Yamato

Bringing the Moon to the Earth...Again

LEXco

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2.1 Systems Engineering Process Planning

The Yamato mission to the lunar South Pole-Aitken Basin returns samples that enable dating of lunar formation and the lunar bombardment period. The design of the Yamato mission is based on a systems engineering process which takes an advanced consideration of cost and mission risk to give the mission a high probability of success.

The recent NASA Announcement of Opportunity (AO) for a South Pole Aitken-Basin (SPA) sample return mission is answered by Lunar Excavation Company's (LEXco) Yamato mission. The Yamato mission is an orbiting spacecraft coupled with dual landers that host a multitude of scientific payloads to locate and collect scientifically interesting samples. The Yamato mission is named for the Yamato meteorites found in Antarctica that were discovered to be samples from the moon liberated during early lunar impacts. The LEXco proposed Yamato mission is the future of returning samples from the moon.

NASA's Vision for Space Exploration and Solar System Exploration Roadmap calls for the continued exploration of the origins and formation of the solar system as well as the exploration of the moon by robotic and human missions. The Yamato mission in the New Frontiers mission class is a direct response to these goals. The South Pole Aitken-Basin is the deepest impact crater on the moon. The depth of the impact crater is believed to have reached the boundaries of the inner crust and mantle of the moon. Dating and finding the composition of lunar rocks from the inner crust and mantle assists in the discovery of the formation period of the solar system and the bombardment period on the Earth itself.

The goals and science of the Yamato mission can be traced directly to the major Solar System Exploration (SSE) Roadmap Questions 1 and 5:

- How did the Sun's family of planets and minor bodies originate?
- What are the hazards and resources in the Solar System environment that will affect the extension of human presence in space?

The objectives, science requirements, and mission requirements shown in Section 2.2 address these major roadmap questions. The New Frontiers mission class, formed in response to the Vision for Space Exploration, is the source of the AO and also specifically lays out the desire for a South Pole Aitken-Basin sample return mission. NASA has considered creating a permanent base on the moon and one of the most likely locations for this base would be in a SPA crater designed to shield solar radiation. All of these initiatives clearly express interest in the science and information that could be attained by a successful sample return mission to the lunar south pole region.

The Yamato mission is designed to help answer these major NASA roadmap questions while using a system engineering design effort to take into account the cost, risk, and performance values that must be carefully balanced in any major deign. The results of this design effort can be seen in Figure 1:

