NASA Propulsion Investments for Exploration and Science

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The National Aeronautics and Space Administration (NASA) invests in chemical and electric propulsion systems to achieve future mission objectives for both human exploration and robotic science. Propulsion system requirements for human missions are derived from the exploration architecture being implemented in the Constellation Program. The Constellation Program first develops a system consisting of the Ares I launch vehicle and Orion spacecraft to access the Space Station, then builds on this initial system with the heavy-lift Ares V launch vehicle, Earth departure stage, and lunar module to enable missions to the lunar surface. A variety of chemical engines for all mission phases including primary propulsion, reaction control, abort, lunar ascent, and lunar descent are under development or are in early risk reduction to meet the specific requirements of the Ares I and V launch vehicles, Orion crew and service modules, and Altair lunar module. Exploration propulsion systems draw from Apollo, space shuttle, and commercial heritage and are applied across the Constellation architecture vehicles. Selection of these launch systems and engines is driven by numerous factors including development cost, existing infrastructure, operations cost, and reliability. Incorporation of green systems for sustained operations and extensibility into future systems is an additional consideration for system design. Science missions will directly benefit from the development of Constellation launch systems, and are making advancements in electric and chemical propulsion systems for challenging deep space, rendezvous, and sample return missions. Both Hall effect and ion electric propulsion systems are in development or qualification to address the range of NASA's Heliophysics, Planetary Science, and Astrophysics mission requirements. These address the spectrum of potential requirements from cost-capped missions to enabling challenging high delta-v, long-life missions. Additionally, a high specific impulse chemical engine is in development that will add additional capability to performance-demanding space science missions. In summary, the paper provides a survey of current NASA development and risk reduction propulsion investments for exploration and science.

INTRODUCTION

The National Aeronautics and Space Administration (NASA) has a unified plan to meet goals for both human and robotic exploration. The U.S. Space Exploration Policy sets goals to complete the International Space Station (ISS), retire the space shuttle, and develop launch and crew systems that will enable both access to and the ability to move beyond low-Earth orbit (LEO). These projects will provide for a sustained human presence on the Moon and prepare for destinations beyond. Using both launch systems developed for humans and advanced chemical and electrical propulsion developed for scientific spacecraft, robotic science missions will simultaneously venture to more challenging destinations. Developing and applying several critical propulsion systems is vital to achieving these goals. The following paper provides a summary of the current propulsion systems under development for the Ares I crew launch vehicle, Orion crew exploration vehicle (CEV), and current propulsion systems under advanced development for enabling In-Space Propulsion Projects. Key requirements, element descriptions, design heritage, development challenges, and current status (including recent testing) are included. Also described are propulsion systems under consideration for the Ares V cargo launch vehicle and the Altair lunar lander.

Architectural Requirements for Exploration and Science Missions

Propulsion system requirements are derived from the overall human space flight architecture developed in NASA's Exploration Systems Architecture Study (September 2005). The architecture will enable transport of six-member crews to ISS and progressively builds towards lunar access for a crew of four. A single launch vehicle and capsule will provide initial LEO/ISS access capability, followed by a dual-launch Earth rendezvous assembly sequence that departs LEO for the Moon. This architecture is implemented by the Constellation Program and includes Orion, Ares I and V, Altair, Ground Systems, Mission Systems, Future Destination Surface Systems, and interfaces with the ISS. Performance, risk, and cost drive the use of existing assets and infrastructure. Heritage propulsion systems must have physical and performance properties that can be incorporated into new vehicles while also having the flexibility to go beyond their original applications. The overall architecture requires expanded propulsion systems that are much greater than the mission-driven requirements of Apollo.

NASA's robotic science missions are organized into Earth science, planetary science, heliophysics, and astrophysics pursuits. The architecture that influences propulsion systems is derived from a mix of small, medium, and large missions used to achieve the science objectives. Flagship missions, in contrast to mission lines, are individual strategic missions and are in excess of $1 billion. With competitive cost-capped small and midsized missions, NASA also considers using existing or requalified propulsion systems adapted to new mission specifications. Alternatively, advancements to a propulsion system can be fundamentally enabling to the mission itself, requiring specific new advancements in chemical and/or electric propulsion.

THE ARES SYSTEM

The Ares projects are moving forward with design of the first stage propulsion system for the Ares I crew launch vehicle and the Ares V cargo launch vehicle. Together, the Ares I and V will provide the space launch capabilities to fulfill NASA's exploration strategy of sending humans to the Moon and beyond.

Ares First Stage

The first stage of NASA's Ares I crew launch vehicle, which will loft the Orion CEV into LEO, will consist of a space shuttle-derived five-segment reusable solid rocket booster

(RSRB) and an upper stage powered by a J–2X engine. A pair of five-segment reusable solid rocket motors (RSRMs), in conjunction with a liquid-fuel core stage, will also be used on the Ares V cargo launch vehicle.

Performance requirements, basic architecture, and obsolescence issues were all factors in determining the Ares first stage design and configuration. In an effort to minimize both schedule and design risks, the derived first stage incorporates key hardware components and design features from the RSRB. Because of Ares I's unique inline configuration, the first stage will require entirely new forward structures and a modified systems tunnel (fig. 1). The new forward structures include a frustum, a forward skirt extension, and a forward skirt.

The frustum primarily provides the physical transition from the smaller diameter of the first stage to the larger diameter of the upper stage. During separation, the booster deceleration motors (BDMs) on the aft skirt are fired in forward motion to pull the booster away from the upper stage. Shortly after separation, booster tumble motors on the frustum ignite to initiate tumbling in the first stage. As the first stage begins to tumble, the frustum-interstage assembly is jettisoned in a secondary separation at a separation plane near the aft end of the frustum. The centrifugal force will be sufficiently vigorous to propel the frustum-interstage assembly safely away from the first stage. The frustum and interstage are not reused.

