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Comparative Assessment of Lunar Propellant Options

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In July 2006, the NASA Administrator's Office and the Constellation Program Office jointly commissioned a propellant options study to perform a comprehensive detailed analysis of the major propulsion system alternatives for the planned crew exploration vehicle and lunar surface access module before the crew exploration vehicle systems requirements review. The study team, which involved over 50 NASA engineers and analysts, performed detailed modeling of the propulsion systems and the crew exploration vehicle and lunar surface access module configurations to calculate the integrated system-level impacts on payload, mass, reliability, safety, risk, and life-cycle costs. This paper presents an overview of the tools and methods used to assess the propulsion system and architecture-level effects of employing the propellant options. It also provides the architecture-level relative performance, life-cycle costs, reliability, safety, schedule, programmatic risk, and sustainability assessments for scenarios that employ a wide variety of main and auxiliary propellant combinations. It also includes a qualitative discussion of the major risks of each scenario. As a result of the study, pressure-fed hypergolic propellants were selected as the baseline for the crew exploration vehicle service module main and auxiliary propulsion systems. Pump-fed hydrogen propellants were kept on the descent stage of the lunar surface access module, but the lunar surface access module ascent stage propellant baseline was changed to pressure-fed methane for the main and auxiliary propulsion systems, with a recommendation that the trade space be kept open while more detailed configuration studies are performed.

I. Introduction

IN JANUARY 2004, President George W. Bush announced a new Vision for Space Exploration for NASA that would return humans to the Moon by 2020 in preparation for the human exploration of Mars. In July 2005, the exploration systems architecture study (ESAS) was completed to define the systems, schedule, budgets, programs, and technologies required to return humans to the Moon; to service the International Space Station (ISS) after space shuttle retirement; and to eventually transport humans to Mars [1]. The results of the ESAS study became the initial blueprint for NASA's Constellation Program for the human exploration of the solar system.

The ESAS study recommended using a new pressure-fed methane propulsion system for the crew exploration vehicle (CEV) service module (SM) and for the ascent stage (AS) of a lunar surface access module (LSAM). Methane was selected because it has relatively high performance and lack of toxicity. In situ Mars and lunar resources could also potentially be used as propellants. Hydrogen was selected as the propellant of choice for the LSAM descent stage because of its high performance. In January 2006, NASA's Constellation Program Office decided to change the baseline propulsion systems of the CEV service module and the LSAM ascent stage to a hypergolic monomethyl hydrazine and nitrogen tetroxide (MMH/NTO) propellants to reduce the program risk and cost.

In July 2006, the NASA Administrator's Office and the Constellation Program Office jointly commissioned a propellant options study to perform a comprehensive detailed analysis of the major propulsion system alternatives for the CEV and LSAM before the CEV systems requirements review. This study, which involved over 50 NASA engineers and analysts, performed detailed modeling of the propulsion systems and the CEV and LSAM configurations to calculate the integrated architecture-level impacts on mass, reliability, safety, risk, and life-cycle costs.

This paper will present an overview of the tools and methods used to assess the propulsion system and architecture-level effects of employing various propellant options. It will include the architecture-level relative performance, life-cycle costs, reliability, safety, schedule, programmatic risk, and sustainability assessments for scenarios that employ a wide variety of main and auxiliary propellant combinations.

II. Architecture

To assess the effects of the propulsion system modifications to the CEV and LSAM, the integrated impacts on the entire architecture must be considered. For the purposes of this study, two design reference missions were used to assess the integrated impacts.

The lunar architecture developed as a part of the ESAS effort would provide the capability for up to four crew members to explore any site on the Moon (i.e., global access) for 4–7 days. These missions, referred to as lunar sorties, are analogous to the Apollo surface missions and demonstrate the capability of the architecture to land humans on the Moon, operate for a limited period on the surface, and safely return them to Earth. Figure 1 illustrates the lunar sortie crew and cargo mission. The following architecture elements are required to perform the mission: an Ares 1 crew launch vehicle (CLV), an Ares V cargo launch vehicle (CaLV) capable of delivering at least 125 t to low Earth orbit (LEO), a CEV, an LSAM, and an Earth departure stage (EDS). The technical characteristics of each of these elements are described in detail in [1]. The LSAM and EDS are predeployed in a single CaLV launch to LEO, and the CLV delivers the CEV and crew in Earth orbit, where the two vehicles initially rendezvous and dock. The EDS performs the translunar injection burn and is discarded. The LSAM then performs the lunar orbit insertion (LOI) burn for both the CEV and LSAM. The entire crew then transfers to the LSAM, undocks from the CEV, and performs a descent to the lunar surface in the LSAM. After a

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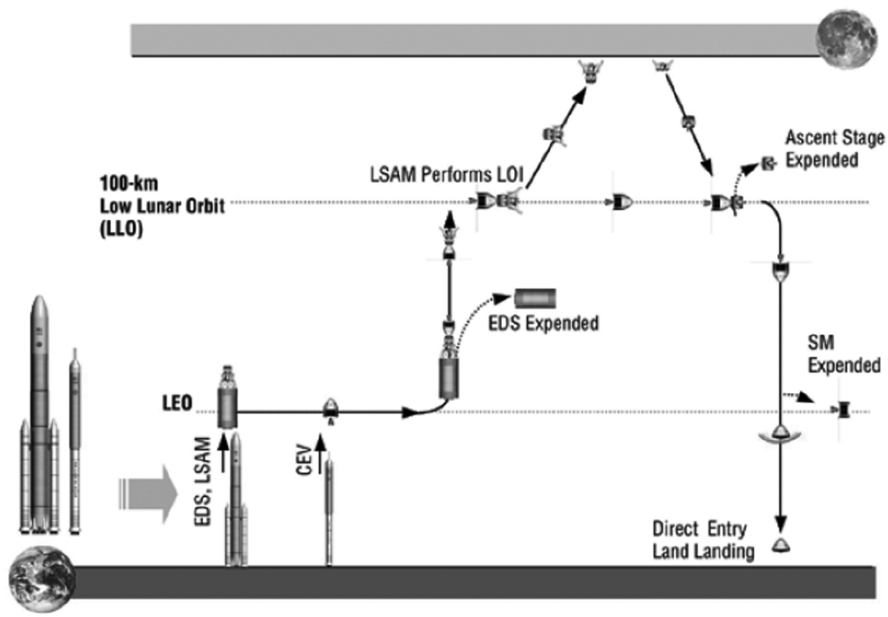


Fig. 1 Lunar design reference mission.