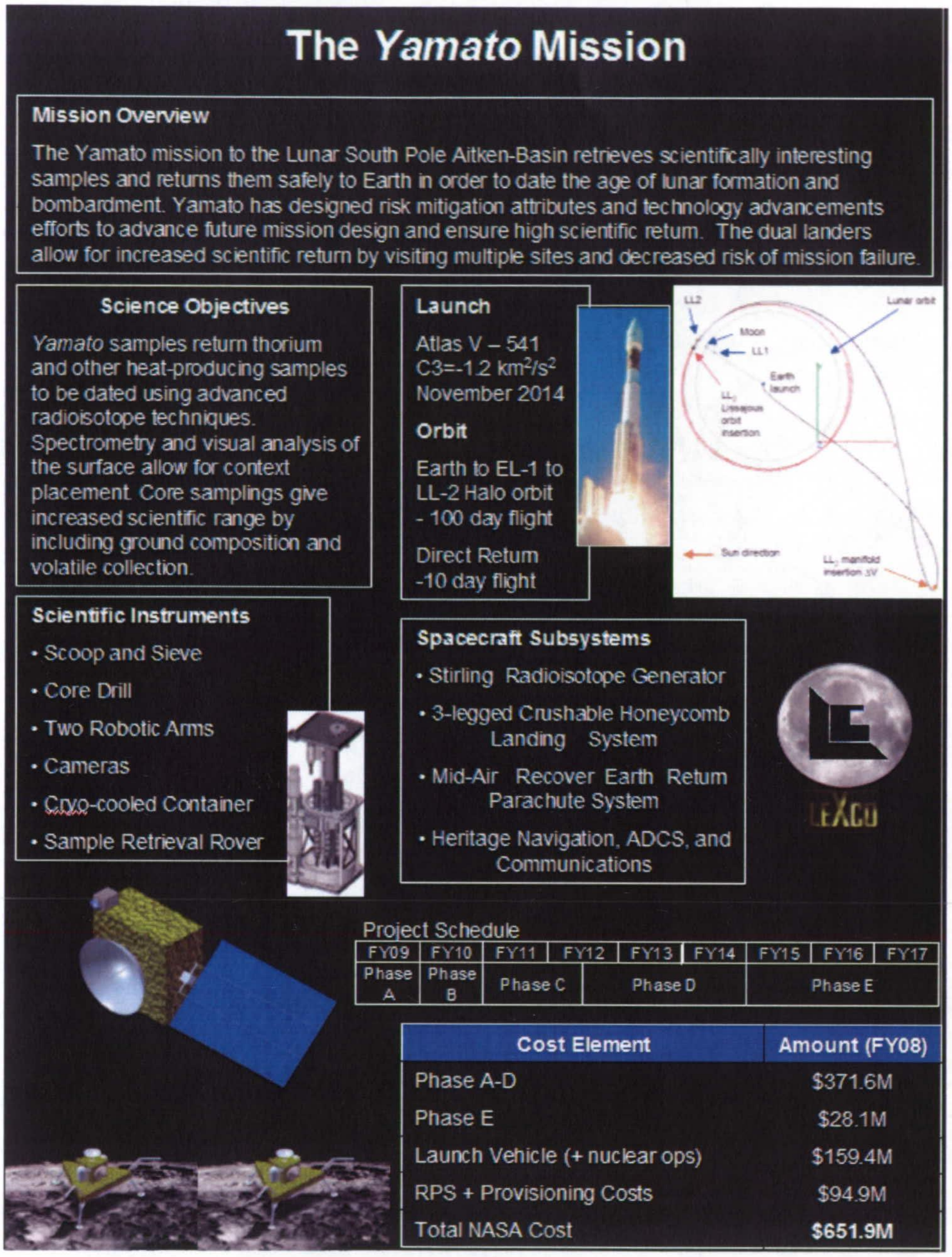


Figure 1: ***Yamato*** **mission fact sheet demonstrating high scientific return within cost and schedule margins.**

The selection and implementation of the necessary systems engineering techniques is an important part of the initiation of the design process. Planning is required to assure that the methods chosen will adequately reflect the major design decisions. The selection method used in the creation of the Yamato mission was designed to include major factors such as cost and performance while allowing subjective measure such as risk and complexity to be included in the design decision.

The systems engineering process initiated with a detailed analysis of the AO and the creation of requirement documents. Ideation of the entire mission decomposed the design into smaller functional phases. Multiple differing approaches were developed for each functional phase and synthesized into a set of possible mission concepts. The importance of the design variables was traded and the results were integrated into a major trade comparing the differing potential architectures. Major candidate architectures were then selected and further developed before a final system analysis was completed to finalize the mission architecture design. Following the architecture selection, minor trades were completed to develop the detailed system design. The details of these major system engineering steps, summarized in Figure 2, are further described in detail in the following sections.