The forward skirt extension houses the main parachute support system and the main parachutes for first stage recovery. The forward skirt of the Ares I houses the first stage avionics controls for ignition, thrust, and separation commands. The new systems tunnel is a hollow protuberance that runs the length of the first stage exterior and houses the avionics, control cabling, and flight termination system (FTS). The FTS is used to terminate vehicle thrust and break up the rocket if the vehicle steers off its intended trajectory.

Increasing Performance and Updating Hardware.—Several first stage motor design modifications were required to meet Ares I's operational requisites. The Constellation Program requires a first stage capable of a specific thrust trace (fig. 2). This led to a redesign of the propellant grain geometry and formulation as well as design modifications to the motor nozzle. However, shuttle-legacy hardware will still be used for the exterior motor casings of the five-segment motor.

The number of "fins" at the forward end of the propellant chamber was increased, and the length of the "fin slots" between them was reduced. This modification expands the initial aggregate burn surface area. The second and fourth middle segments will include chamfers (bevels) and inhibitors to ensure the propellant burns evenly from the axis to the outer casing, and will mitigate the risk of the bore choking.

A nozzle throat area increase was essential to maintain the casing's current maximum expected operating pressure. This change delivers additional thrust consistent with the motor performance requirements.

A different burn rate will be utilized, demanding a slightly different propellant formulation. The new variant of polybutadiene acrylonitrile incorporates a different, coarser grade of iron oxide. Because the new iron oxide formulation has a lower specific surface area, it is not as good a catalyst and consequently provides the capacity to lower the burn rate slightly,

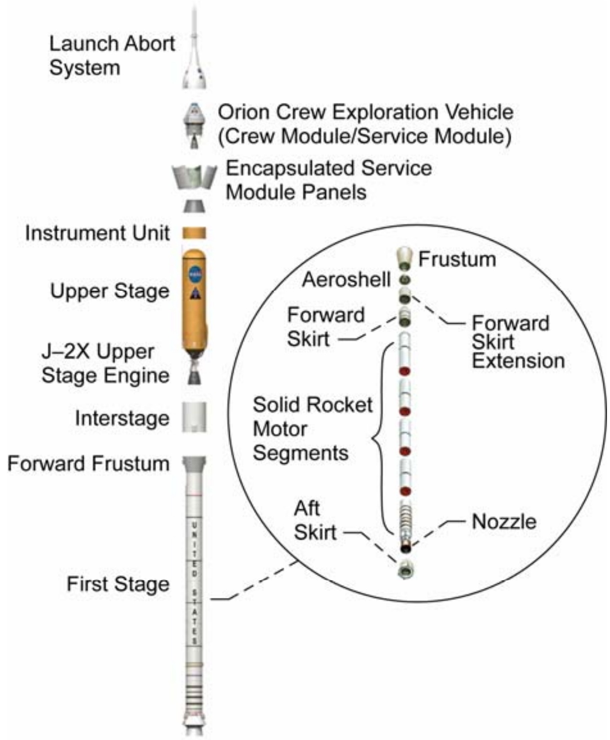

Figure 1.—Ares I's inline configuration improves crew safety and aerodynamics. (inset) The RSRM includes space shuttle legacy hardware and new hardware for the Ares launch vehicles.

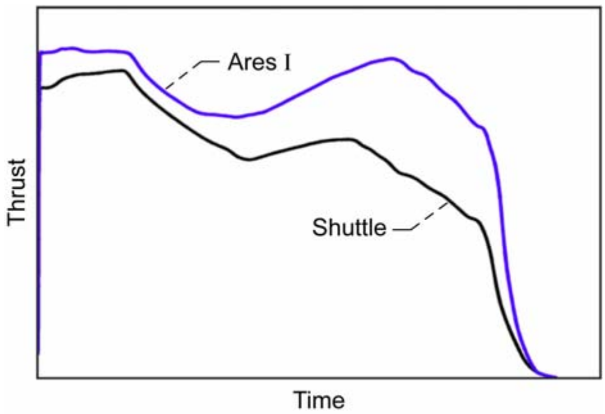

Figure 2.—Thrust trace comparison: Shuttle versus Ares I.

while maintaining an identical level of iron oxide. This modification allows for greater control of burn time and provides the ability to tailor the burn rate to achieve the desired burn time. This combination of modifications produces a significantly higher total impulse than the existing RSRM because of the additional mass flow rate and slightly longer burn time.

A new internal insulation will support the thermal protection demands of this new propellant formulation. It will be lighter and more environmentally friendly (asbestos-free) than the current insulation used on the shuttle's RSRBs. In addition, this motor will incorporate new lower-temperature materials in the O-rings, enabling the removal of joint heaters and simplifying launch site operations. Other requirements, including longer shelf life for the boosters, have also been incorporated into the design.

Ares I–X Flight Test.—To obtain data on controlling the narrow, elongated crew launch vehicle configuration, the Ares I–X flight will employ a combination of flight and simulation hardware. The Ares I–X flight profile will closely mimic the flight conditions the launch vehicle experiences, through the maximum dynamic pressure quotient (max. Q). Mission elapsed time for first stage burnout and upper stage separation will be closely matched (within a few seconds). The upper stage simulator and the Orion Command Module/Launch Abort System (LAS) simulator hardware will fly approximately 125 miles downrange and descend into the Atlantic Ocean. It will not be retrieved. The first stage booster will fly through a complete recovery sequence. Following recovery, the first stage hardware will be returned to the Kennedy Space Center (KSC) for analysis. The resulting data will provide performance statistics and a solid foundation of knowledge for hardware and software design decisions, and will help operation processes and products. The Ares I–X flight test will be the first flight test for the parachutes as well.

NASA has already begun manufacturing the Ares I–X forward structures. This hardware, together with the existing four-segment SRB, must be delivered to KSC for final assembly and vehicle stacking by August 2008 in order to support an April 2009 launch date for the Ares I–X test flight.