4–7-day surface stay, the LSAM returns the crew to lunar orbit where the LSAM and CEV dock, and the crew transfers to the CEV. The CEV then returns the crew to Earth with a direct entry and land touchdown.

The architecture will also provide the capability of taking 4–6 crew and the required cargo to the ISS for a 6-month stay and returning them safely to Earth at any time during the mission. The architecture elements that satisfy the mission consist of a CEV and an Ares 1 CLV. The technical characteristics of each of these elements are described in detail in [1]. Figure 2 illustrates the mission. The CEV, consisting of a crew module (CM) and a service module (SM), is launched by the CLV into a 56 × 296-km insertion orbit at a 51.6-deg inclination with a crew of four–six destined for a 6-month ISS expedition. The CEV performs orbit-raising burns per a pre-mission-defined rendezvous phasing profile to close on the ISS. The CEV crew conducts a standard approach to the ISS, docking to one of two available CEV-compatible docking ports. Once the crew and related cargo is transferred to the ISS, the CEV is configured to a quiescent state and assumes a "rescue vehicle" role for the duration of the crew increment. Periodic systems health checks and monitoring are performed by Mission Control throughout the increment. Upon completion of up to a 180-day mission on the ISS, the crew stows any return manifest items in the CEV crew cabin, performs a preundock health check of all entry critical systems, closes hatches and performs leak checks, and undocks from the station. The CEV departs the vicinity of the ISS and conducts a deorbit burn. After burn completion, the CEV SM is discarded, and the CEV CM is maneuvered to the proper entry interface attitude for a guided entry to the landing site. The CEV performs a nominal landing at the primary land-based landing site.

III. Evaluation Approach

A wide range of CEV and LSAM propellants were initially examined for potential system-level benefits, including pressure- and pump-fed hypergols (MMH/NTO), hydrogen, methane, ethanol, rocket propellant 2 (RP-2), methanol, and ethylene. To reduce the number of cases to be examined, these combinations were screened based on performance, maturity level, and qualitative assessments of risk, cost, reliability, and safety. Based on this initial screening, pressure- and pump-fed hydrogen and pressure-fed MMH/NTO, methane, and ethanol orbital maneuvering systems (OMS), or main propulsion systems, were selected for detailed assessment on the CEV SM and LSAM AS. Based on initial performance assessments, pump-fed hydrogen was considered to be the only reasonable choice for the LSAM descent stage, which is also responsible for performing the LOI burn and is required to provide over 3100 m/s of delta-V capability. MMH/NTO, methane, and ethanol were also considered as candidates for reaction control system (RCS) propellants for the CEV SM and LSAM AS. Figure 3 summarizes the nine major design cases that were examined by the study in detail. Engine schematics were developed, and state-of-the-art NASA engine modeling tools were used to develop the engine characteristics for each of these cases, including thrust, specific impulse, chamber pressure, mixture ratio, and area ratio. As summarized next, detailed performance characteristics were then developed for the CEV SM and LSAM AS for each of the cases, including resized internal geometry packaging, subsystem masses, tank and stage volumes, tank pressures, and thermal characteristics.

Once these performance models were developed and implemented, a variety of life-cycle scenario options were developed:

Option 1: Baseline: hypergols on the ISS CEV SM, lunar CEV SM, and LSAM AS.

Option 2a: Liquid oxygen (LO2)/methane on the ISS CEV SM, lunar CEV SM, and LSAM AS.

Option 2b: Same as option 2a, but with a backup hypergol dual development program for the CEV SM through the preliminary design review.

Option 2b+: Same as option 2a, but with hypergols on the ISS CEV SM, followed by a block upgrade to LO2/methane for lunar missions.

Option 3a: Pump-fed LO2/liquid hydrogen (LH2) on the ISS CEV SM, lunar CEV SM, and LSAM AS.

Option 3b: Same as option 3a, but with a backup hypergol dual development program for the CEV SM through the preliminary design review.

Option 3b+: Same as option 3a, but with hypergols on the ISS CEV SM, followed by a block upgrade to LO2/LH2 for lunar missions.

Option 4a: Hypergols on the ISS CEV SM and lunar CEV SM and LO2/LH2 on the LSAM AS.

Option 4b: Hypergols on the ISS CEV SM and lunar CEV SM and LO2/methane on the LSAM AS.

No pressure-fed hydrogen-fueled systems were included in the scenario analysis due to their inadequate performance as discussed next. Each of these scenarios has common OMS and RCS propellants. For each of the scenarios, figures of merit (FOMs) such as performance, life-cycle costs, safety, reliability, schedule, programmatic risk, and sustainability were assessed using the quantitative and qualitative methods discussed next. The strengths

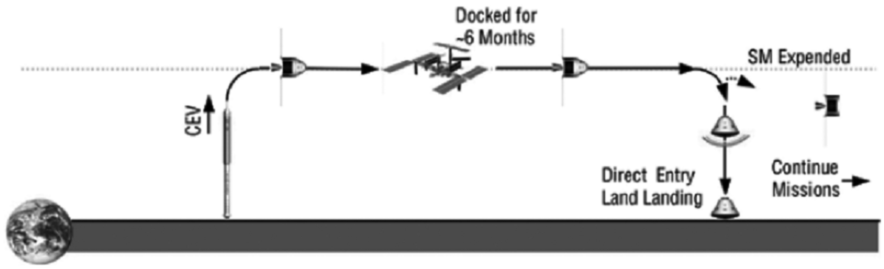


Fig. 2 ISS design reference mission.

Case	1	2	3	4	5	6	7	8	9
OMS Propellants	NTO/ MMH	LO2/ Ethanol	LO2/ Methane	LO2/ Hydrogen			LO2/ Hydrogen		
Feed Type	Pressure			Pump			Pressure		
OMS/RCS Commonality	Integrated			Separate			Separate		
RCS Propellants	NTO/ MMH	LO2/ Ethanol	LO2/ Methane	NTO/ MMH	LO2/ Ethanol	GO2/ Ethanol	NTO/ MMH	LO2/ Ethanol	GO2/ Ethanol

Fig. 3 Propellant Options for CEV SM and LSAM AS.

and weaknesses of each scenario were then documented relative to each FOM, and the results and recommendations were provided to NASA decision makers.