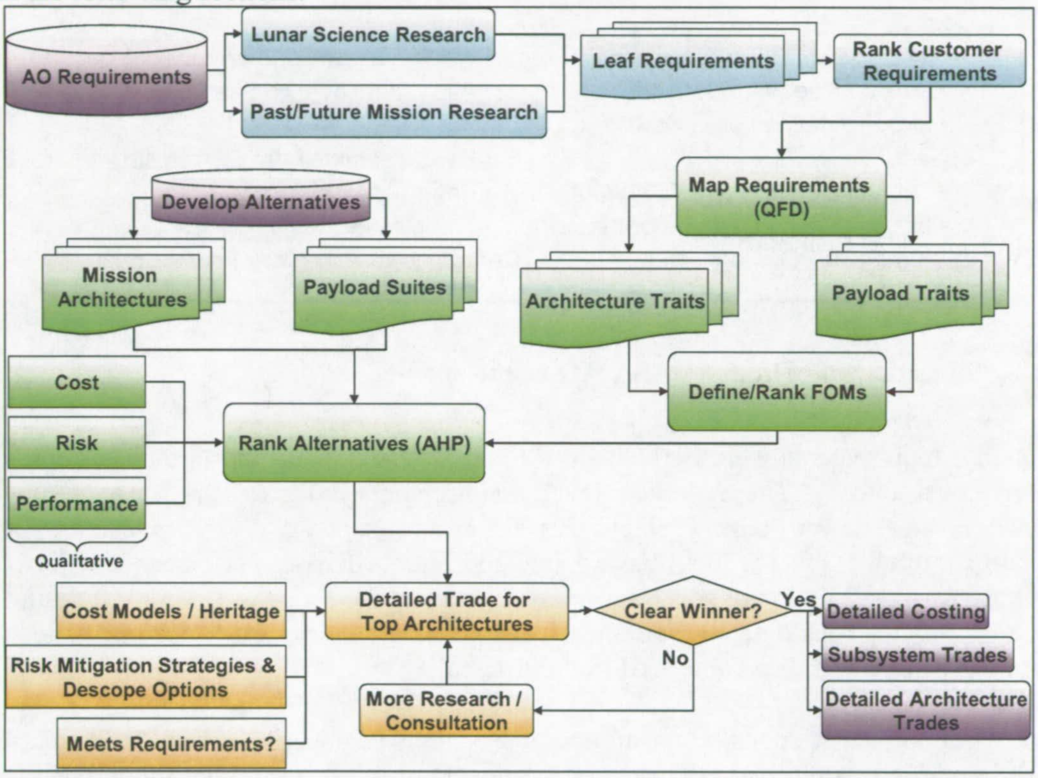


Figure 2: Summary of System Engineering Process designed for Yamato Mission selection.

2.2 Requirements Analysis and Validation

The flow down of requirements addressed by the systems engineering processes begin with the New Frontiers Program Announcement of Opportunity to propose a high quality, cost effective scientific investigation of a lunar sample return mission to and from the Moon's South Pole-Aitkin Basin. In doing so, this investigation adopts the programs objectives to:

- Pursue innovative ways of doing business with industry, university, and government partners.
- Encourage the use of advanced technologies to reduce mission cost and achieve performance enhancements.
- Contribute to improvement of science education and public understanding of science.

These expectations are further developed with verifiable customer requirements that drive the design of the mission, specifically the requirements on cost and schedule. The mission has a cost cap of $700 million ($FY08) and a launch date no later than December 31, 2014. Requirements on planetary protection and education and public outreach are considered as well. These requirements are summarized in Figure 3 and divide the customer requirements into program and mission level requirements.

NASA AO Requirements

Program-Level	Mission-Level
Cost	**Launch**
• $700 million ($FY08) cap	• As primary payload on ELV
• Max cost per year defined	• Atlas/Delta dual compatibility
• Margins defined per phase	**Science**
Schedule	• Return sample from SPA Basin
• Launch no later than 12/31/2008	• Return at least 1 kg
• Concept study up to 7 months	• Samples include soil and rock chips
Implement E/PO Program	**Comply with Planetary Protection**

Figure 3. Summary of primary NASA AO requirements.

Additionally, requirements were derived from the needs of the AO that are consistent with the customer constraints. These lower level requirements address the mission functions, performance, and design during all phases to ensure success of the mission objectives. Specifically, the lower level requirements define appropriate science performance objectives that are not only feasible, but do not constrain solutions beyond the scope of the current design phase. In addition to the customer requirements above, the lower level mission and science requirements can be found in Table VIII in the Appendix.

With the creation of these constraints, an iterative systems engineering process is then utilized to refine and optimize a solution to the customer needs. During subsystem design, the requirements are verified and validated with design decisions. This analysis and validation methodology is useful to ensure a practical solution during tradeoff studies and to prevent conflicts with requirements during system design.

2.3 Functional Analysis and Allocation

The identification of the requirements is transformed into a functional architecture that describes the physical needs of the mission. This top-down method defines the methods and approaches necessary to achieve the system objectives and details the mission timeline. The functional architecture shown in Figure 4 analyzes the functions that accomplish the performance requirements of the Lunar Sample Return mission. The vertical flow of functions represents activities done in series during the mission, while the horizontal direction represents activities accomplished in parallel.