First Stage Conclusion and Technical Status.—Ares first stage design progress is robust. The system requirements review (SRR) was completed in December 2006. The Preliminary Design Review (PDR) is scheduled to be complete in June 2008. Ares I–X hardware is in the midst of fabrication. The first Ares I development motor (DM–1) is in the manufacturing process. The DM–1 static firing is slated for April 2009. The Ares I and Ares I–X first stage teams are pursuing the design and development of propulsion hardware for America's next generation of human-rated launch vehicles. The Ares I–X first stage team has already conducted subelement and component-level major design reviews and critical design reviews to ensure that the vehicle specifications meet the stage and vehicle requirements. Drogue and main parachute tests continue in 2008. Recovery system testing is well underway. The Ares I–X launch is scheduled for April 2009, and the Ares I Critical Design Review (CDR) will follow.

Ares Upper Stage Engine

The J–2X Upper Stage Engine represents the first new cryogenic booster engine development by NASA since the Space Shuttle Main Engine (SSME). The J–2X will provide upper stage propulsion for both the Ares I and the Ares V. In operation on the Ares I upper stage, the J–2X will ignite at an altitude of roughly 190,000 ft and burn for roughly 500 s to put the Orion CEV into orbit. In operation on the Ares V Earth departure stage (EDS), the J–2X will start at a similar altitude, operate for roughly 500 s, and then shut down. After a loiter time that could be as long as 5 to 14 days, it will then restart and operate for roughly 300 s to perform a translunar injection burn, followed by final engine shutdown. For both applications, the engines will operate with a primary mode thrust of 294,000 lbf (1307 kN) and a propellant mixture ratio of 5.5 for the ascent to orbit. For the EDS reignition, the J–2X will shift to a secondary mode mixture ratio of 4.5 and attain roughly 241,000 lbf (1072 kN—82 percent) thrust to accommodate load limits on the Orion/Altair lunar lander docking system.

Performance and Schedule Challenges.—The J–2X embodies the Constellation goals of heritage-based hardware, legacy knowledge, and commonality among hardware elements where feasible. The engine employs the liquid hydrogen/liquid oxygen (LH_2/LOX) gas generator cycle found in both the Apollo-era J–2 engine and the contemporary RS–68 engine. It also leverages hardware and experience from the XRS–2200 aerospike engine, SSME, and the RS–68. However, Constellation mission performance requirements demand that the J–2X operate at much higher temperatures, pressures, and flow rates than the Apollo J–2. The design now being pursued will make it one of the highest performing gas generator cycle engines ever built, almost as efficient as more complex, staged combustion engines such as the SSME. Whereas the heritage J–2 provided 230,000 lb of thrust at a specific impulse of 425 s, the J–2X will produce 294,000 lb of thrust at a specific impulse of 448 s.

An additional challenge of the J–2X is the significantly reduced engine development schedule. The J–2X development team is attempting to complete design and development 2 years earlier than the normal development timeline for a liquid rocket engine of this size. This reduced schedule is driven by the engine-need dates for Ares I to reduce the human space flight gap following shuttle retirement.

J–2X Key Components and Upgrades to Meet the Challenges.—The major components of the J–2X engine—turbo-machinery, gas generator, main injector, main combustion chamber, nozzle, nozzle extension, propellant valves, and main engine controller—are all evolutions of flight-proven designs adapted and upgraded to meet Ares and Constellation performance and safety requirements. Also, because the team working on the J–2X comprises people with recent experience in RS–68 development, it was more efficient to derive some components from the RS–68 heritage rather than the older J–2.

Based on the J–2S Mk-29 design, the turbomachinery is being modified to meet J–2X performance and current design standards. The turbopumps were upgraded to include more robust modern materials. Internal changes were made to provide for rotordynamic and structural margin when operating at the more challenging J–2X conditions.

The gas generator is based on the RS–68, scaled down to meet the J–2X needs. The RS–68 lineage was chosen over scaling up heritage J–2 gas generator because of the design team's greater familiarity with the design as well as its operational success.

Several changes are being made to the thrust chamber assembly to increase the thrust and efficiency of the engine. The main injector has been upgraded from the original J–2 design. The total number of injector elements is being increased to improve the characteristic exhaust velocity (C*) efficiency and, thereby, improving specific impulse. The hot-isostatic pressure (HIP) bonding process has been applied to the manufacturing of the main combustion chamber, based on RS–68-demonstrated technology. The tube-wall regeneratively cooled nozzle is based on a long history of the RS–27 engine's success on the Delta II and III vehicles. A new feature of the J–2X engine is the use of a turbine exhaust gas manifold that will supersonically inject the gas into the nozzle extension to provide cooling to the nozzle extension and increased performance. Although the original J–2 engine also dumped the turbine exhaust into the nozzle, its primary purpose was for cooling and not for performance.

Because the J–2X is an altitude start engine, a large area ratio nozzle extension (92:1), can be employed to provide additional thrust and to meet an aggressive 448-s specific impulse requirement. After evaluating both metal and composite nozzle extensions to provide the necessary I_{sp}, designers selected the composite option, based on RL10 B–2 design and experience, which offers the best balance of thermal margin and technical maturity. Its main advantage is thermal margin over metallic materials considered. When assembled, a two-cone configuration, selected to reduce manufacturing risk, will be the largest shell nozzle extension for a liquid rocket engine created to date. A backup metal design is being studied for risk mitigation.

The engine features open-loop, pneumatic control of ball-sector valves traceable to the XRS–2200 and RS–68. This control is effected through an electronic engine controller—directly based on the RS–68 design and software architecture—and a pneumatic control assembly to provide power to the valve actuators.

Testing and Progress to Date.—A key part of the accelerated J–2X development plan is early risk reduction testing at the component and subsystem level to guide the overall development of the engine. Tests to characterize or recharacterize heritage components are being followed by tests of increasingly flightlike components and subsystems. System-level testing will verify design of the integrated system under flightlike conditions in an altitude test facility.

Component Testing.—Testing in 2006 focused on injector and valve hardware. In 2007, the key turbomachinery development program included water flow testing of existing pump inlets, a turbine air flow test, and water flow tests of the inducers, which provided insights important to planning the later powerpack series. In 2008, workhorse gas generator testing will be conducted to influence the flight design.