IV. Performance Assessment

To accurately assess the architecture-level impacts of introducing various propulsion systems into individual elements, such as the CEV SM and LSAM AS, complex high-fidelity models are required. The performance analysis for this study was greatly facilitated by the use of NASA's Exploration Architecture Model for In-Space & Earth-to-Orbit (EXAMINE) computer-aided design tool. The EXAMINE tool, illustrated in Fig. 4, uses a collection of models and algorithms from various NASA sources integrated into a common framework to enable the close-coupled sizing of reference architecture elements to the subsystem and component level. It is continuously updated to reflect the latest available data based on equal or higher-fidelity design and analysis.

The EXAMINE tool employs a "front-end" parameter driver called the architecture trade manager (ATM), which facilitates the rapid and consistent definition of architectures. The trade space manager is a driver model for assembling a reference architecture concept of operations and mission events based on customer-defined trade study variables. It is a listing of all independent variables and ranges including mission mode and requirements, architecture attributes, and segment and element design attributes. Several tools within this program were used to support the model. The optional architectural element manager uses an Excel spreadsheet with embedded logic and rules to convert input architecture attributes into launch and transfer stack manifests and critical mission operations flags. The Concept of Operations and Mission Event Tool (COMET) is a discrete event sequencing tool used for the rapid construction of operational concepts and mission events. The mission and systems requirements is an interface between ATM and the performance and sizing modeler, which uses autogenerated data from COMET to integrate all elements across the architecture and to verify that all user-supplied requirements are met.

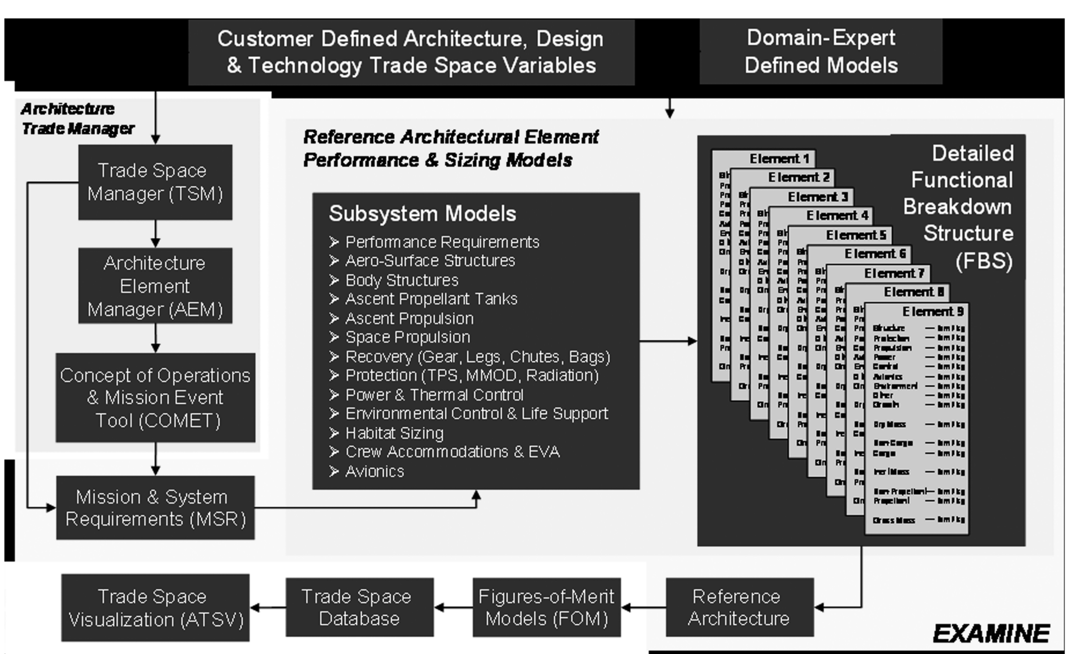


Fig. 4 EXAMINE.

The "heart" of EXAMINE is the subsystem-level performance and sizing modeler that uses historical data, physics-based models, and parametric inputs from more detailed engineering models to calculate subsystem and component-level masses. The tank and subsystem packaging are properly taken into account in the mission sizing process integral to the model. Any design variable of interest can be tracked and traded. An integrated trade space visualization tool allows rapid multidimensional visualization of the design space.

To support the EXAMINE analysis, engine schematics were developed and state-of-the-art NASA engine modeling tools were used to develop the engine characteristics for each of these cases, including thrust, specific impulse, chamber pressure, mixture ratio, and area ratio. EXAMINE was then used to develop detailed performance characteristics for the architecture elements for each of the nine cases for the aforementioned design reference missions. For each case, the tanks were resized based on the required pressures, temperatures, and volumes. The stages were repackaged and iteratively sized to accommodate the new tanks. The thermal control and insulation systems were resized to minimize propellant boil off based on a thermal analysis. Finally, detailed mass statements were generated at the subsystem and component levels.

The comparative results of these performance analyses are shown in Figs. 5 and 6. Figure 5 shows the relative gross and dry masses for the CEV SM for each of the nine different propulsion system approaches shown in Fig. 3. Figure 6 provides the same data for the LSAM AS. Based on these results, a number of observations were made:

1) The LO2-based propellant cases have higher dry masses due to the longer tanks and larger stages required to accommodate the less-dense propellants and the lower thrust-to-weight ratios of the engines. As discussed next, this will translate into higher relative development and production costs.

2) Most of the LO2-based propellant cases have lower gross masses due to the higher specific impulses. As discussed next, this translates into lower relative launch costs and increased margins.

3) The pressure-fed hydrogen-fueled systems were not competitive due to the very high relative masses of the propellant tanks. These systems were eliminated from further consideration.

4) The ethanol-fueled cases were eliminated from further study because of the performance advantages of methane and the fact that methane systems were already under development for these applications.

5) Cases 5 and 6 were eliminated from further consideration because they offered no significant advantage over case 4.

Figure 7 shows the relative gross masses of all of the architecture elements and the total mass required as payload to the launch vehicles. These results lead to a couple of important observations:

1) The mass savings from LO2-based propellants on the CEV SM translate directly kilogram to kilogram to reduced payload requirements for the Ares 1 CLV.