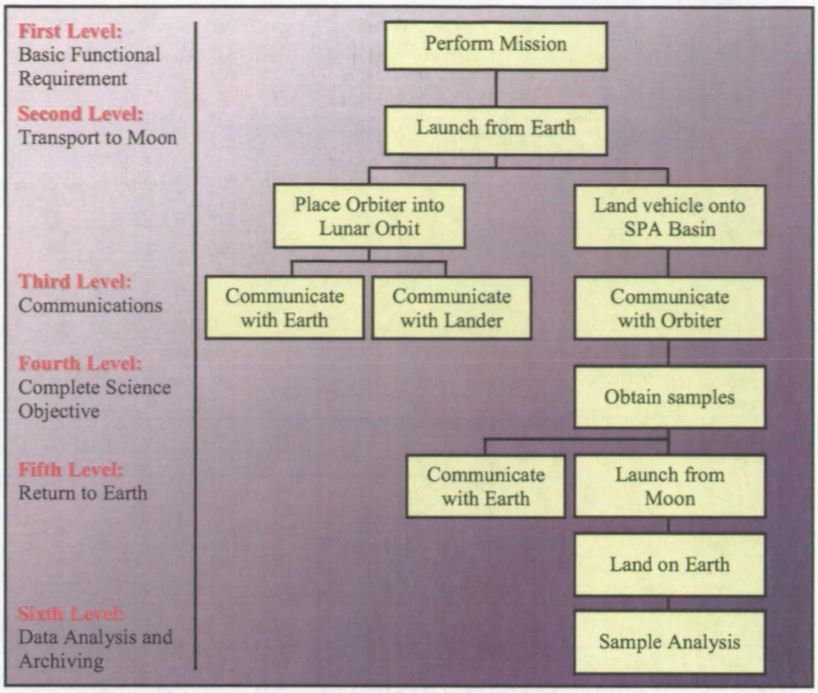


Figure 4. Functional Architecture for the Yamato Mission.

These basic functions not only summarize the timeline of the mission, but also encompass the subsystem needs for mission success. Allocation of these high level functions to the design requirements can be seen in Table I. Requirements Allocation Sheet for Top-Level Functions.Table I.

Table I. Requirements Allocation Sheet for Top-Level Functions.

Function	Function Design Requirements
Perform Mission	NASA AO science objective to perform sample return mission to SPA Basin.
Transport to Moon	Launch will take place no later than Dec. 31, 2008. An orbiter and landing vehicle is required on the far side of the moon. The orbiter must be placed in orbit with coverage of the SPA basin and the Earth. The lander must safely reach the surface in an area with thorium and iron oxide concentrations.
Communications	The orbiter will relay communications between the Earth and the lander and vice versa as required by the location of the SPA basin. The lander will constantly send images and telemetry information to Earth and will receive commands.

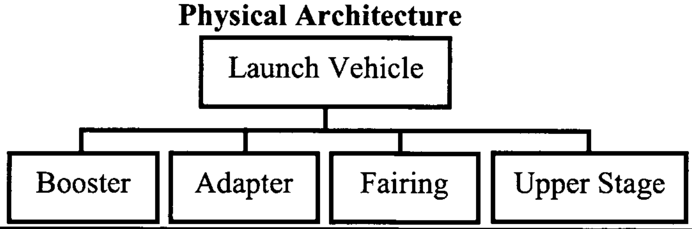
Complete Science Objective	The lander must obtain at least 1 kg of desired lunar samples from the basin for further analysis.
Return to Earth	Once the required samples are obtained, the lander will launch from the surface en route to the Earth. During this time communications will be required directly with the Earth for telemetry and command data. A safe landing on Earth will mark the end of mission operations and begin analysis of the samples.
Data Analysis and Archiving	Once the samples are sent to NASA JSC, they will be archived and distributed. The samples can then be analyzed for scientific gain. During this time data will be published on the NASA Planetary Data System.

Using these functions as a baseline for more detailed development, complete traceability studies are completed. The mapping of functions to the requirements they address are repeated throughout the design process to maintain a design focused on customer needs.

