Evolution of MPCV Service Module Propulsion and GNC Interface Requirements

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The Orion Multi-Purpose Crew Vehicle Service Module Propulsion Subsystem provides propulsion for the integrated Crew and Service Module. Updates in the exploration architecture between Constellation and MPCV as well as NASA's partnership with the European Space Agency have resulted in design changes to the SM Propulsion Subsystem and updates to the Propulsion interface requirements with Guidance Navigation and Control. This paper focuses on the Propulsion and GNC interface requirement updates between the Constellation Service Module and the European Service Module and how the requirement updates were driven or supported by architecture updates and the desired use of hardware with heritage to United States and European spacecraft for the Exploration Missions, EM-1 and EM-2.

I. Introduction

OVER the last five years there have been updates to the National Aeronautics and Space Administration (NASA) human exploration architecture with the implementation of the Multi-Purpose Crew Vehicle (MPCV), Space Launch System (SLS), and Ground Systems Development and Operations (GSDO) programs as compared to the Constellation Program (CxP). Similar to Orion development under CxP, the MPCV consists of a Crew Module (CM), Service Module (SM), and Launch Abort System (LAS), and the prime contractor and integrator for MPCV

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remains Lockheed Martin. However, a significant change in the Orion development occurred in 2012 with NASA and the European Space Agency (ESA) entering into a partnership for the delivery of the European Service Module (ESM) that will be integrated into the MPCV. The prime contractor for the ESM is Airbus Defense and Space (AD&S).[1-3]

In addition to the planned flight test, Exploration Flight Test 1 (EFT-1), MPCV currently has two primary missions: Exploration Mission 1 (EM-1) in 2017/2018 and EM-2 in 2021. The EM-1 mission, described in more detal later in this paper, is the first uncrewed MPCV mission to a Lunar Distant Retrograde Orbit (DRO) for six days and is the first SLS launch. EM-2 is the first crewed MPCV mission to a High Lunar Orbit (HLO) for three days. Other Design Reference Missions (DRMs) were previously considered for the Exploration Missions; specifically EM-1 was a lunar free-return flyby mission, a subset of the EM-2 HLO mission. The current baseline for the EM-1 mission is now a DRO and should not stress or require any additional performance relative to the ESM DRMs as documented. It has been shown that the MPCV ESM design allows sufficient performance for these missions even when considering the updates to the vehicle configuration and DRM changes from CxP.

The SM has maintained key functions through the implementation of the MPCV and the partnership with ESA. The SM provides propulsion for the integrated Crew and Service Module (CSM) after CSM separation from the launch vehicle, generates electrical power, regulates heat for the spacecraft, and stores commodities for life support. The SM houses the SM Propulsion Subsystem (PSS), which is a storable, pressure-fed bipropellant system feeding a main engine with thrust vector control (TVC), eight auxiliary engines, and a suite of reaction control system (RCS) engines. The total usable propellant load of 18,964 lbm (8600 kg) is also the same between the CxP SM and the ESM. Figure 1 provides an illustration of the CxP SM PSS as installed in the SM on the left and the ESM PSS as installed in the SM on the right.[4,5]

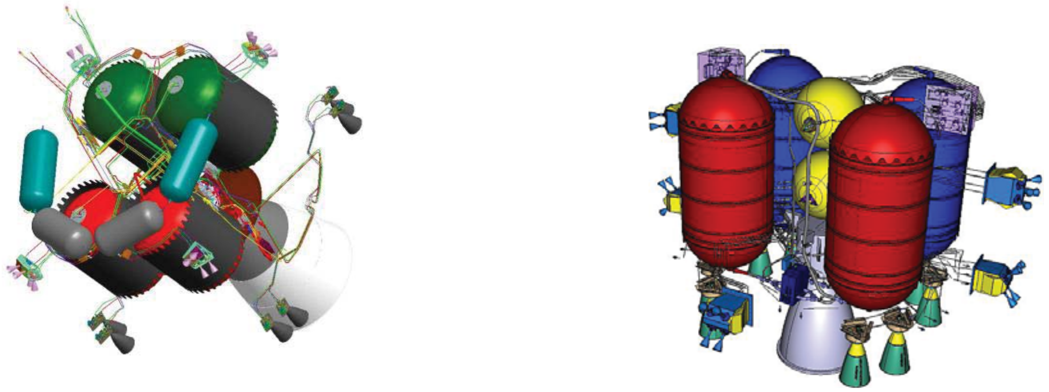

Figure 1. Service Module Propulsion Subsystem Configurations for CxP SM (left) and ESM (right).

