = 4.5 mt.
0.15 mt for Comm/Nav satellites, 2.00 mt for Inflatable Hab & airlock, 0.35 mt for Logistics air/water), 0.38 mt for (2) xEMU suits, 0.30 mt for Science Trailer, 0.40 mt for ISRU pilot plant, 0.92 mt for LATV(s).

Table B.4
Tsiolkovsky Rocket Equation Calculations for Phase C1

C1: AE	Step1	Step2
Vehicle	Tug1AE1	n/a
Init Mass (mt)	31.20	n/a
Final Mass (mt)	11.85	n/a
Prop (mt)	19.35	n/a
Isp (s)	450.50	n/a
ΔV (km/s)	4.28	n/a
Margin (mt)	0.70	n/a
Margin (boil-off days)	7.0	n/a

To LEO: AE1 launched on an A5, Tug1AE1 launched on a F9.
Vehicle Initial mass = Tug1AE1 (22.80 mt) and AE1 (8.4 mt).
Vehicle final mass = Tug1AE1 (3.45 mt) and AE1 (8.4 mt).
To LLFPO: SpaceTug pushes AE1 and conducts RPOD with Gateway (if present).

Table B.5
Tsiolkovsky Rocket Equation Calculations for Phase C2

C2: RST (Return SpaceTug)	Step1	Step2
Vehicle	Tug1RST	RST
Init Mass (mt)	44.20	22.80
Final Mass (mt)	25.70	14.90
Prop (mt)	18.50	7.90
Isp (s)	450.50	450.50
ΔV (km/s)	2.40	1.88
Margin (mt)	0.15	12.15
Margin (boil-off days)	1.5	121.2

Table B5 comments.
To LEO: Tug1RST launched on F9, 2 weeks later RST launched on a F9.
Tug1RST Initial Mass = 21.40 mt after 14 days LOX/LH_2 boil-off.
RST Initial Mass = 22.80 mt (full propellant load of 20.05 mt).
To LLFPO.
Step1: Tug1RST pushes RST towards TLI and then separates and deorbits to Earth.
Step2: RST pushes itself to TLI, then to LLFPO and RPOD with Gateway (if present).

Table B.6
Tsiolkovsky Rocket Equation Calculations for Phase C3

C3: DE2	Step1	Step2
Vehicle	Tug1DE2	Tug2DE2
Init Mass (mt)	55.38	33.99
Final Mass (mt)	36.88	19.39
Prop (mt)	18.50	14.60
Isp (s)	450.50	450.50
ΔV (km/s)	1.80	2.48
Margin (mt)	0.15	4.05
Margin (boil-off days)	1.5	40.4

To LEO: Tug1DE2 launched on a F9, then 1 week later DE2 launched on a A5, then 1 week later Tug2DE2.
is launched on a F9 (Tug2DE2 launched 2 weeks after Tug1DE2).
Tug1DE2 Initial Mass = 21.40 mt after 14 days LOX/LH_2 boil-off.
Tug2DE2 Initial Mass = 22.80 mt (full propellant load of 20.05 mt).
To LLFPO.
Step1: Tug1DE2 pushes stack towards TLI and then separates and deorbits to Earth.
Step2: Tug2DE2 pushes DE2 to TLI, then to LLFPO and RPOD with either AE1 or Gateway.
Note: Tug2DE2 could be used to help DE2 deorbit AE1 in Phase D1 (lower perilune).

Table B.7
Tsiolkovsky Rocket Equation Calculations for Phase C4

C4: Crew	Step1	Step2
Vehicle	ESM	n/a
Init Mass (mt)	25.90	n/a
Final Mass (mt)	18.80	n/a
Prop (mt)	7.10	n/a
Isp (s)	320.00	n/a
ΔV (km/s)	1.01	n/a
Margin (mt)	1.50	n/a
Margin (boil-off days)	n/a	n/a

To TLI: Crew inside Orion + ESM launched on SLS Block 1.
Vehicle Initial Mass = Orion (10.4 mt) and ESM (15.5 mt).
To LLFPO.
ESM pushes Orion to LLFPO and RPOD with AE1+DE2, or Gateway (if present).
ESM propellant in LLFPO = 1.5 mt (no boil-off for storable propellant).