2) The mass savings on the LSAM AS, however, also translate to significantly higher savings on the LSAM DS and Earth departure stage (EDS) for which the LSAM AS is payload. Thus, the Ares V CALV launch requirements can be reduced by over 25 t in some cases.

V. Life-Cycle Cost Assessment

The following ground rules and assumptions were established for the life-cycle costing analysis: all costs are in FY06 $millions, the discount rate is 3%, the production unit learning rate is 95%, time-phased costs are spread using a 60% cost and 50% time beta distribution, all new systems require three orbital flight tests, and costs are estimated for only those elements affected by the change in the propulsion subsystem.

Architecture-level life-cycle costs were developed for each of the nine scenario options described in Sec. III. The relative life-cycle cost differences were calculated for each scenario using several established cost models and approaches.

The NASA/Air Force Cost Model (NAFCOM) is an automated parametric cost-estimating tool that uses historical space data from 122 NASA and Air Force space flight hardware projects to predict the development and production costs of new space programs. It uses parametric relationships to estimate the subsystem or component-level costs for any aerospace hardware including Earth-orbital spacecraft, manned spacecraft, launch vehicles, upper stages, liquid

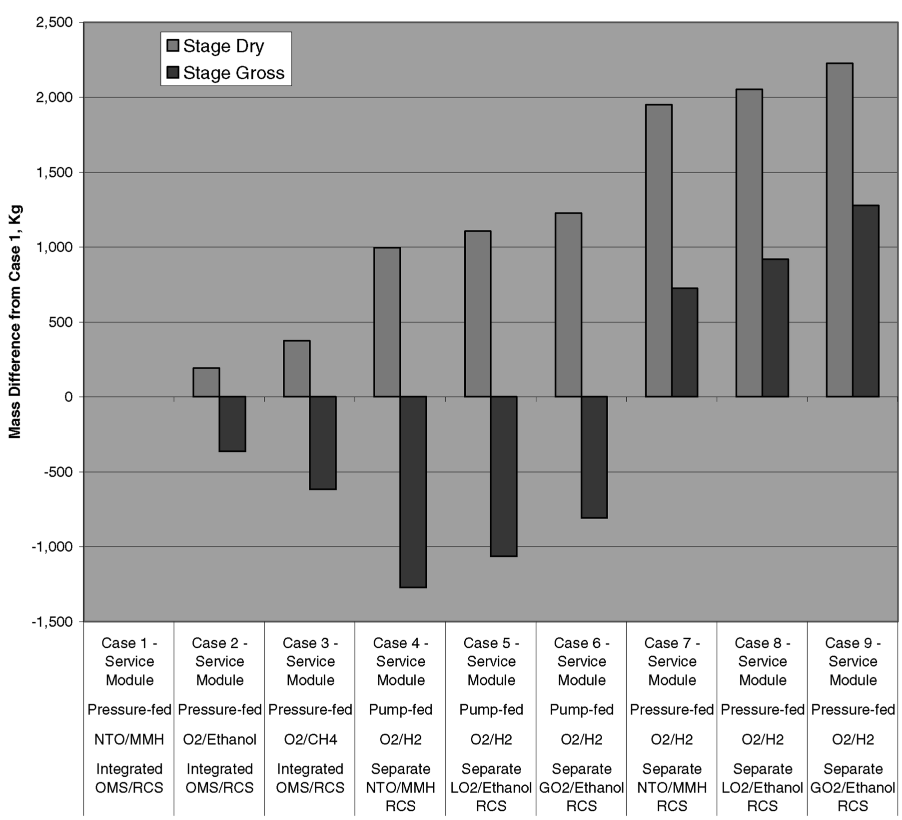


Fig. 5 CEV SM gross and dry mass variation with propulsion system type.

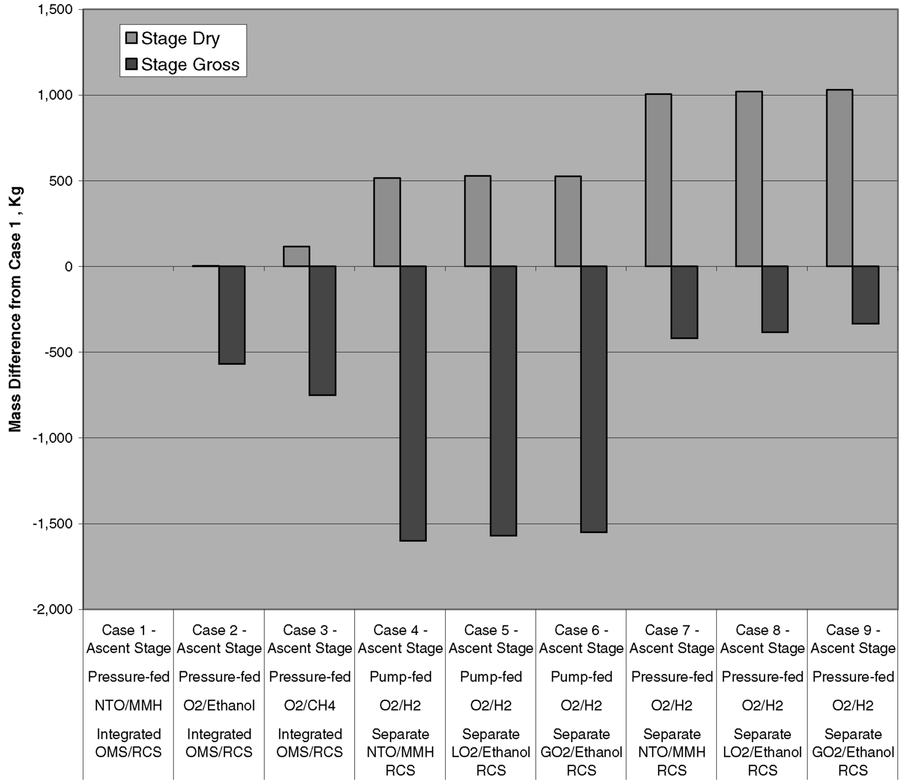


Fig. 6 LSAM AS gross and dry mass variation with propulsion system type.

rocket engines, scientific instruments, or planetary spacecraft [2]. It uses advanced multivariable cost relationships including performance, management, and sizing cost drivers and has been developed over the past 13 years with 10 different releases providing increased accuracy, data content, and functionality. The mass statements and subsystem definition information from the EXAMINE model were key inputs to the NAFCOM analysis.