While the function, general description, and usable propellant load of the SM PSS is upheld between CxP[4] and MPCV[5], the updates in the exploration architecture, and the partnership with ESA have resulted in updates to the Propulsion interface requirements with Guidance, Navigation, and Control (GNC). Some examples of these requirements updates are in the area of engine performance. The main engine thrust, specific impulse, and number of vacuum starts have changed with the updates to trajectories and incorporation of the Space Shuttle Orbital Maneuvering System Engine (OMS-E). The TVC performance requirements have also been updated to enable the use of the Space Shuttle Orbital Maneuvering System TVC, accounting for the new range and slew rates that can be accommodated by MPCV. The auxiliary engine specific impulse, single burn duration, vacuum starts, and duty cycles have been revisited with the use of a different variant of the Aerojet Rocketdyne R-4D-11 and EM-1/EM-2 mission planning. The RCS thrust and minimum electrical pulse width (EPW), have changed due to the use of the 220 N reaction control engines, heritage to the European Automated Transfer Vehicle (ATV). The number and locations of RCS thrusters have also changed with the ESM design.

This paper describes the EM-1/EM-2 missions, provides an overview of the ESM PSS design, and gives examples of Propulsion and GNC interface requirement updates between the CxP SM and the ESM, identifying how the requirement updates were driven or supported by architecture updates and the desired use of hardware with heritage to United States and European spacecraft for the ESM in the EM-1 and EM-2 missions.

II. Mission Description

The ESM is currently required to execute EM-1 and EM-2 with sufficient vehicle performance, as documented in the ESM System Requirements Document. The EM-1 and EM-2 DRMs are outlined in Figures 2 and 3.

EM-1 is a 25 to 26 day mission to a DRO that is approximately 37,797 nmi (70,000 km) away from the Moon, as illustrated in Figure 2. The SLS places the Interim Cryogenic Propulsion Stage (ICPS) and MPCV into an elliptical Earth orbit. The ICPS raises perigee to a stable orbit condition and then places MPCV on a lunar trajectory by performing the Trans-Lunar Injection (TLI) burn. TLI targets the Outbound Powered Flyby (OPF) burn performed at 54 nmi (100 km). MPCV performs and uses the OPF to target the Distant Retrograde orbit Insertion (DRI) burn. MPCV stays in the DRO for approximately 6 days, after which it performs the Distant Retrograde orbit Departure (DRD) burn. This burn targets the Return Powered Flyby (RPF) burn performed at 54 nmi (100 km) perilune altitude. This RPF burn is used to target an Earth entry condition for a landing off the coast of California.

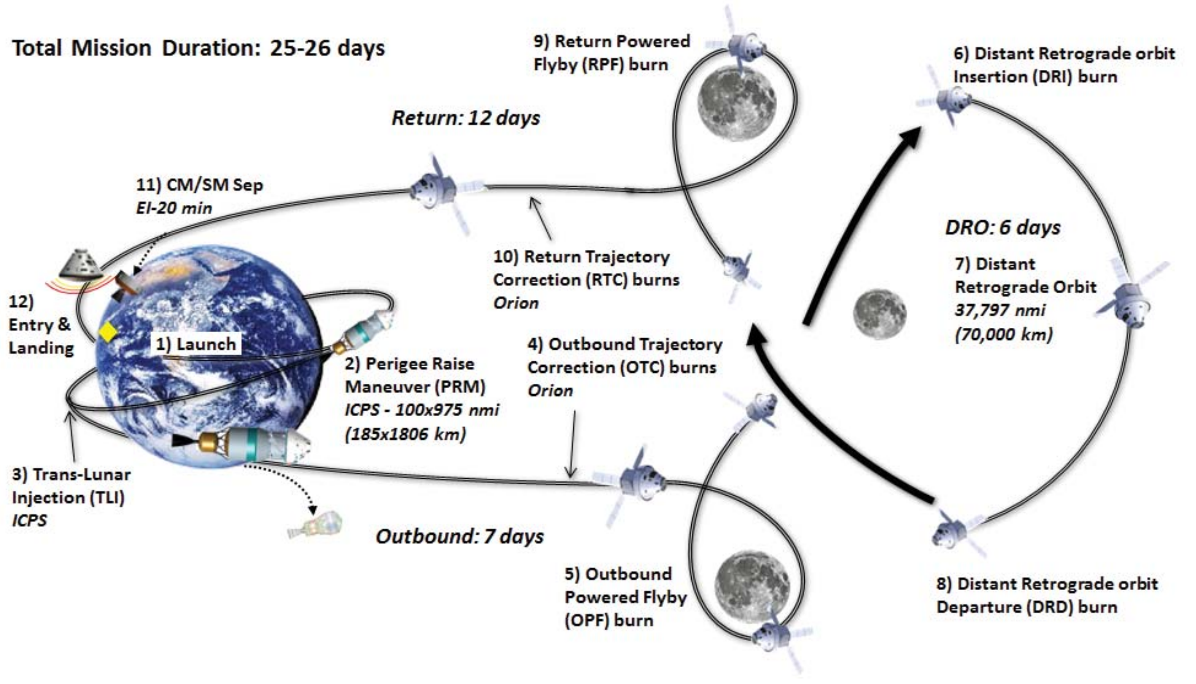

Figure 2. EM-1 Mission Overview.