Table B.8
Tsiolkovsky Rocket Equation Calculations for Phase D1

D1: Crew to Lunar Surface (LS)	Step1	Step2
Vehicle	DE2	n/a
Init Mass (mt)	19.46	n/a
Final Mass (mt)	11.66	n/a
Prop (mt)	7.80	n/a
Isp (s)	450.50	n/a
ΔV (km/s)	2.26	n/a
Margin (mt)	0.26	n/a
Margin (boil-off days)	5.7	n/a

To Lunar Surface.
DE2 Prop available in LLFPO = 8.06 mt after 21 days boil-off LOX/LH_2 (loss = 0.95 mt).
Initial Vehicle Mass = AE1 (8.4 mt), DE2 (10.56 mt), and Payload (crew = 0.4 mt, tools 0.1 mt).
AE1 dry mass (4.0 mt) includes 2 xEMU suits (0.38 mt): additional 2 xEMU suits in DE1 payload.
DE2 final propellant after descent to LS = 0.26 mt, or 5.7 LOX/LH_2 boil-off days.
Note: Tug2DE2 could be used to help DE2 deorbit DE1 (requires Tug2DE2 RPOD with AE1+DE2, then Tug2DE2 pushes to lower lunar orbit and separates, then DE2 completes landing).

Table B.9
Tsiolkovsky Rocket Equation Calculations for Phase D3

D3: Crew from Lunar Surface	Step1	Step2
Vehicle	AE1	n/a
Init Mass (mt)	8.90	n/a
Final Mass (mt)	4.68	n/a
Prop (mt)	4.22	n/a
Isp (s)	320.00	n/a
ΔV (km/s)	2.02	n/a
Margin (mt)	0.18	n/a
Margin (boil-off days)	n/a	n/a

To LLFPO.
Crew in AE1 ascends to RPOD with Orion + ESM (no boil-off for storable propellant).
Initial Vehicle Mass on surface = AE1 (8.4 mt) and Payload (crew = 0.4 mt, samples = 0.1 mt).

Table B.10
Tsiolkovsky Rocket Equation Calculations for Phase E

E: Crew to Earth	Step1	Step2
Vehicle	RST	ESM
Init Mass (mt)	27.99	18.80
Final Mass (mt)	21.59	18.10
Prop (mt)	6.40	0.70
Isp (s)	450.50	320.00
ΔV (km/s)	1.15	0.12
Margin (mt)	0.04	0.80
Margin (boil-off days)	0.4	n/a

Table B10 comments.
In LLFPO.
Crew transfers from AE1 to Orion + ESM.
If no Gateway exists, jettison AE1 then conduct RPOD with RST to dock with Orion + ESM.
To Earth.
Step1: RST pushes Orion + ESM to Earth, then separates and re-enters.
Step2: ESM control Orion to EI, then separates from Orion and deorbits to Earth.

Table B.11
Tsiolkovsky Rocket Equation Calculations Comparing AE for NRHO and LLO (No LOI for DE)

LANDER DESIGN	LLO	Margin	NRHO	Margin	%increase from LLO	LLO % of NRHO
AE prop type	N2O4+MMH		N2O4+MMH			
AE prop (mt)	4.40	0.19	7.30	0.23	65.9%	60.3%
AE dry	4.00		4.00			
AE payload*	0.50		0.50			
AE total (mt)	**8.90**	1.05	**11.80**	0.69	32.6%	75.4%
DE prop type	LOX/LH2		LOX/LH2			
DE prop (mt)	9.00		15.00		66.7%	60.0%
DE dry	2.50		2.80			
DE total (mt)	**11.50**		**17.80**	Difference	54.8%	64.6%
AE + DE total (mt)	**20.40**		**29.60**	9.20	45.1%	68.9%
Descent ΔV (km/s)	2.18		2.92	0.74		
Ascent ΔV (km/s)	2.01		2.87	0.86		

A SpaceTug performs lunar insertion burn, the DE performs the deorbit burn and lands crew in AE.
Using one SpaceTug to TLI: Payload = 15.50 mt.