In addition, heritage turbopumps and a gas generator are being used in powerpack testing to characterize engine conditions that will influence and anchor turbomachinery designs at J–2X operating conditions.

The development program actually includes two powerpack series: early testing with heritage J–2 components, followed by a second series with flight components. The first series, Powerpack 1–A, began in late 2007 on the A–1 test stand at Stennis Space Center (SSC). The heritage-hardware test article is shown in figure 3 during installation.

Among the goals for this series are preparing and testing the facility, obtaining inducer flow environments and pump performance, and evaluating turbomachinery performance coupled to Ares I upper stage-like feedlines at J–2X operating conditions where practical. The second powerpack series, using flight components, is planned for early 2010.

Engine Systems Testing.—The J–2X development plan includes more than 200 engine hot-fire tests of 9 engines for development, certification, and test flight. The majority of these tests will be conducted on a trio of test stands at SSC. Sea-level testing, used to prove basic engine system design and performance, will be on the A–1 and A–2 stands. These stands are being modified to support J–2X testing after being used for SSME testing since the 1970s. Because the J–2X is an altitude engine, however, it is necessary to test in simulated altitude conditions. This requirement, along with additional requirements of the large composite nozzle extension, drove the need for a new altitude test capability. This new facility is currently being built at SSC.

Figure 3.—Powerpack 1–A during test stand installation at SSC.

Figure 4.—Artist's concept of A–3 test stand at SSC for J–2X altitude testing.

A–3 Altitude Test Stand.—NASA-approved site work began on A–3 in spring 2007. The new stand (fig. 4) includes the ability to test the engine over its full 500-s mission duration at simulated altitudes of more than 100,000 ft. Particularly important is the ability to perform system start and shutdown without the magnitude of sea-level pressure transients that could damage the nozzle extension and regeneratively cooled metal nozzle. The existence of A–3 also permits other J–2X testing to continue in parallel on the Stennis A–1 and A–2 test stands. The stand will be completed and ready for J–2X testing in December 2010.

J–2X Summary.—The J–2X will be the workhorse upper stage engine for the next generation of space exploration. The overarching intent is to keep the engine as simple as possible while still fulfilling the challenging performance requirements. The development schedule is short and already quite active, with PDR completed August 2007 and CDR scheduled for November 2008. The first engine systems test is scheduled for spring 2010, driving toward completion of development by the end of 2012.

ARES V

The initial role of the Ares V launch vehicle will be to deliver the EDS, carrying the lunar lander, Altair, or cargo, to LEO (see fig. 5). Similar in sequence to the space shuttle, the solids are released during ascent to LEO and are designed for recovery and reuse. The core stage continues the LEO insertion and separates using booster separation motors. After the EDS jettisons from the core stage, it performs a circularization burn, placing the EDS and Altair in orbit with Orion. After Orion docks with Altair, the EDS will initiate a second burn, leaving Earth orbit and placing Altair and Orion on a course to the Moon. The EDS will separate once sufficient velocity is attained, allowing the Altair descent engines to later perform lunar orbit capture.

The Ares V first stage propulsion system consists of an LH_2/LOX core stage powered by five commercial RS–68 engines. The engines are upgraded versions of those currently used for the Delta-IV launch vehicle. Adjacent to the core stage are two steel-cased polybutadiene acrylonitride (PBAN) five-segment reusable solid rocket boosters, based on the Ares I first stage design and shuttle heritage that will allow for the use of common infrastructure for manufacturing and processing. On top of the core stage is the EDS, powered by a single J–2X upper stage engine that similarly leverages investments in the Ares I upper stage engine. The EDS uses aluminum-lithium (Al-Li) tanks and, like the core stage, is 33 ft in diameter.

Test and analyses will verify that the design of the Ares V will meet the lunar architecture performance requirements that are being refined concurrently with the EDS and Altair designs. Current studies are investigating changes to the propulsion systems for potential performance augmentation. The trade space alternatives include the use of composite casings for the solids, core stage length, number of core RS–68 engines, and engine I_{sp}.

Although the Ares V heavy-lift vehicle will initially be employed for human exploration, application of the capability of Ares V, with its heavy payload capability of 131,800 kg to LEO and large fairing volume, enables a new range of science missions. The capability allows science mission designers to fully optimize or take full advantage of the following characteristics: mass delivered, trip time, delta-V, unique spacecraft architectures, complexity, and other mission-enabling variables to achieve new classes of missions. For example, preliminary performance assessments indicate that Ares V could deliver 5 times the payload to Mars as compared to the most capable U.S. launch vehicle available today, the Delta-IV Heavy [1]. Alternatively, the capability could also be used to achieve high launch energies that would reduce total mission time for outer solar system missions.

ORION PROPULSION SYSTEMS AND DEVELOPMENT STATUS

Launch Abort System

The Launch Abort System (LAS) assures that the Orion crew can be removed from a hazardous condition either on the pad or during ascent. The engine is configured in the tractor mode implementing reverse flow nozzles. The LAS uses solid propellant to perform its function in three separate motors. Each of these motors will require a new development and qualification program. The escape motor is the engine used to provide the primary impulse to move the crew away from the Ares vehicle. The escape motor moves the crew away from the vehicle imparting approximately 15g on the crew. The launch abort vehicle (LAV) flight is defined as the crew module (CM) and the LAS. During the powered and unpowered portion of the LAV flight, the second of the three motors operates to provide control of the LAV (fig. 6). This motor is the attitude control motor (ACM), which produces 2500 lb of thrust (per nozzle) nominal and operates until the CM is in a stable mode of flight. The final LAS motor is the jettison motor used in either abort or nominal scenarios. The motor provides the impulse to separate the LAS from the CM. Each of the LAS motors has undergone extensive development to date; the escape to date; the escape motor has undergone a successful reverse flow proof of concept testing. The first full-scale firing of the jettison motor occurred in March 2008 (fig. 7). The ACM motor will be undergoing development testing in June 2008. The first test of the integrated system will be used to demonstrate a pad abort condition. The test, designated PA–1, is scheduled for late 2008. The LAS will then undergo an additional pad abort test as well as several abort flight tests at a variety of conditions including aborts at maximum dynamic pressure and transonic condition.