Figure 3 shows that EM-2 starts with the SLS placing the ICPS and MPCV into an elliptical Earth orbit. The ICPS places MPCV on a free return lunar flyby trajectory. After TLI, MPCV performs an Outbound Trajectory Adjust (OTA) burn, which targets the desired Lunar Orbit Insertion (LOI) condition. This, subsequently, takes the MPCV off of the free return trajectory. MPCV then performs LOI to insert into a high elliptical lunar orbit. MPCV stays in lunar orbit for about three days, after which it performs the Trans-Earth Injection (TEI) burn. TEI targets an Earth entry condition for a landing off the coast of California. This mission typically has a total mission duration of 10 to 14 days.

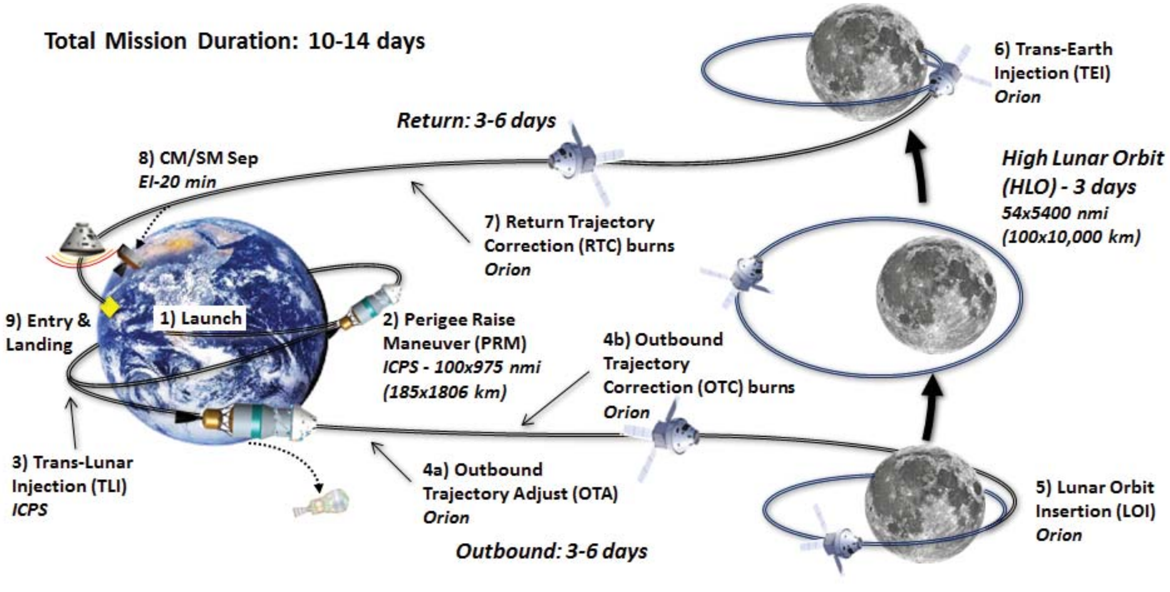

Figure 3. EM-2 Mission Overview.

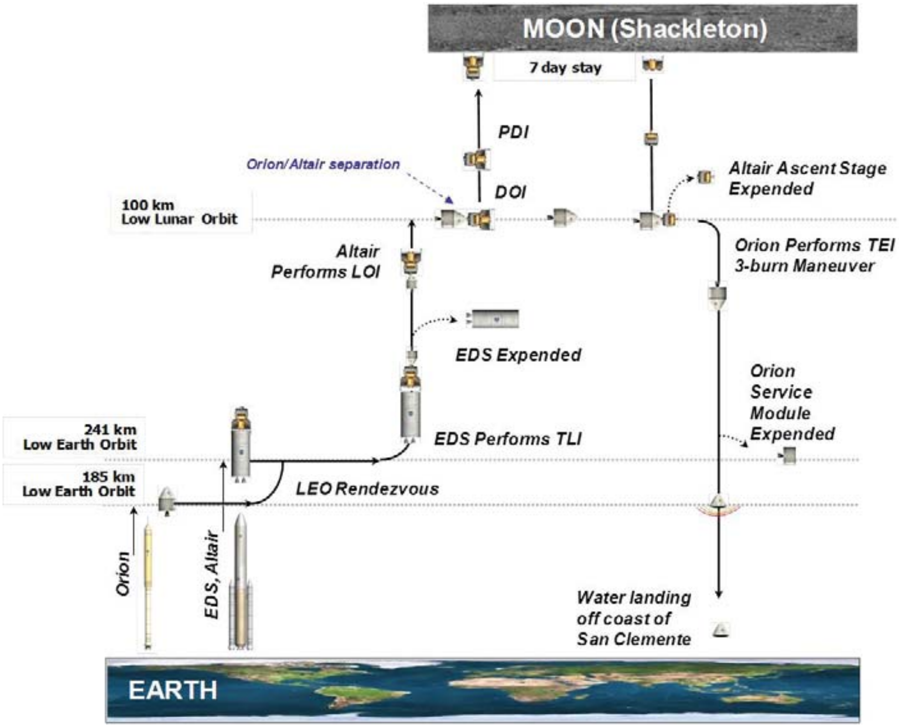

Figure 4. CxP LLO Mission Overview.