2.4 Synthesis

A physical architecture can be derived from the functional analysis of the mission. Using this methodology, the resultant system is capable of performing the required mission functions within the set performance limits. Each physical component addresses at least one required function and the accretion of all physical components completely defines the system. A Functional/Physical Matrix graphically represents the relationship between the functional and physical architectures. Figure 5 shows this Matrix for the launch vehicle. Similarly, Figure 6 and Figure 7 represent the Orbiter and Lander Matrices, respectively.

Physical Architecture



Functional Architecture

Function Performed	Booster	Adapter	Fairing	Upper Stage
Pre-launch check	X	X	X	X
Launch				
LV Separation		X	X	X
TLI Maneuver				X

Figure 5. Functional/Physical Matrix for the Launch Vehicle.

Orbiter

Function Performed	Communications	Propulsion	Power	ADACS
Operations	X			
Transport to Moon				
Cruise	X		X	X
LOI Maneuver	X	X	X	X
Communicate				
With Lander	X			
With Earth	X			

Figure 6. Functional/Physical Matrix for the Orbiter

Lander

Function Performed	Comms	Prop.	Power	Instruments	ADACS	Sample Container	Return Vehicle
Operations	X						
Transport to Moon							
Cruise	X		X		X		
Lunar Orbit Insertion	X	X	X		X		
Lunar Landing	X	X	X		X		
Communicate	X						
Science							
Sample Selection	X		X	X			
Sample Retrieval			X			X	
Earth Return							
Lunar Ascent Vehicle Launch	X	X	X		X	X	X
Cruise	X		X		X	X	X
EEV Separation	X		X		X	X	
Landing and Recovery	X					X	

Figure 7. Functional/Physical Matrix for the Lander

During detailed design and trade studies, reassessment of the functionality of the mission leads to reconsideration of the decomposed physical architecture, functional architecture, or even derived requirements. Problems arise in schedule and therefore cost, if the design of a system requires many iterations of this methodology. To mitigate this issue, Commercial Off The Shelf (COTS) physical components that address the functional and performance requirements of the mission are used. These products that have been flight-tested to accelerate the schedule and diminish risk of failure for the overall system. Similarly, the reuse of products such as software helps prevent schedule slips and cost overruns.

2.5 Systems Analysis & Control

The LEXco systems engineering process flow is depicted in Figure 8. The NASA New Frontiers Announcement of Opportunity is carefully analyzed for program, mission, and science requirements. These requirements serve as the top-level in the requirements flow. Leaf requirements are derived after performing a thorough literature review of lunar South Pole-Aitken Basin science and consulting industry experts in lunar mission design. Customer requirements are then ranked and mapped to system architecture and payload traits using a Quality Function Deployment (QFD) matrix. Precise definitions of the Figures of Merit (FOMs) are defined in order to eliminate ambiguity when evaluating and down-selecting the various architectures. The FOM utility of each alternative mission architecture and payload suite is then ranked using an Analytical Hierarchy Process (AHP) based on the results of the QFD. A much more detailed assessment of the highest rated architectures is performed using quantitative cost estimation, risk determination and mitigation, and performance analysis. Detailed architecture trades, costing, and subsystem design is performed for the selected architecture and payload suite.