Figure 5.—Ares V vehicle elements.

Crew Module

The CM reaction control system (RCS) controls the CM when separate from the service module (SM) and separate from the LAS in nominal and abort modes (fig. 8). The system provides orientation and directional control for the CM during landings using roll, pitch, and yaw control jets in a monopropellant hydrazine system. The system has primary and backup engines in each control axis, as well as a manual control function for emergency cases. The CM RCS will serve as the primary means for the CM to be retargeted after separation from the SM by maneuvering the vehicle in a technique called skip reentry. Because of orbital mechanics, the point of return for the CM is set when the return burn comes from the Moon or when the deorbit burn is performed by the SM. However, the

CM

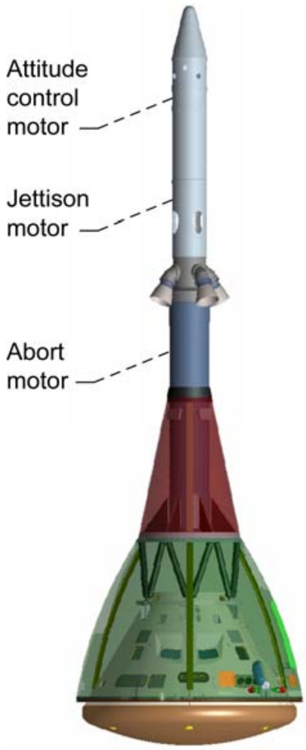

Figure 6.—Launch abort system production configuration.

Figure 7.—Orion launch abort system jettison motor test.

RCS can be burned to skip the CM through the atmosphere and target a specific area, which thus increases the cross range of the vehicle. The CM RCS is also used to control the vehicle during the reentry phase using pulses to keep the vehicle orientation. After the parachutes are deployed, the CM RCS will be used to maintain the vehicle in the proper orientation under the chutes in off-nominal cases. The system utilizes monopropellant hydrazine (N_2H_4), flowed over a catalyst bed, to provide the required impulse during all mission phases. The CM RCS utilizes heritage thrusters (Aerojet MR–104) to perform its mission. Some modification is required for the engines to operate in a “scarfed mode,” but the remaining components use the significant monopropellant component history. Although the CM has reusability requirements, the CM RCS is not planned to be reused under any circumstances and will be completely replaced after each mission. The CM RCS is not used during orbit as the SM propulsion system is the primary means of control during all ISS and lunar in-orbit phases.

Service Module

The SM propulsion system is used during ascent and in-orbit phases of flight. In a nominal ISS mission, the SM propulsion system is used to raise the orbit from which the Orion vehicle is delivered by Ares and maneuver to and from the ISS. In a nominal lunar mission, the SM propulsion system

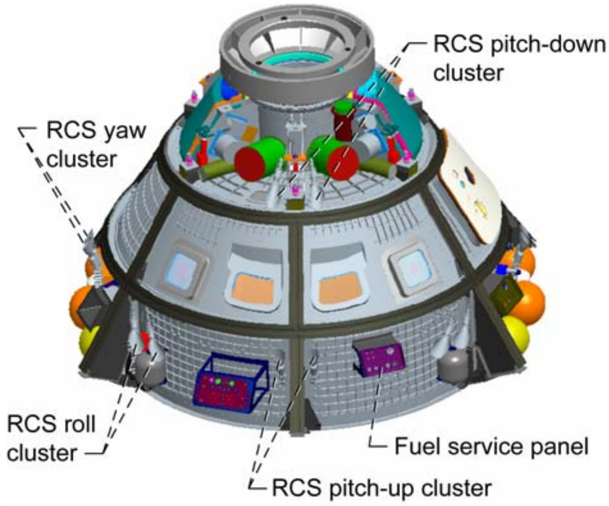

Figure 8.—Orion crew module.

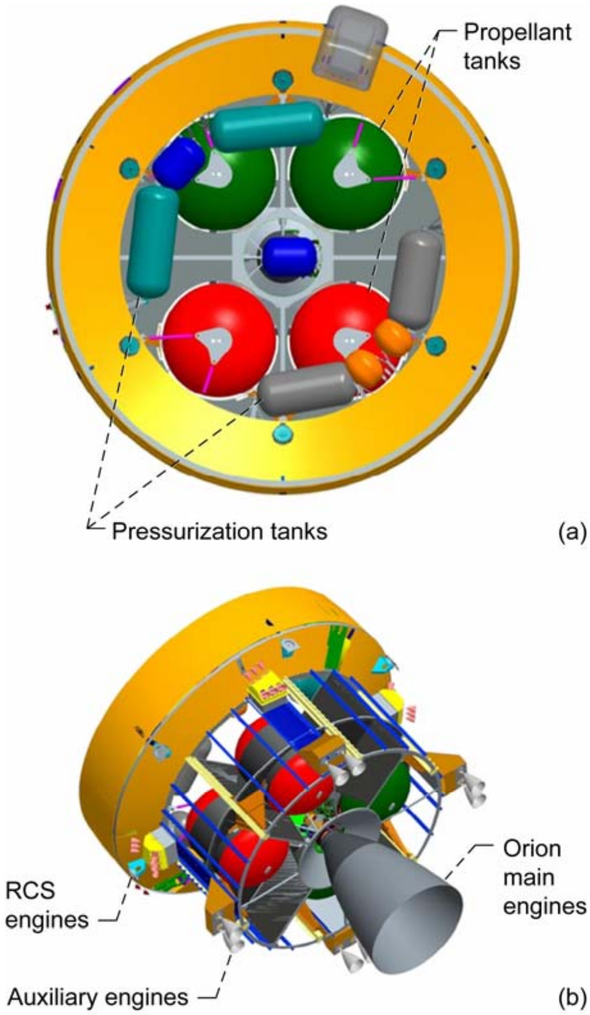

Figure 9.—Orion propulsion systems.