One major difference between the missions defined for CxP and the current EM-1 and EM-2 reference missions is that for CxP there was a lunar lander, Altair, which performed the LOI burn to get into a lunar orbit. Currently MPCV has to be able to get into and out of lunar orbit with only the capability of its own ESM. Figure 4 outlines a CxP DRM for a Low Lunar Orbit (LLO) mission involving a 1.5 launch architecture. Table 1 provides a high level list of the major mission architecture differences between the CxP mission design requirements and the EM-1/EM-2 mission design requirements. The reference missions outlined in Table 1 require a different amount of performance capability. The CxP LLO sortie mission required a delta-V of approximately 3901 ft/s (1189 m/s) or 6686 ft/s (2038 m/s) if Orion had to perform LOI as well, whereas EM-2 HLO requires 3556 ft/s (1084 m/s). Note that the LLO sortie mission is provided for reference; however, CxP also included different missions that required increased vehicle performance.

Table 1. CxP and EM-2 HLO Comparison.

	Constellation	**EM-2**
Mission Type	LLO Sortie	HLO
Lunar Lander	Yes, Altair	No
Launches	Two launches, Ares I and Ares V	One SLS Launch
Mission Duration	~23 Days	9-12 Days
Rendezvous	In LEO ~315 ft/s (96 m/s)	N/A
Upper Stage Separation	From Ares I launch vehicle in LEO, Altair separated from upper stage post-TLI	From upper stage post-TLI
Outbound Trajectory Correction Burns	No, performed by Altair	Yes
Free Return Trajectory	No	Yes
OTA	N/A	Takes MPCV off of free return, ~33 ft/s (10 m/s)
LOI	No, performed by Altair ~2,786 ft/s (849 m/s)	Performed by MPCV ~1358 ft/s (414 m/s)
Time in Lunar Orbit	8 Days	3 Days
Orbit Maintenance	No, only a two-burn orbit cleanup to align orbit with lunar lander ascent ~85 ft/s (26 m/s)	Yes ~33 ft/s (10 m/s)
TEI	Three-Burn sequence ~3,500 ft/s (1,067 m/s)	One Burn ~2,133 ft/s (650 m/s)
Landing	Off the cost of California	Off the coast of California

III. ESM Propulsion Subsystem Design

The ESM PSS is a pressure-fed, bipropellant liquid propulsion system which utilizes hypergolic propellants. Monomethylhydrazine (MMH) and Mixed Oxides of Nitrogen (MON-3) are the fuel and oxidizer respectively.[5] The subsystem can be grouped into five major assemblies, the pressurization assembly, the propellant storage and distribution assembly, the main engine assembly, the reaction control assembly, and the auxiliary engine assembly. Figure 5 shows a schematic of the ESM PSS, incorporating updates from the ESM system preliminary design review. Heritage hardware and heritage hardware designs have been utilized wherever possible in the propulsion subsystem in order to expedite development and minimize cost.

The pressurization assembly utilizes gaseous helium for tank pressurization. Helium is stored at 5800 psi (40 MPa) in two high pressure storage bottles, one for each propellant. The bottles are spherical composite overwrapped pressure vessels (COPVs) and are a heritage design from both the Ariane EPS upper stage and the ATV. Helium pressure regulation and delivery is performed by an Electronic Pressure Regulation (EPR) system. It consists of serial solenoid valves which open on demand to maintain tank ullage pressures within the specified pressure dead band. A latching valve is placed upstream of the solenoids for isolation and an orifice is placed downstream of the solenoids to choke the flow of helium. The active components of the EPR system are parallel redundant to provide failure tolerance.