Some detailed subsystem and component costs were calculated using the component cost model from Science Applications International Corporation when more applicable data models were available than those in NAFCOM. Using these models, historical development and production costs are adjusted using estimated differences in mass, complexity, percent new hardware, development degree of difficulty, and production degree of difficulty.

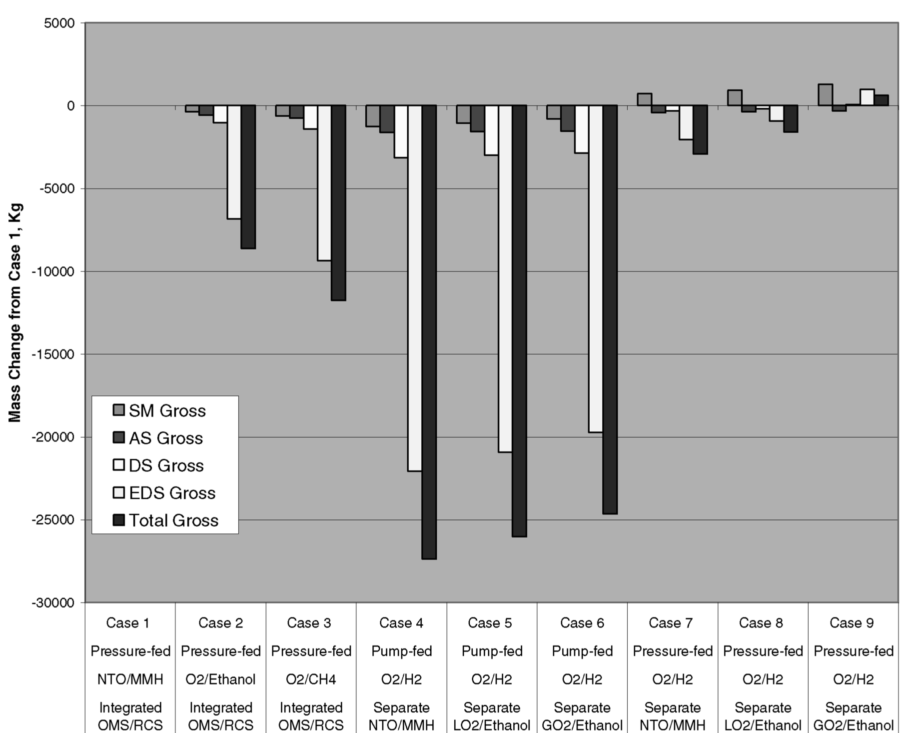


Fig. 7 Gross mass variation of each element with propulsion system type.

The technology and advanced development costs were calculated using "bottoms-up" estimates from individual technologists and technology project managers. The technology and advanced development efforts required to enable each of the scenario options were already under development or planned for development when the study began; hence, relatively mature cost and schedule estimates already existed.

The operations costs were calculated using bottoms-up estimates for each scenario option, including unique facility and support equipment development and operations costs, relative propellant infrastructure and manufacturing costs, and relative fixed and variable mission operations costs.

NASA's Architecture Cost Model (ARCOM) was used to integrate all of the costs into a time-phased life-cycle cost model for trade studies and scenario option evaluation. ARCOM is an Excel-based tool that allows rapid changes to cost assumptions and easy visualization of life-cycle cost results at the architecture level.

As an example, Fig. 8 shows the changes in costs for option 2a, which uses methane propulsion on the CEV SM and LSAM AS, relative to the baseline option 1, which uses MMH/NTO propulsion systems on each. The life-cycle costs are broken out in two different ways for analysis: 1) by architecture element (CEV SM, LSAM AS, LSAM DS, etc.), and 2) by type of cost (operations, production, development, etc.). Option 2a requires additional technology and advanced development costs in the near term because of the lack of relative maturity of the methane systems. The methane systems also cost more up front to develop because of the higher engine hardware costs and the higher stage hardware costs driven by the additional complexity and higher dry masses. The operations infrastructure costs are slightly higher initially for the methane systems due to the acquisition costs of the additional infrastructure required for cryogenic propellant storage and conditioning before launch. Once flight operations begin, the operations costs for the MMH/NTO systems are somewhat higher due to the higher propellant costs and additional labor and equipment required for the handling of the toxic propellants. The slightly lower operations costs of the methane systems in the out years, however, are dwarfed by the higher annual recurring production costs of the methane-based engines and stages due to their higher complexity and dry masses. The higher production costs of the methane CEV SM and LSAM AS are partially offset by the lower production costs of the smaller LSAM DS and EDS. The cumulative effects of all of these factors lead to higher life-cycle costs for the option 2a methane-based architecture.

A similar life-cycle cost integration and analysis effort was conducted for each of the scenario options, and the results are shown in Fig. 9. All of the options were found to be higher in life-cycle costs than the baseline option 1 hypergol case. The option 3 pump-fed hydrogen cases were found to be significantly higher in cumulative and discounted life-cycle costs due to the higher complexity and higher engine and stage masses. The option 4 cases that keep hypergols on the CEV SM and employ methane or hydrogen on the LSAM AS are much closer in life-cycle cost to the baseline case. For these options, the higher-performing propellants on the LSAM AS reduce the size and production costs of the LSAM DS and EDS, significantly offsetting the higher production costs of the LSAM AS.

A number of observations were made from the architecture-level life-cycle cost analysis of the various options:

1) The baseline option 1 scenario with hypergols has a significantly lower life-cycle cost than all of other scenarios.

a) It is dominated by lower development and technology costs.

b) The lower propellant and ground ops costs of LO2-based propellants are more than offset by stage unit production cost savings of hypergols on an annual basis.

c) The lower propellant and ground ops costs of LO2-based propellants are also offset by higher nonrecurring ops costs.

2) The pump-fed hydrogen stage development and production costs are significantly higher than the alternatives due to higher dry mass and complexity.

3) The higher LSAM ascent stage costs in options 4a and 4b are partially offset by the lower costs of the lighter LSAM descent stage and Earth departure stage.