is used to raise the orbit from which the Orion vehicle is delivered by Ares, provide RCS around the Moon, and to perform the trans-Earth injection (TEI) maneuver to bring the Orion vehicle back from the Moon. In an ISS or lunar abort scenario, the SM propulsion system is used to prevent the Orion capsule from landing in the downrange abort exclusion zone. To meet these mission scenarios, the SM propulsion system employs a common tankage and feed system for the entire system. The system utilizes hypergolic propellants to provide the required impulse during all mission phases. The oxidizer is nitrogen tetroxide (NTO), and the fuel is monomethyl hydrazine (MMH). For an ISS mission, the SM carries approximately 8500 lb of propellant and for the lunar mission; the SM carries approximately 18,000 lb of propellant. The Orion main engine (OME) is a shuttle-derivative engine, and its primary function is to provide the majority of thrust in an abort case and to perform the TEI burn (fig. 9). The main engine thrust size is determined by the thrust-weight of the Orion vehicle in a North Atlantic abort. The auxiliary engines (Aerojet R–4D) are a derivative of the original Apollo RCS engines and are used as backup for the main engine during any failure of the OME during or prior to the TEI. The auxiliary engines are also used to augment the OME during an abort maneuver to increase the thrust-weight ratio. The SM RCS engines (Aerojet R–1E) are used for fine control of the Orion vehicle during docking with ISS, Altair, and low lunar orbit maneuvers. The OME is currently in design and development. Early hot-fire testing of a development engine is planned for 2009. Although based on space shuttle orbital maneuvering system heritage, the changes made to incorporate the additional performance as well as changes in manufacturing techniques will require a new qualification program. The auxiliary engines are in testing to validate operational parameters at a higher mixture ratio. The RCS engines are undergoing final packaging studies, and additional qualification testing is anticipated to be centered on higher mechanical vibration levels.

ALTAIR TECHNOLOGY DEVELOPMENT

NASA is conducting studies to determine designs for Altair while concurrently performing technology development for critical propulsion systems. The approach includes government and industry teams, and maintains a minimally functional design that serves as a reference for capability, cost, and risk studies. Altair has three unique design reference missions: lunar sortie crew, lunar outpost crew, and lunar cargo. The goal is to have a single design that supports these missions, including descent and ascent propulsion modules. Returning to the lunar surface for extended durations requires increased payloads and corresponding increases in ascent and descent thrust and impulse capabilities, compared to the Apollo Program. Propulsion trades include descent module engine characteristics, the number of descent module engines, tank pressurization concepts, ascent module engine and propellant selection, RCS propellant selection, and fluid management.

The current Altair reference identifies a single 82,857-N liquid LOX/LH_2 main engine capable of throttling and restart, four LOX tanks, four LH_2 tanks, four He tanks, and sixteen 445-N nitrogen-tetroxide/monomethylhydrazine (NTO/MMH) RCS thrusters. This main engine is used in the Constellation architecture for both lunar orbit insertion and descent to the surface of the Moon. For the ascent module propulsion system, the reference system consists of a single 24,475-N pressure-fed main engine, sixteen 445-N (NTO/MMH) RCS engines in four quadrant packs, two MMH tanks, two NTO tanks, and four He tanks (fig. 10(a)). The reference descent module RCS identi-

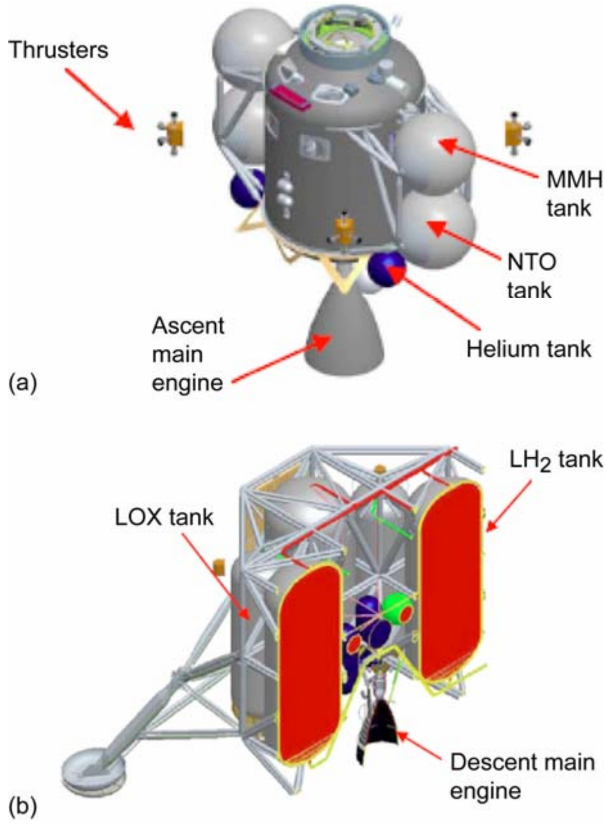

Figure 10.—Altair. (a) Ascent module propulsion. (b) Descent module propulsion.

fies NTO/MMH as the baseline, but NASA continues to consider both LOX/ethanol and LOX/LH_2 (fig. 10(b)). The main engine is a fixed thrust vector engine designed for restart. Due to the potential performance gains that the ascent module could achieve with liquid oxygen and liquid methane (LOX/CH_4), a LOX/CH_4 propellant trade is made for each major ascent module trade study redesign. NASA's Propulsion and Cryogenic Advanced Development Project (PCAD) is also developing propulsion system technologies for alternative nontoxic or "green" propellants that can be used to satisfy the lunar mission requirements.