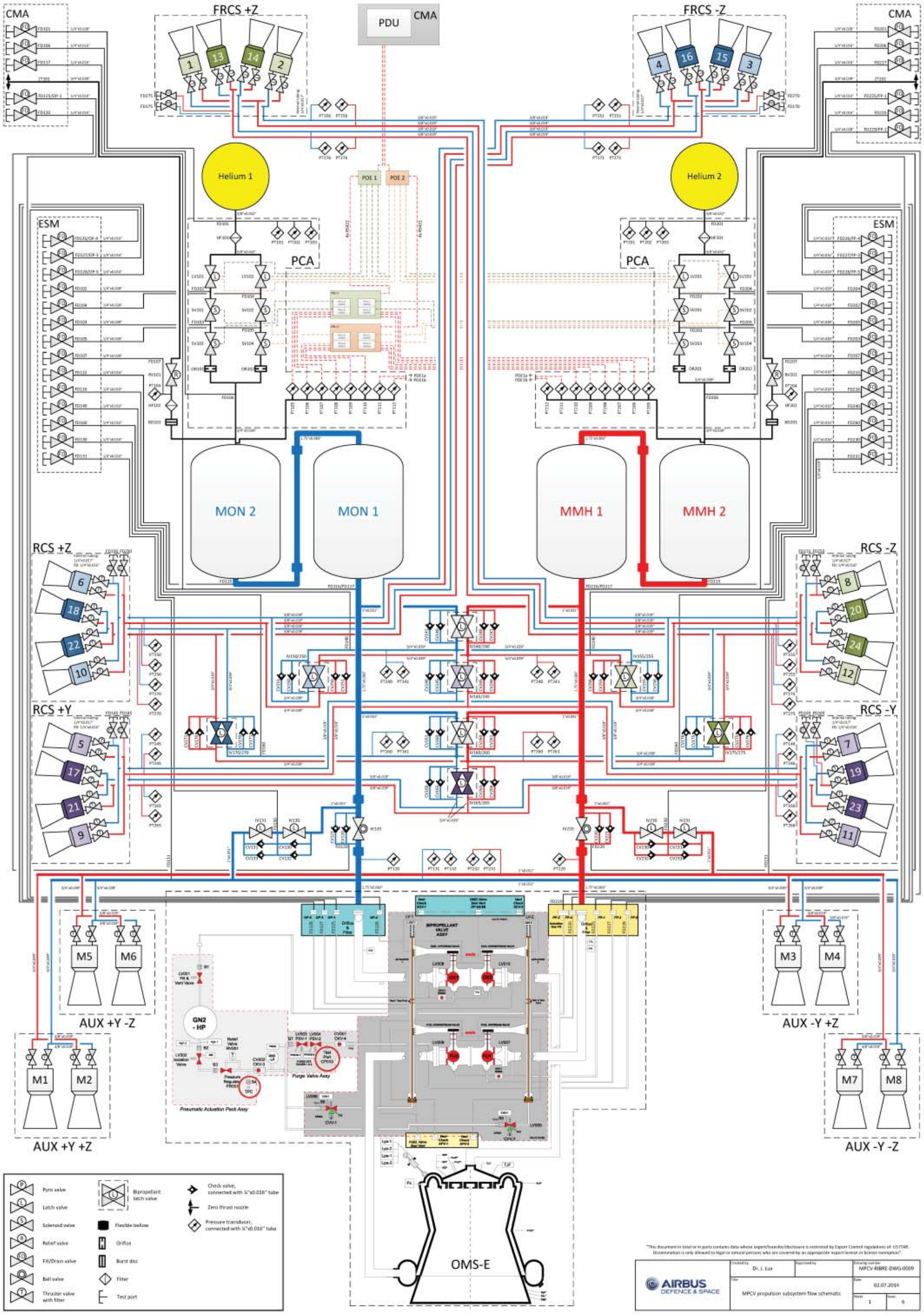

Figure 5. ESM Propulsion Schematic (Post ESM Preliminary Design Review).

The propellant storage and distribution assembly consists of four all-metal propellant tanks, two per commodity. The tank volumes are all identical due to the equal volumetric mixture ratio used by the engines. The tanks are titanium and are plumbed in a serial configuration with the upstream tank being pressurized and the downstream tank feeding the propellant manifolds to the engines. The downstream tanks for each commodity are equipped with Propellant Management Devices (PMDs) to ensure gas-free propellant is provided to the engines. The upstream tanks are equipped with Burst Disk/Relief Valves (BDRVs) to protect against tank over-pressurization. These four propellant tanks provide fuel and oxidizer to all three classes of engines on the Service Module. The tank design is heritage from model OST-23 and the PMD design is heritage from the ATV propellant tanks.

The main engine assembly consists of the OMS-E and the associated TVC assembly from the Space Shuttle Program. Both the OMS-E and TVC are provided to ESA and AD&S by NASA for use on the ESM. The OMS-E is the primary source for providing translational thrust.

The auxiliary engine assembly utilizes eight R-4D-11 thrusters, each of which provides 110 lbf (490 N) of thrust. The primary function of the auxiliary thrusters is to provide a redundant source of translational thrust in the event of a loss of function of the OMS-E. In the contingency scenario of an abort to orbit, the auxiliary engines are used in conjunction with the OMS-E to provide a higher overall thrust level. Nominally, the auxiliary engines are also used during separation from the SLS upper stage and for trajectory correction burns.

The reaction control assembly consists of 24 engines that have heritage from the ATV. The engines are configured in two redundant strings of twelve engines and each string of twelve engines is isolated into three groups of four engines. These 50 lbf (220 N) engines provide pitch, yaw, and roll control of the spacecraft for all required maneuvers. Additionally they can be utilized to back up the auxiliary engines for low thrust translational maneuvers.

IV. Service Module Propulsion and GNC Interfaces

Throughout CxP and MPCV the Propulsion and GNC subsystems have worked together to develop interface requirements that both identify the GNC need and the PSS capability. The SM PSS has always leveraged existing engine designs that meet the high level GNC needs and identified operational limitations for GNC and mission analysis planning. In this section the key performance requirements and design of the engines and TVC are contrasted between CxP and MPCV and the impacts to GNC design are highlighted.