Table B.12
Tsiolkovsky Rocket Equation Calculation for Max Payload to TLI with one SpaceTug.

SpaceTug +">+ Payload to TLI	Step1	Step2
Vehicle	SpaceTug	n/a
Init Mass (mt)	38.30	n/a
Final Mass (mt)	18.25	n/a
Prop (mt)	20.05	n/a
Isp (s)	450.50	n/a
ΔV (km/s)	3.28	n/a

Using one SpaceTug to NRHO: Payload = 12.00 mt.

Table B.13
Tsiolkovsky Rocket Equation Calculation for Max Payload to TLI with one SpaceTug

SpaceTug + Payload NRHO	Step1	Step2
Vehicle	SpaceTug	n/a
Init Mass (mt)	34.80	n/a
Final Mass (mt)	14.75	n/a
Prop (mt)	20.05	n/a
Isp (s)	450.50	n/a
ΔV (km/s)	3.79	n/a

Using one SpaceTug to LLO: Payload = 9.5 mt.

Table B.14
Tsiolkovsky Rocket Equation Calculation for Max Payload to LLO with one SpaceTug

SpaceTug+">+Payload to LLO	Step1	Step2
Vehicle	SpaceTug	n/a
Init Mass (mt)	32.30	n/a
Final Mass (mt)	12.25	n/a
Prop (mt)	20.05	n/a
Isp (s)	450.50	n/a
ΔV (km/s)	4.28	n/a

Using two SpaceTugs to TLI: Payload = 32.90 mt.

Table B.15
Tsiolkovsky Rocket Equation Calculation for Max Payload to TLI with two SpaceTugs

SpaceTugs+">+PL to TLI	Step1	Step2
Vehicle	Tug1	Tug2
Init Mass (mt)	78.50	55.70
Final Mass (mt)	58.45	35.65
Prop (mt)	20.05	20.05
Isp (s)	450.50	450.50
ΔV (km/s)	1.30	1.97

Using two SpaceTugs to NRHO: Payload = 25.90 mt.

Table B.16
Tsiolkovsky Rocket Equation Calculation for Max Payload to NRHO with two SpaceTugs

SpaceTugs+">+PL to NRHO	Step1	Step2
Vehicle	Tug1	Tug2
Init Mass (mt)	71.50	48.70
Final Mass (mt)	51.45	28.65
Prop (mt)	20.05	20.05
Isp (s)	450.50	450.50
ΔV (km/s)	1.45	2.34

Using two SpaceTugs to LLO: Payload = 21.20 mt.

Table B.17
Tsiolkovsky Rocket Equation Calculation for Max Payload to NRHO with two SpaceTugs

SpaceTugs + PL to LLO	Step1	Step2
Vehicle	Tug1	Tug2
Init Mass (mt)	66.80	44.00
Final Mass (mt)	46.75	23.95
Prop (mt)	20.05	20.05
Isp (s)	450.50	450.50
ΔV (km/s)	1.58	2.69

Note two SpaceTugs can not deliver Orion + ESM (total = 25.90 mt) from LEO to LLO (need 3rd SpaceTug).
Using a special Lander SpaceTug to refill from surface ISRU, then deliver 11.25 mt prop to LLO.

Table B.18
Tsiolkovsky Rocket Equation Calculation for SpaceTug Max Propellant to LLO from surface

SpaceTug from LS to LLO	Step1
Vehicle	Tug1
Init Mass (mt)	24.05
Final Mass (mt)	15.25
Prop (mt)	8.80
Isp (s)	450.50
ΔV (km/s)	2.01

SpaceTug Lander Dry mass = 4.0 mt (extra 1.25 mt from SpaceTug for landing gear and surface refill equip), Prop = 20.05 mt (same as for SpaceTug), Total = 24.05 mt (launched from Earth on a FH to TLI).