Parallel to the Altair reference design effort, the PCAD Project Team and industrial partners are focusing technology development efforts to advance high-performance green propulsion systems for potential use on Altair. PCAD has a series of tasks and contracts to conduct risk reduction activities, demonstrating that cryogenic green propellants can be a feasible option for the lunar lander propulsion system. The focus is currently on LOX/CH_4 for ascent main engine and RCS and LOX/LH_2 for descent main engine. Implementation of green propellants in high-performance propulsion systems offers NASA an opportunity to advance beyond the hypergolic propellant option, while also providing a path for in situ resource utilization (ISRU) for propellant resupply. Additionally, using cryogenic propellant can also enhance the safety of vehicle systems and ground operations on the Earth and Moon. The following activities are structured to obtain key performance data using components and critical subsystems to reduce the risk associated with a green propulsion system.

For RCS engines, the objective is to reduce LOX/CH_4 ignition risk and demonstrate and validate performance levels for the Altair descent and ascent stages. Specific technologies include spark exciter development, spark plug durability improvements, ignition operating envelope investigations, and engine performance assessments. Key variables in the RCS task include spark energy, spark rate, propellant inlet temperatures, engine mixture ratio, and ambient operating environment. Activities examine technologies for both steady-state and pulse operation. The activity includes a combination of experimental and analytical tasks.

Contract activities are centered on Aerojet and Northrop Grumman Space Technology 445-N (100-lbf) LOX/CH_4 thrusters. Hot-fire testing will be conducted at sea level and vacuum conditions to determine the ignition and performance characteristics for both contracts. The goal will be to advance the technology readiness level (TRL) through hot-fire demonstration of key performance parameters such as electrical pulse width and minimum impulse bit, propellant thermal isolation, quiescent mode operation, and vacuum-specific impulse. Wide operating range testing will be performed using a single RCS thruster, utilizing propellant temperature conditioning in the Research Combustion Lab Cell 23 and Altitude Combustion Stand at Glenn Research Center (GRC) to obtain operational experience and ignition reliability. Integrated testing will be performed using multiple RCS thrusters coupled with a feed system in the Auxiliary Propulsion System Test Bed at White Sands Test Facility to obtain operational experience, ignition reliability, quantify thruster and main engine interactions, and develop performance data.

Although the design reference identifies hypergolic propellants for the main ascent engine, NASA has developed ascent module designs for both systems. The goal is to advance technologies that can be used for a future development of a LOX/CH_4 prototype ascent main engine. Activities focus on demonstrating reliable engine ignition at relevant operating conditions and environments, assessing and characterizing high-performance engine concepts and components, and then verifying operational performance through hot-fire tests at relevant high-altitude conditions. Performance demonstration and characterization of a thrust chamber assembly include igniter, injector, thrust chamber, and nozzle extension, and includes hot-fire testing to verify reliable engine ignition, determine engine combustion efficiency, and vacuum thrust performance. Key test data will be used to anchor the tools to allow for higher fidelity models that are used for system models for vehicle sizing. These activities will be performed by NASA and in contract with Aerojet. The current focus will be on a pressure-fed system with a nominal inlet pressure of 2.24 MPa (325 psia) with variable propellant inlet temperatures and will have fixed thrust with multiple restart capability. Target performance level is a thrust of 24,475 N (5500 lbf) and 355-s vacuum-specific impulse with a continuous burn time of 450 s with reduced performance for an additional burn of 100 s. Recent NASA Marshall Space Flight Center (MSFC) LOX/CH_4 injector tests have demonstrated adequate performance to achieve the desired LOX/CH_4 engine performance levels. As a result, NASA has confidence that a technology demonstrator engine will provide high performance for a representative mission duty cycle. System interactions among the feed system, thruster, and main engine will also be characterized over a simulated mission profile. Additionally a propulsion system testbed will be used to evaluate feed system thermodynamic vent system (TVS) technology and compare the feed system test data to earlier test series.

For the Altair descent engine, the goal is to demonstrate a LOX/LH_2 pump-fed rocket engine with restart capability and throttling over a range of 3:1 to 10:1 for deceleration and landing maneuvers. Technology is focused on propulsion applications using candidate closed cycle expander engine technology. NASA has provided technology funding to two contractors, Pratt & Whitney Rocketdyne (PWR) and Northrop Grumman Space Technology, to mature this engine technology, demonstrate deep throttling capability and raise the TRL. Under PWR's Common Extensible Cryogenic Engine (CECE) contract, an RL10 has been modified and successfully demonstrated to 11.4:1 throttling. Planned NASA/contractor descent main engine work over the next 2 years includes investments in an operable high-performance and deep throttling CECE and pintle variable area injector technology. In addition, NASA MSFC has also initiated a project to build a pump-fed 40,050-N (9000-lbf) thrust LOX/LH_2 testbed to retire risks should the Altair descent main engine configuration change to a multiple engine design.

SCIENCE/IN-SPACE PROPULSION

In order to maximize science returns, spacecraft propulsion investments must also focus on performance, reliability, and cost. NASA continues to pursue advances in both chemical and electric propulsion systems. Recent international successes with four deep space electric propulsion missions using electric propulsion systems for primary propulsion (Deep Space 1, SMART–1, Hayabusa, and Dawn) demonstrate the overall mission utility of high specific impulse thrusters. Current NASA development activities in electric propulsion include the NASA Evolutionary Xenon Thruster (NEXT) and the High Voltage Hall Accelerator (HIVHAC). The objectives of the NEXT project are to develop electric propulsion subsystems and demonstrate an integrated ion propulsion system that will meet demanding requirements for future science missions. The NEXT project, with partners Aerojet and L3 Communications, includes flight level development of major ion propulsion subsystems: thruster, power processing unit (PPU), propellant management system (PMS), and gimbal (fig. 11). Performance characteristics include power range: 0.5 to 6.9 kW, specific impulse: >4100 s, maximum thrust: 236 mN, thruster efficiency: >70 percent, PPU efficiency: 95 percent [2].