A. Main Engine and TVC

For CxP SM, the Orion Main Engine (OME) was a new engine under development with key characteristics derived from the Space Shuttle OMS-E and a new TVC system[6], while the ESM will utilize the actual Space Shuttle OMS-E and OMS-E TVC[7]. Table 2 outlines key performance parameter differences between the CxP SM and the ESM.

The OME was sized for 7500 lbf (33.3 kN) thrust and 328 seconds specific impulse at standard inlet conditions while the OMS-E is 6000 lbf (26.6 kN) and 315 seconds specific impulse at standard inlet conditions. The OME design was derived from the heritage OMS-E and the higher thrust was achieved by increasing the chamber pressure from 125 psia (0.86 MPa) to 150 psia (1.03 MPa). The higher specific impulse was achieved through an increase in chamber pressure, change in mixture ratio from 1.65 to 1.85, and an increase in the nozzle area ratio from 55:1 to 150:1.

Table 2. Main Engine and TVC Key Performance Parameters.

	CxP SM	ESM
Nominal Thrust	7500 lbf (33.3 kN)	6000 lbf (26.6 kN)
Nominal Specific Impulse	328 s	315 s
Number of vacuum starts per mission	18	10
TVC Range	+/- 8 degrees	+/- 6 degrees
TVC Slew Rate	8-10 degrees/sec	3-6 degrees/sec

The higher thrust main engine was needed for CxP due to requirements for maintaining continuous abort coverage throughout all ascent phases with the Ares I trajectory. Given the CxP vehicle thrust-to-weight ratio, GNC had very little margin with respect to abort coverage requirements, especially for the ISS mission that has an additional requirement to avoid landing in the North Atlantic Ocean. See Figure 6 for a depiction of the CxP ascent timeline. GNC developed abort modes using the thrust of the main engine and auxiliary thrusters that included Retrograde Transatlantic Abort Landing (RTAL) and Transatlantic Abort Landing (TAL). For high-inclination International Space Station (ISS) missions, RTAL uses SM thrust in a retrograde manner to keep the CM landing footprint near the coast of Newfoundland, and TAL uses SM thrust in a posigrade manner to keep the CM landing footprint near the coast of Ireland. Abort overlap for this mission is measured from the start of capability to reach coastal waters near Ireland (TAL) to the end of capability to remain near Canada (RTAL). The 7500 lbf main engine allowed an abort overlap between the RTAL and TAL; however, with the reduced thrust and TVC capability of the OMS-E there would have been a gap in abort coverage for the Ares ISS trajectories.

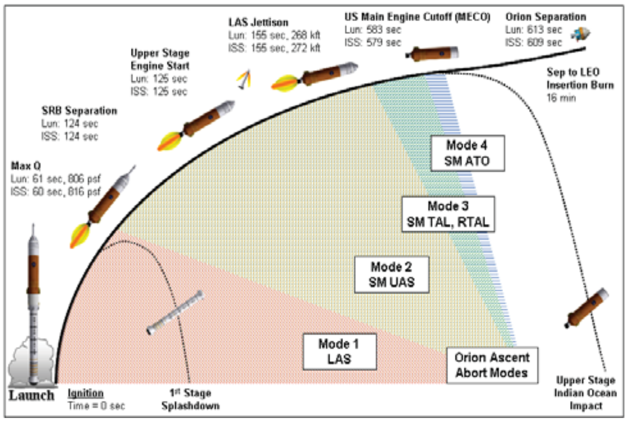

Figure 6. CxP Ascent Timeline with Abort Modes

The CxP Ares trajectories had a droop in altitude close to the RTAL/TAL overlap region. The current SLS trajectories, with a higher performing core stage, maintain a positive flight path angle throughout the SM abort regions. This results in less gravity loss in trying to maintain the minimum MPCV powered flight altitude and allows the ESM thrust to be pointed more optimally to improve range. For these SLS ISS trajectories, analysis results show an abort overlap of 10 seconds for an SM configuration with the CxP 7500 lbf main engine, and the ESM configuration with the 6000 lbf main engine only has a 2 second loss on the ISS RTAL boundary due to the reduced thrust of the OMS-E versus the OME. This loss is because the vehicle has a reduced delta-V capability for a given available duration of the burn. The TAL boundary is relatively unaffected for the ESM configuration, having a small delta-V loss due only to the Isp difference between the engines. Thus, the change in launch vehicle trajectory allowed for a system with reduced thrust to still meet ascent abort coverage requirements.

When comparing the use of the OME against the OMS-E for the EM-2 HLO mission with nominal control mass and maximum available propellant, the OME would provide 170 ft/s (52 m/s) of additional delta V performance due to the higher specific impulse. The higher thrust OME also provides slightly better delta V performance than the OMS-E due to higher gravity losses with the OMS-E. However, this does not greatly impact the nominal mission unless the burn is performed close to a central body. While the lower thrust and specific impulse has impacts on vehicle performance, the lower chamber pressure of the OMS-E provides the opportunity for lower system pressures, as compared to the OME. Also, the lower thrust of the OMS-E is beneficial for the loads on the ESM solar arrays during LOI and TEI.