4) The scenarios requiring a block upgrade of the SM, options 2b+ and 3b+, incur prohibitively high stage upgrade and flight-testing costs.

VI. Reliability/Safety Assessment

The architecture-level reliability and safety assessment methodology for this project used a variety of tools and methods developed over the past decade [3]. The top-down, scenario-based risk assessment approach used by this study is a complex process that incorporates many sources of information to produce a representative analysis. This approach combines modules that represent risk drivers in a transparent fashion so that design teams can easily understand risks, and analysts can quickly generate models.

The mission description provides the fundamental basis for the analysis. The abort options are identified as part of the mission design. The mission description identifies the concepts of mission operations, propulsion system type and number, vehicle systems employed, level of redundancy, etc. This information is used to create modular element reliability models tailored to key mission events (engine burns, rendezvous and docking, landing, etc.) and associated time durations. The risk assessment methodology builds on conventional, well-known techniques used to evaluate risk in complex systems, such as fault-tree and event-tree analysis. The methodology extends the accuracy and applicability of these

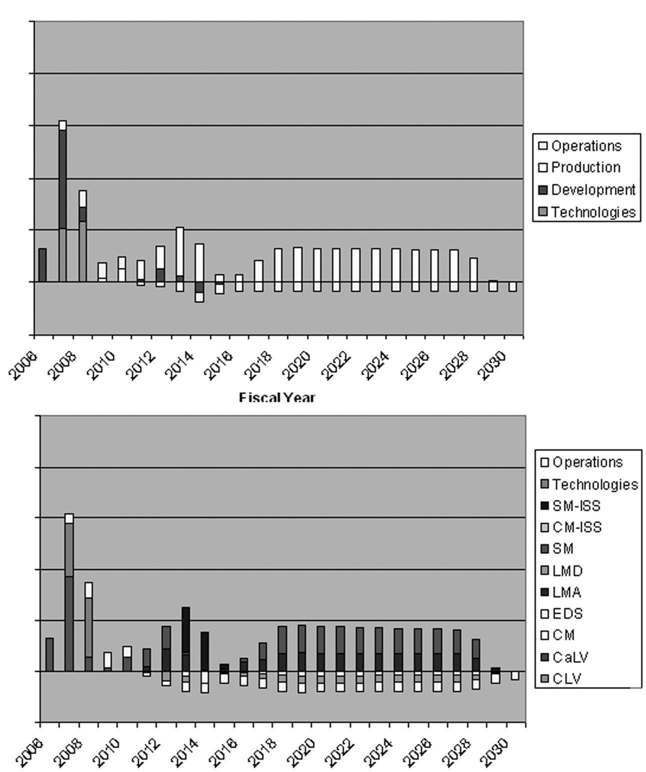


Fig. 8 Option 2a life-cycle costs.

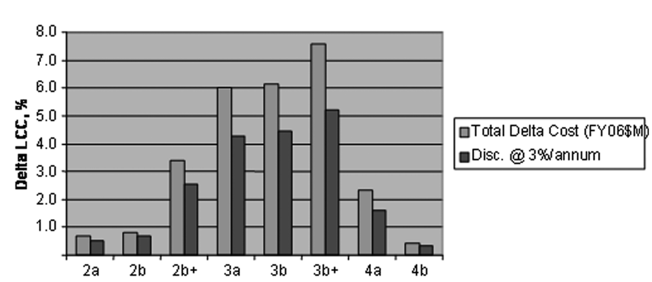


Fig. 9 Relative life-cycle cost comparison.

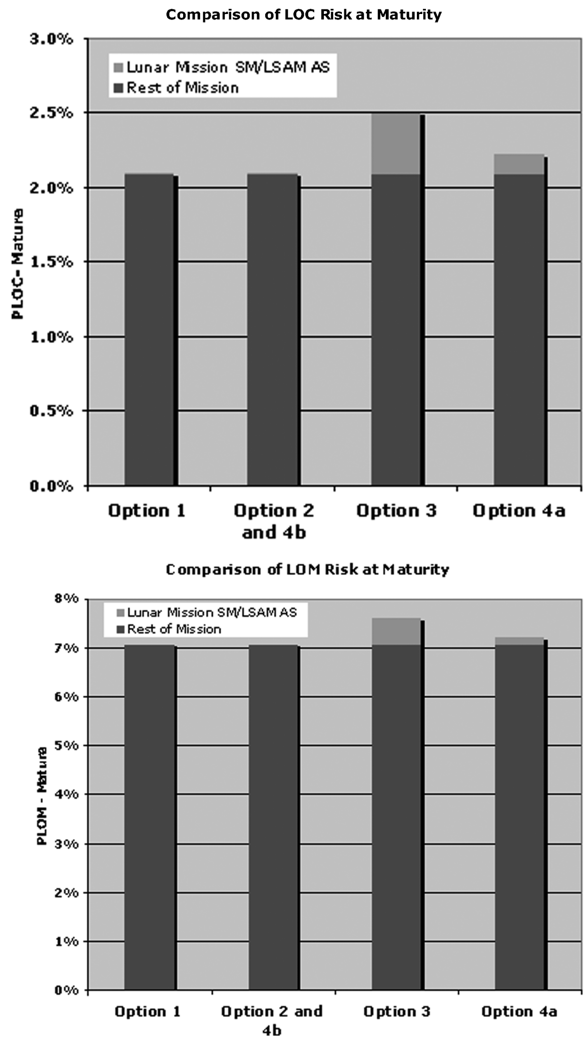


Fig. 10 Mature PLOC and PLOM values.

techniques through the use of physics-based algorithms to estimate the probability of the failure of the vehicles based on their operational characteristics. The development of these element models is enabled by an extensive database of historical component and subsystem reliability data. A maturity model based on relevant historical data is used, together with the time-phased traffic model, to predict reliability and safety improvements over time.

Probabilities of loss of mission (PLOM) and loss of crew (PLOC) were calculated for each of the scenario options. Because of the differences in reliability between the pressure- and pump-fed propulsion systems, all pressure-fed systems (hypergols and methane) use a single engine and all pump-fed systems (hydrogen) use two identical engines with full engine-out capability. Figure 10 shows the results for a typical human lunar mission, assuming the propulsion systems have all achieved a mature, steady-state level of reliability and safety. There is very little difference between the scenarios. The mature reliability and safety of the pressure-fed methane systems are almost identical to the hypergol systems. The pump-fed hydrogen systems with engine out are slightly worse.