Table B.19
ΔV calculation for 2.5° inclination change in LLFPO.

ORBIT PARAMETER CALCULATIONS:							
Calculate Orbital Velocity			ENTER			Moon	
	V = sqrt(G*M/r)	Altitude (km)	100		V (m/s)	1633.8	
					V (km/s)	1.634	
Calculate Orbital Period							
Eq.1	P=2*pi*sqrt(r^3/(GM))		Kepler 3rd law				
E1.2	P=2*pi*((R+H)/v)		Kepler 3rd law		P (s)	7065.2	
					P (min)	117.8	
					P (hr)	1.96	
Calculate Plane Change			ENTER				
	ΔV = 2*v*sin(Δi/2)	ΔI (deg)	2.5		ΔV (km/s)	0.071	
	convert deg to radian	ΔI (rad)	0.0436 <== converted				
						Moon	Units
			Universal Constant of Gravitation		G	6.6743E-11	m3*kg-1*s-2
			Gravity		g	1.62	m*s-2
			Mass		M	7.347E+22	kg
			calculated	GM		4.9036E+12	m3s-2
			Published Planet Gravitational Constant (GM)		μ	4.9049E+12	m3s-2
			Density		ρ		g/cc
			Mean		Radius	1737.1	km
			Equatorial		Radius	1738	km
			Polar		Radius	1736	km
			Select a radius here ==>			1737.1	km

Table B.20
Transportation Cost Summary for FLARE Required Phases

Crewed Phases (C, D, E)	F9	FH	A5	SLS	Tugs	Notes
C1a: AE1 to LEO	1		1		1	AE1 and Tug1AE1
C1b: AE1 to LLFPO						Tug1 pushes AE1
C2a: RST to LEO	2				2	Tug1 and RST
C2b: RST to LLFPO						Tug1 pushes RST
C3a: DE2 to LEO	2		1		2	DE2 amd 2 Tugs
C3b: DE2 to LLFPO						2 Tugs push DE2
C4a: Crew to TLI				1		Orion + ESM to TLI
C4b: Crew to LLFPO						ESM push to LLFPO
D1a: Refuel demo						Between Tugs/DE2
D1b: Crew to surface						DE2 pushes to LS
D2: Surface Campaign						
D3a: Crew from surface						AE1 pushes to LLFPO
E1a: Crew in LLFPO						RST pushes to TEI
E1b: Crew to Earth						ESM push to Entry
Cost ($M) per vehicle	$62	$90	$153	$2,000	$80	
Vehicles	5	0	2	1	5	Total ($M)
Crewed Phase Costs	$310	$0	$306	$2,000	$400	$3,016

Table B.21
Transportation Cost Summary for FLARE Optional Phase A (Gateway in LLFPO)

Phase A (Gateway)	F9	FH	A5	SLS	Tugs	Notes
A1a: PPE to LEO	1		1			Deliver to LEO
A1b: PPE to LLFPO					1	Tug pushes to LLFPO
A2a: HALO to LEO	1		1			Deliver to LEO
A2b: HALO to LLFPO					1	Tug pushes to LLFPO
Cost ($M) per vehicle	$62	$90	$153	$2,000	$80	
Phase A (PPE & HALO)	2	0	2	0	2	Total ($M)
Total	$124	$0	$306	$0	$160	$590

Table B.22
Transportation Cost Summary for FLARE Optional Phase B (Lunar Surface Precursor Hardware)

Phase B (Precursor Hardware)	F9	FH	A5	SLS	Tugs	Notes
B1a: DE1+PC1 to TLI		1				Deliver to TLI
B1b: DE1+PC1 to LS						DE1 pushes to LS
Cost ($M) per vehicle	$62	$90	$153	$2,000	$80	
Phase B (LS Precursor Hardware)	0	1	0	0	0	Total ($M)
Total	$0	$90	$0	$0	$0	$90

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