Another key performance parameter for the main engine is the number of vacuum starts per mission. Both the OME and OMS-E use a self-contained pneumatic pack to store nitrogen for pneumatically operating its bipropellant valve and performing the post burn fuel purge. While the OMS-E is specified to have 500 starts over a range of start conditions in its lifetime, the number of starts per mission is limited to 10 by the amount of nitrogen that is available in its pneumatic pack. Note that each opening of the bipropellant valve or purge operation uses a certain amount of nitrogen, and there is some level of nitrogen leakage during the mission, so 10 start and purge cycles is assumed per mission due to estimated nitrogen consumption and leak rates. Because the worst case CxP mission profiles included as many as 11 starts nominal plus 7 starts for contingency operations, the OME pneumatic pack was being sized to account for these operations as well as nominal leakage over the mission duration, giving a total of 18 starts. The MPCV design reference missions now take into account the 10 OMS-E starts constraint as part of GNC mission planning and design.

Finally, the CxP SM TVC minimum range of motion was +/- 8 degrees and the slew rate was between 8 and 10 degrees per second while the ESM TVC has a minimum range of +/- 6 degrees and a slew rate between 3 and 6 degrees per second. The minimum range of motion of the TVC and the high thrust OME were selected during CxP to provide adequate angular control authority for the CxP missions and aborts. The slew rate for CxP SM TVC was limited to 10 degrees per second to avoid OME structural damage if the TVC actuator mechanical stop was reached, and the minimum slew rate was selected to provide margin on acceptable abort performance. The ESM TVC is the Shuttle OMS-E TVC, and the existing range and slew rates were assessed to be adequate against MPCV needs for aborts and for steering during nominal burns for EM-1 and EM-2. Work is ongoing to finalize TVC requirements for frequency response, command threshold, and stiction.

B. Auxiliary Engines

The auxiliary engines in both CxP SM and ESM are the Aerojet R-4D-11. However, the CxP SM used the 164:1 area ratio R-4D-11[8] while the 300:1 area ratio R-4D-11 has been selected for use on the ESM. The auxiliary engines are used to both back up the main engine in the case of failure and to perform the nominal MPCV-SLS separation burn and trajectory correction burns. Table 3 compares some key performance parameters for the auxiliary engines between the CxP SM and ESM.

Table 3. Auxiliary Engine Driving Key Performance Parameters

	CxP SM	ESM
Nominal Specific Impulse	311 s	315 s
Single Burn Duration	4,500 s	7,200 s
Expected Duty Cycles	> 50%	> 50%
Number of Starts	10,000	7,200

The selection of the 300:1 R-4D-11 for the ESM allows for higher auxiliary engine specific impulse as compared to the CxP SM. The CxP SM auxiliary engines have a nominal specific impulse of 311 seconds, while the ESM auxiliary engines have a nominal specific impulse of 315 seconds. This represents a DV performance improvement of 66 ft/s (20 m/s) for the ESM configuration relative to the CxP SM, when also accounting for inefficiency factors for off pulsing and plume impingement. Although the 300:1 R-4D-11 provides higher specific impulse, the 164:1 configuration was selected for the CxP SM for accommodation reasons. The CxP SM had the larger OME, causing the auxiliary engines to be located on the exterior of the radiator and therefore needing the smaller nozzle to fit under the fairing. Having the auxiliary engines radially outward also improves vehicle controllability due to the increased moment arm between the auxiliary engine and the vehicle center of mass. With the OMS-E on the ESM, the auxiliary engines can be located radially inward of the radiator, and the 300:1 nozzle can be accommodated to provide increased specific impulse performance.

The single burn duration for the auxiliary engines increased from 4500 seconds to 7200 seconds between the CxP SM and the ESM. The longest burn with the auxiliary engines for the CxP SM was the TEI-1, performed by the auxiliary engines if the main engine had failed. In this case an estimated 1,550 lbm (705 kg) of propellant was used by the auxiliary engines, and the single burn duration was calculated using three sigma low specific impulse and nominal thrust, arriving at 4330 seconds. 4500 seconds was selected for margin. For ESM, the longest possible burn with the auxiliary engines is calculated by assuming all propellant would be consumed by the auxiliary engines. This results in a single burn duration of 7200 seconds, which covers the potential for a single or two-burn TEI backup and for a missed LOI abort for EM-1 and EM-2. Both the CxP and ESM requirements for single burn duration are within the heritage operation of the R-4D-11.