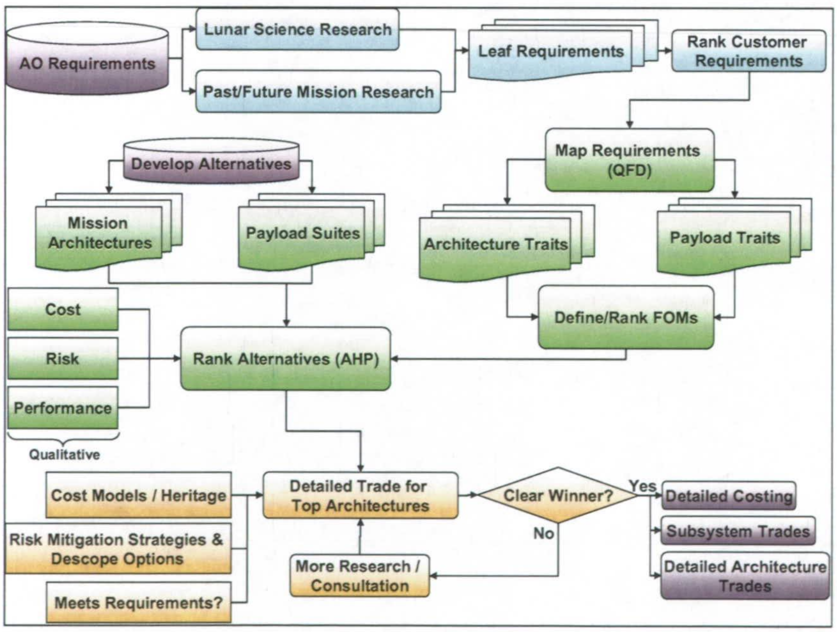


Figure 8. LEXco systems engineering process chart. Our process flow ensures clear traceability between design choices and customer requirements.

LEXco employs the above systems engineering process because design decisions are directly traceable to top-level requirements. The QFD serves to map each aspect of the mission architecture to the customer requirements by considering the importance of each requirement to the customer as well as the inherent risk of each architecture trait. The Yamato mission architecture QFD is shown in Figure 9. The relative risk weighted importance row describes the importance of each architecture trait to the overall merit of the entire mission. This process defines clear objectives for the design team. The red boxes indicate the most important design characteristics. This means that the overall mission architecture is best served by focusing on the sample collection method, sample storage method, and minimizing Life-Cycle Cost (LCC).

		Customer Importance	Sample identification	Sample Collection Method	Sample Storage	Communications network	Lunar Descent/Ascent Propulsion	Orbit propulsion	EDL Architecture	Launch Vehicle	System Technology Readiness Level (TRL)	Program Life-Cycle Cost (LCC)
Performance	Size of return sample (≥ 1kg)	5		9	9		9	3			3	3
	No sample contamination	3		9	3		9		9			3
	Variety of size and type of rocks in sample	5	9	3	9						3	3
	Actively seek rocks of interest (FeO, Thorium sensors)	2	9			9				1	3	3
	Actively seek rocks of interest (camera)	2	9			9				1	3	3
	Move around neighborhood of landing site	1		3	3		3		3	3	9	9
Programmatic	Earth/Moon environmental safety	3		3			3		9	3		1
	System reliability	4	3	9	9	9	9	1	3	3	9	3
	Minimize cost	10	9	9	3	1	3	1	3	9	3	9
Weighted Importance			183	225	168	82	150	29	99	118	117	165
Risk (1-5)			2	5	5	3	4	2	4	2	4	5
Risk Weighted Importance			366	1125	840	246	600	58	396	236	468	825
Relative Risk Weighted Importance			7.1%	21.8%	16.3%	4.8%	11.6%	1.1%	7.7%	4.6%	9.1%	16.0%

Figure 9. Mission architecture QFD. Key customer requirements are mapped to design traits to determine component importance. This is a key input to the AHP down-selection process.

2.5.1 Mission Architecture Trade Studies

Alternative architectures are ranked based on cost, risk, and performance FOMs. Additionally, the FOMs are weighed against one another using the AHP to quantify the evaluation metrics (see Figure 10).

Prioritization Matrix	Low Risk	Cost below $700M	Good Performance	Priority Vector
Low Risk	1.00	3.00	0.33	28.6%
Cost below $700M	0.33	1.00	0.33	14.0%
Good Performance	3.00	3.00	1.00	57.4%

Figure 10. Figure of Merit AHP. The priority vector specifies the importance to the FOM to the overall merit of the mission architecture

Risk is first analyzed based on qualitative analysis of architecture failure modes and potential descope options. Good performance is defined by the ability of a particular architecture to meet our low level science requirements. In particular, the best architectures are those that retrieve the most diverse set of samples both in physical nature and site location. The AHP process reveals the five best design architectures as:

- Communications orbiter + 1 stationary lander
- Communications orbiter + 2 stationary landers
- Communications orbiter + 3 stationary landers
- Communications orbiter + 1 stationary lander with long-range rover
- Communications orbiter + 1 long-range mobile landing craft

The objective of AHP is not to definitively select the baseline architecture, but rather to eliminate unfeasible architectures and help narrow the trade space. Many designs that do not show up in the above list were determined to be prohibitively expensive and were immediately eliminated regardless of the mission's intrinsic technical merit.