When a historic maturity model is incorporated and the results are examined over the architecture life-cycle of the ISS and lunar missions in accordance with a projected traffic model, more dramatic differences can be seen. Figure 11 shows the cumulative effect over the architecture life cycle on the probability of loss of crew for each of the scenario options. The PLOC for the ISS missions is an order of magnitude better than that of the lunar missions due to the high reliability of the Ares 1 CLV and the capability of returning the crew to the Earth's surface relatively easily in the event of most system failures. For the first few lunar missions, the PLOC for options 4a and 4b are relatively high because of the lack of maturity of the LSAM AS propulsion system. The other options benefit from having a propulsion system that is used on the CEV SM for many flights before the first crewed lunar mission. The pump-fed hydrogen systems initially have a slightly higher reliability than the pressure-fed methane systems because of their engine-out capability; however, they historically take many more flights to reach maturity. The pressure-fed methane systems reach maturity relatively quickly. In fact, option 2a, which was the ESAS baseline, is hardly distinguishable from the baseline option 1 hypergol case because of the experience gained with the engine on the earlier CEV SM flights.

A number of observations were made based on the architecture-level reliability and safety analyses of the various options:

1) Pressure-fed hypergol systems have an extremely high initial and mature reliability due to the lack of igniters and the extensive experience base.

2) Pressure-fed methane systems have a comparably high mature reliability; however, the initial reliability is lower due to a lack of experience base.

3) Pump-fed hydrogen systems have a significantly lower mature reliability due to higher complexity. This can be partially mitigated through engine-out capability.

4) Options with no commonality between the LSAM AS and CEV SM engine have poor reliability and safety on the first few crewed lunar flights.

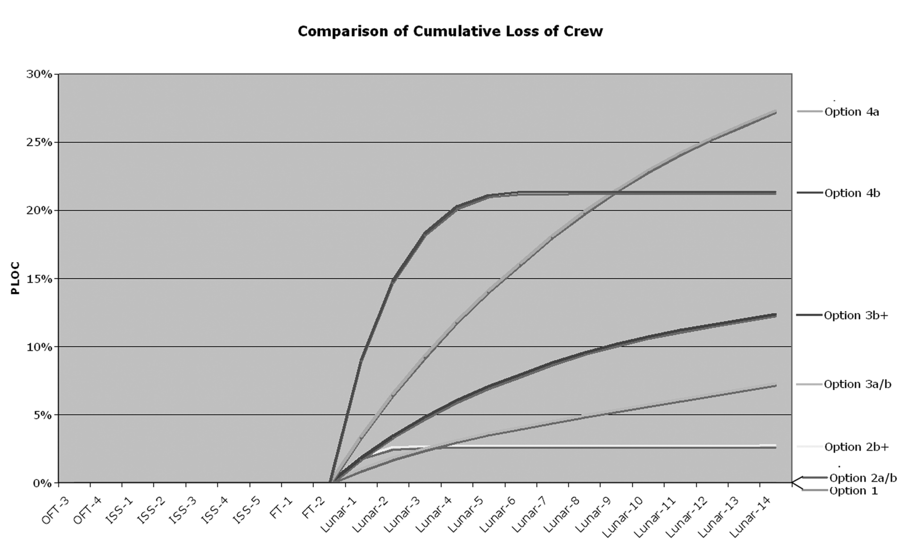


Fig. 11 Cumulative PLOC for options.

VII. Schedule Assessment

In support of this study, a bottoms-up schedule analysis was performed for the development of a pressure-fed hypergol, a pressure-fed ethanol (EtOH), a pressure-fed methane, and a pump-fed hydrogen engine for the CEV service module. The schedules, shown in Fig. 12, were developed by the same group, using consistent assumptions and methods. The following comparative observations were made as a result of this analysis:

1) Cases with hypergols have shorter schedules to the first CEV flight. Because of the large experience base, no significant advanced development is required.

2) Cases with pressure-fed methane and ethanol propellants have only slightly longer schedules, assuming advanced development can be done in parallel with the preliminary design. Some additional testing and design iteration is required.

3) Cases with pump-fed hydrogen engines have longer schedules, leading to a delay in the first CEV flight. The development schedule is longer due to additional engine complexity and delayed start.

4) Cases with oxygen-based propellants, however, have more schedule uncertainty (higher schedule risk) than scenarios with hypergols.

One important link between the cost and the schedule should be noted for the CEV development program. The schedule differences shown in Fig. 12 are driven solely by technical and programmatic discriminators. However, because of limited resources, the required budget is actually the primary CEV development schedule driver to first flight. Any scenarios with higher up-front costs will cause a delay of the CEV first flight because no additional funds are available under the fixed budget. If additional funds were available, the baseline schedule could be shortened.

VIII. Programmatic Risk Assessment

A qualitative assessment was made of the major unique risks associated with developing and operating each of the scenario options. The common risks associated with all of the options that were not discriminators were not assessed. Once the risk areas were identified, the likelihood and consequences of each risk were assessed, scored, and ranked using 5 × 5 risk matrices as shown in Fig. 13.

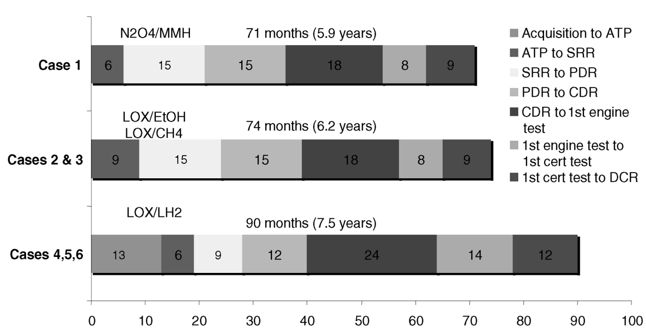
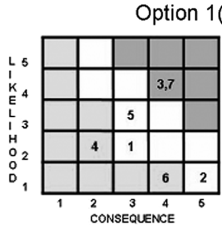


Fig. 12 Development schedules for cases.