While burns with the main engine use TVC to control the vehicle, burns with the auxiliary engines use off-pulsing. Off-pulsing accomplishes pitch and yaw attitude control by pulsing some of the engines as the others continue to fire in order to get the required torque. This is required for all auxiliary burns in order to balance the torques caused by the center of mass offset of the MPCV vehicle. The location of the auxiliary engines directly affects the amount of required off-pulsing for vehicle pointing. In working with ESA and AD&S to select the auxiliary engine locations for the ESM, a trade study of various configurations was done to evaluate the effective thrust of the auxiliary engines during a burn, taking into account the off-pulsing required to maintain pointing for a range of vehicle center of mass locations. In addition, the minimum duty cycle of the off-pulsing engine(s) for the same range of vehicle center of mass locations was evaluated because there is a minimum 50% duty cycle for the engines which must be obeyed during auxiliary burns. With the CxP location of the auxiliary engines farther from the vehicle centerline they had a greater moment arm and therefore required less off-pulsing than the current MPCV ESM configuration. The CxP auxiliary configuration gave an effective thrust of approximately 95-98% with a minimum Auxiliary engine duty cycle of 80-90% during the burn. The current MPCV auxiliary engine locations result in a greater thrust loss due to off-pulsing, with an effective thrust of 93-97%. The minimum thruster duty cycle is also lower at 65-80%. These results still meet the constraint requirements, and GNC accounts for the thrust loss due to off-pulsing during auxiliary burns in mission planning and design.

Although the auxiliary engines are used for a small number of burns, they have a high number of starts because of the off-pulsing operation. To effectively perform thrust vector pointing and attitude control the pulsing engines are off pulsed at a 1 Hz frequency, so it is possible that an engine could pulse every second for its total cumulative life. This gives a total cycle life of the auxiliary engines of 7200 cycles. The cumulative burn time of the auxiliary engine is enveloped by the single burn duration requirement because the margin in the single burn duration requirement is greater than the duration of nominal burns performed by the auxiliary engines.

C. RCS Engines

The RCS engines for CxP were the 25 lbf (110 N) Aerojet R-1E while the RCS engines for MPCV are the 50 lbf (220 N) reaction control thruster developed for ATV[9]. Table 4 compares some key performance parameters between the RCS engines between CxP SM and ESM, including the thrust, minimum electrical pulse width (EPW) and number of RCS engines.

Table 4. RCS Key Performance Parameters

	CxP SM	ESM
Nominal Thrust	25 lbf (110 N)	50 lbf (220 N)
Minimum EPW	40 ms	28 ms
Number of Engines	16	24

The MPCV RCS engines are used for 6 degree-of-freedom vehicle control including attitude maneuvering and small translational burns. During the CxP Orion SM design, an iterative approach between propulsion, GNC, and vehicle configuration was used to determine the number and location of the RCS engines in order to meet mission objectives for the integrated vehicle. With the current arrangement of ESA providing the ESM the ESM, requirements were needed that would allow ESA and AD&S to complete ESM RCS engine selection and layout independently for the propulsion subsystem. Two critical areas for RCS spacecraft control are the force and torque authority and the control precision, so requirements were developed in these areas. The control authority requirement was written in terms of the minimum translational and rotational accelerations needed in each axis. This effectively allows for independent design of the vehicle mass properties, RCS engine thrust, and location. For control precision, requirements were written to specify the maximum tolerable single engine impulse and the maximum allowable cross-coupling residuals between rotation and translation. The control precision requirements are especially important to ensure MPCV capability for both manual and autonomous rendezvous and docking.[10]

While the CxP SM had 2 strings each with 8 RCS engines, the ESM baseline design now has 2 strings of 12 engines. Also, the CxP SM RCS engines were arranged in a canted and skewed configuration, with each engine able to provide impulse for pitch/yaw and roll control. This was a benefit for optimizing redundancy with a minimum number of thrusters and for having small impulses for attitude control during the CxP extended lunar duration missions in which the SM was required to hold attitude during a six month stay in lunar orbit. The ESM RCS engines are arranged in an orthogonal configuration, and the ESM engine location was driven primarily by accommodation of the new solar array “x-wing” configuration in its stowed condition as well as the need to minimize plume impingement from the RCS engines on the solar arrays.[11] Having the control authority and precision requirements allowed AD&S to design an RCS engine configuration that balances the vehicle controllability needs with the solar array concerns. The ATV-heritage RCS engines selected for ESM have twice the thrust of the CxP SM RCS engines, which is a benefit to the control authority available. The ESM RCS engines also have a smaller minimum on-time than the CxP SM RCS engines, so the increased thrust can be accommodated while still having a small enough impulse to meet the rendezvous and docking control requirements.

V. Summary

This paper reviewed the EM-1/EM-2 missions, an overview of the ESM PSS design, and examples of Propulsion and GNC interface requirement updates between the CxP SM and the ESM. The use of hardware heritage to the United States and European spacecraft and its impact on GNC design and planning was highlighted. Assessments have shown that heritage engine performance should meet the EM-1 and EM-2 mission needs and MPCV performance requirements. Forward work for the ESM PSS includes the completion of the PSS preliminary design review and transition to critical design review.

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