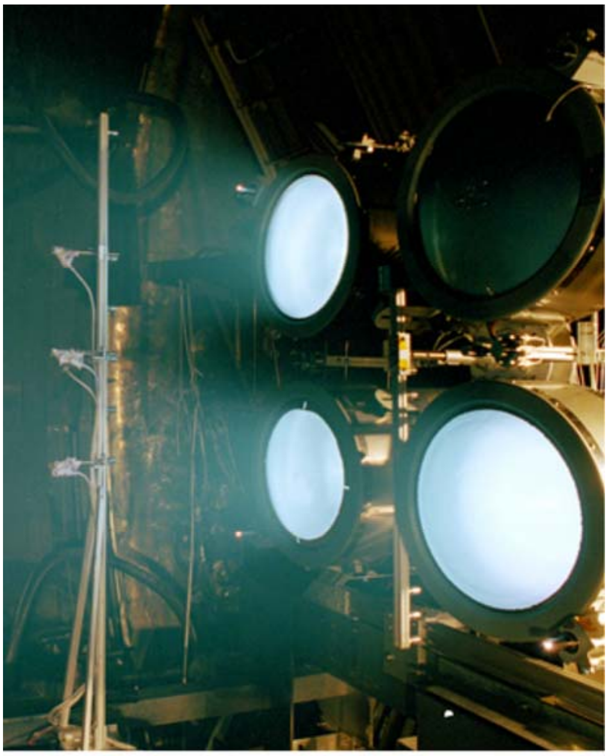

Figure 11.—NEXT multithruster array test.

Critical tests include life testing of the thruster and a single-string integrated system-level test of the major components of the propulsion system that are being conducted in facilities at NASA GRC. An engineering model (EM) thruster has successfully completed a 2000-hr wear test, and a second higher fidelity thruster is presently undergoing long-duration life testing at full power and, as of March 2008, has completed approximately 14,700 hr of operation, processing over 300 kg of xenon (Xe). The thruster has demonstrated over 12×10^6 N-s total impulse—the highest total impulse ever demonstrated by an ion thruster. The life test was operated at full power to about 267 kg Xe throughput, at which point the thruster was power throttled to simulate mission profile operations. Thruster performance tests are conducted periodically over the entire NEXT throttle table with input power ranging from 0.5 to 6.9 kW [3]. The upcoming single-string integration test will demonstrate system functions across the entire throttle range of operations including off-nominal electrical and flow conditions, control, and operation in fault modes and will prove to spacecraft manufacturers that the system is ready for flight transition.

NASA GRC is also developing Hall electric propulsion thruster technology for future cost-capped NASA science missions through the HIVHAC task. Mission analysis studies have indicated that depending on the desired characteristic, such as trip time, payload, number of thrusters or total power, and mission scenario, Hall thrusters can offer specific mission benefits relative to high specific impulse ion thrusters [4]. The objective of this activity is to increase the performance of Hall thrusters sufficiently to enable deep space science missions while retaining the cost-effective elements of commercial systems. To meet this objective, NASA and Aerojet designed and fabricated a thruster with the following capabilities: operation at input powers ranging from 300 to 3500 W, specific impulses to 2800 s, and a total propellant throughput capability of 300 kg of Xe (fig. 12). The design and fabrication of this new thruster builds on flight-qualified Hall thrusters and flight-proven electric propulsion systems. Targeted improvements include throttling to accommodate variations to power throughout the mission profile, specific impulses greater than 2500 s, and lifetimes greater than 15,000 hr [5]. Beginning in 2007, the thruster designated the NASA–100M.XL, was subjected to long-duration wear testing with thruster performance and propellant throughput being continuously monitored. This thruster had operated at full-power conditions of 700 V and 5 A with a Xe flow rate of 60 mg/s for over 3000 hr. The total propellant throughput experimentally demonstrated was in excess of 50 kg of Xe. Wear profiles also were measured during this test to provide data needed to validate numerical wear simulations. According to current projections from these simulations, the current thruster design will be able to meet the total propellant throughput objective of 300 kg of Xe. This represents an approximately tenfold increase in the throughput capability of high-voltage Hall thrusters.

Additionally, for cost-capped competitive missions such as NASA's Discovery class, NASA Jet Propulsion Laboratory (JPL) is pursuing the potential use of the commercially available Xenon Ion Propulsion System (XIPS) 25-cm thruster produced by L–3 Communications Electron Technologies, Inc. Successful adoption of these thrusters for primary propulsion for science missions requires requalification and possible modification of the system for new potential science applications. Specific differences between current geosynchronous use and science missions are being addressed and include throttling, lifetime, and operating environment [6].

Figure 12.—HIVHAC NASA–103M.XL.

NASA's In-Space Propulsion Technology Project is also advancing chemical propulsion, through a partnership with Aerojet, to develop a high-performance bipropellant engine. Current goals focus on increasing specific impulse of a pressure-fed engine to at least 330 s for NTO/MMH and at least 335 s for NTO/hydrazine propellants. Although increases in specific impulse are desired to achieve greater science payloads, additive systems penalties in mass and complexity must also be balanced at system level. Using iridium/rhenium combustion chambers, the approach is to modify existing designs to operate at higher pressures and temperatures. Changes to the current design will include injector optimization, chamber/nozzle contour optimization, reduced chamber emissivity, and increased thermal resistance between the injector and chamber [7]. Both component and engine systems tests to date have yielded potential benefits and have moved the project closer to the goal to have an engine ready for final design and qualification for science missions. The effort is expected to double the thrust and increase I_{sp} within the same volume as the state-of-the-art chemical engine for at least a 30 percent reduction in cost.

SUMMARY

The completion of the International Space Station and subsequent retirement of the space shuttle transitions Agency priorities to the development of a new human space transportation system with a return to the Moon as an initial step in a long-term journey to explore the solar system. Science missions will continue to discover and reveal exciting information for the international scientific community and will serve as precursor for future human endeavors. Although there are many challenges to overcome, the successful and sustained development of new propulsion systems will remain fundamental to progress in exploration and science missions.

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