Option 1(Hypergols) - Candidate Risks



Candidate Risk Title	L	xC SAFETY	PERF	SCHED	COST
1. Complexity of propellant heaters in development and operation	2	3	3	2	3
2. Toxicity requires complex handling procedures that put ground crew safety at risk	1	5	1	1	2
3. Formation of fuel/oxidizer reaction products (FORP)	4	1	4	1	-
4. Iron nitrate formation and deposition	2	1	2	1	-
5. Limited development and acceptance testing sites	3	-	-	3	3
6. Propellant availability/cost -sole source risk	1	-	-	-	4
7. Possibility of servicing in an offline facility	4	-	-	-	4

Fig. 13 Example risk assessment.

The major risk areas identified for option 1, which employs hypergolic propellants, include the complexity of the propellant heaters in development and operation, the toxicity that requires complex handling procedures that put ground crew safety at risk, the formation of fuel/oxidizer reaction products, the formation and deposition of iron nitrate, the limited number of development and acceptance testing sites, the environmental regulation issues, and the availability/cost of the propellant (sole source risk).

The major risk areas identified for option 2, which employs methane propellants, include the difficulty of maintaining the propellant quality to thrusters upon demand; the difficulty of providing repeatable, reliable ignition for a range of propellant conditions; the complexity of the thermodynamic vent system in development and operation; the fragility of multilayer insulation during production, ground handling, and operation; the accuracy of the modeling of heat leaks and thermal environments; the thermal conditioning and management for engine restart; the lack of historic experience base; and the launch pad complexity of the T-0 line for propellant conditioning.

The major risk areas identified for option 3, which employs hydrogen propellants, include the complexity of pump-fed hydrogen system development and operation; the complexity of the broad area cooling boil-off reduction system during development and operations; the fragility of MLI systems during production, ground handling, and operation; the accuracy of modeling of heat leaks and thermal environments; the engine start time for abort operations; the difficulty of controlling potential hydrogen and helium leakage sources; the thermal conditioning and management for engine restart; the limited historic in-space database; and the launch pad complexity of the T-0 line for propellant conditioning.

The major risk areas for options 4a and 4b are combinations of those described.

IX. Sustainability Assessment

For the purposes of this study, "sustainability" refers to the ability of technologies, systems, and operations approaches to contribute to the long-term sustained human exploration of the solar system. Introducing different propulsion systems into the lunar architecture primarily affects sustainability through the capability for in situ resource utilization (ISRU) for propellant production. The lunar and Mars regolith have a very high concentration of oxygen that can potentially be extracted for use as a propellant oxidizer. Hydrogen could potentially be directly extracted from the lunar and Mars regolith or be manufactured from in situ water ice. Methane can be manufactured from the constituents of the Martian atmosphere. Hypergolic propellants, however, are extremely difficult and potentially hazardous to manufacture from in situ planetary resources. The observations from a sustainability assessment for this study include the following:

1) Hypergolic propellants have little or no potential for lunar or Mars ISRU because of the complexity of manufacturing the propellants and the toxicity of handling the propellants.

2) Oxygen-based propellants have significant potential for lunar and Mars ISRU because the methods for extracting oxygen from the regolith, atmosphere, and water ice are well understood.

3) Hydrogen propellants also have a high potential for lunar and Mars ISRU.

4) Methane (and ethanol) have potential for Mars, and maybe lunar, ISRU because they can be extracted from Mars' atmosphere through well-understood but somewhat complex cycles and because they could also be extracted from crew carbon dioxide and plastic trash.

The cost of taking a payload to a planetary surface is extremely high. The use of in situ resources can potentially save costs to enable more sustainable operations if systems are designed to use them; however, for in situ propellant production to be economically feasible, a sufficient quantity must be used to justify the production infrastructure development, operations, and transportation costs.

FOMs	Performance	Safety/ Reliability	Schedule	Discounted Life-Cycle Costs	Programmatic Risk	Sustainability
Ordinal Ranking (Highest to Lowest)	3a	1	1, 4a, 4b ,2b+ ,3b+	1	1	3a, 3b, 3b+
	3b, 3b+	2a, 2b	2a, 2b	4b	4b	4a
	4a	4a	3a, 3b	2a	4a	2a, 2b, 2b+
	2a	2b+		2b	2b	4b
	2b, 2b+	3a, 3b		4a	2a	1
	4b	3b+		2b+	2b+	
	1	4b		3a	3b	
				3b	3a	
				3b+	3b+	

Fig. 14 Ordinal ranking of options against FOMs.

X. Integrated Architecture Assessment

For each of the scenarios, FOMs such as performance, life-cycle costs, safety, reliability, schedule, programmatic risk, and sustainability were assessed using the aforementioned quantitative and qualitative methods. The strengths and weaknesses of each scenario were then documented relative to each FOM as already summarized. The role of the study team was not to make a final decision, but rather to perform the analyses and provide all of the necessary data to support NASA decision makers. A summary of the ranking of each of those scenarios listed in Sec.III relative to the FOMs is provided in Fig. 14. The results of the FOM assessment are as follows:

1) Pressure-fed hypergolic propellants were selected as the baseline for the CEV service module main and auxiliary propulsion systems. The hypergols were found to have the highest predicted safety and reliability, supported by an extensive operational flight history. They also provided the shortest development schedule and significantly lower levels of programmatic and technical risk. In addition, the scenario options with hypergolic CEV SMs were found to have significantly lower life-cycle costs. The higher Ares 1 CLV launch masses were judged by the decision makers to be more than offset by the cost, risk, safety, and schedule advantages.

2) Block upgrades of the ISS CEV SM to other propellants for lunar missions were not found to be acceptable because of the significantly higher life-cycle costs and the negative impact on reliability and safety.

3) Pump-fed hydrogen propellants were kept on the descent stage of the LSAM because of the mission enabling performance advantages.

4) The LSAM ascent stage propellant baseline was changed to pressure-fed methane for the main and auxiliary propulsion systems, with a recommendation that the trade space be kept open while more detailed configuration studies are performed. The performance and sustainability benefits of employing methane (or hydrogen) propellants on the LSAM AS are quite large due to the impact on the LSAM DS and Earth departure stage. It is not clear if these benefits are worth the additional life-cycle costs and lower crew safety on the first few lunar flights. The LSAM configurations currently under study with smaller ascent stages will reduce this performance benefit; hence, further analysis is required.

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