All five of the top architectures include a communications orbiter. The nature of this mission requires that a communications relay be placed in an orbit to enable constant communication between the Earth's DSN network and the landing craft on the far-side of the moon. Without this aspect of the architecture the landing craft would need to be completely autonomous and mission risk would increase substantially. The first three architectures all include one or more stationary landers. Each lander could be placed at a different landing site to greatly increase sample diversity. Having multiple vehicles also substantially mitigates intrinsic programmatic risk. The last two architectures introduce local mobility into the system. This increases the ability of mission controllers to choose the area they want to sample within a specified radius of the landing site. The only difference between these two architectures is that the long-range rover is a separate craft from the lander, and the mobile landing craft is a single vehicle.

The results of the detailed trades for the five top architectures are shown in Figure 11. Life-Cycle Cost is estimated based on data from past lunar and planetary missions as well as other proposed lunar sample return missions. Mission-driving risk elements, such as intrinsic lander safety, programmatic descope options, and system complexity are identified and analyzed for each option. Mitigation strategies for each risk element are determined, and the effect of each strategy on overall mission risk is analyzed. The final FOM-weighted scores of each option are determined based on the priority vector shown in Figure 10. The highest scoring option contains three stationary landers, but cost research and industry consultation suggest that this architecture is too close to the $700M cap. The selected baseline architecture contains two stationary landers with one communications relay. A similar down-selection process is used to determine the optimal suite of instruments.

Architecture (incl. relay)

	1 Lander	2 Landers	3 Landers	Lander + long-range rover	Long-range mobile landing craft
Cost Element (FY08$M)					
DDT&E + Production	390	490	590	450	500
Launch Vehicle	60	60	60	125	125
Estimated NASA LCC	450	550	650	575	625
Risk Element					
Intrinsic Lander Safety	1	6	9	1	1
Mitigation Strategy	Use safer landing site	Use safer landing site		Use safer landing site	Use safer landing site
Effect of strategy on risk element	4	8	9	4	4
Intrinsic Descope Options	0	6	10	6	6
Mitigation Strategy	None	Descope to one stationary		Descope to lander only	Immobilize
Effect of strategy on risk element	0	7	10	7	7
Intrinsic System Complexity	8	6	4	2	3
Mitigation Strategy		Descope to one stationary	Descope to one or two stationary	Descope to lander only	Immobilize
Effect of strategy on risk element	8	9	1	5	6
Performance Element					
Diversity of Locations	2	7	10	6	6
Total Scores					
Cost Score	10	7	1	6	1
Risk Score	2	8	10	3	3
Performance Score	2	7	10	6	6
Totals	3.1	★7.3	8.7	5.1	4.4

Good = 7-10
Marginal = 4-6
Poor = 0-3

Figure 11. Results of detailed architecture trade. The selected baseline architecture (indicated by a star) provides NASA with the best combination of cost, risk, and performance.

2.5.2 Baseline Architecture and Traceability

Our baseline architecture provides NASA with the optimal combination of cost, science capability, and reliability. Cost engineering tools are used throughout the design process to insure the design would fit within the NASA funding profile with adequate margin. The performance floor of the architecture is determined by analyzing descope options of the overall architecture and the instrumentation suite. A requirements traceability analysis is performed in order to insure all program, mission, and science requirements are met by both the performance floor and baseline designs. The system traits of the Yamato mission baseline architecture are shown in Table II.

Table II. Baseline architecture system traits.

System Trait	Baseline
Landers	Two Stationary (identical)
Program LCC w/ Margin	$592M (FY08)
Launch Vehicle	Atlas V (541)
Communications	Relay Satellite + S-Band DSN Relay
Lunar Landing	Three Legs + Crushables
Science Collection	Two Arms/Sieves/Scoops, Core Drill, Small Rover
Visibility	Camera + Illumination

Mission Duration	14 Earth Days (1 lunar day)
Sample Storage	Cryocooler + Storage Container
Lunar Ascent Vehicle	Includes Earth Entry Vehicle
EDL	PICA heat shield + Mid Air Retrieval

Like the Spirit and Opportunity rovers currently on Mars, Yamato is a two-vehicle mission. The landers are named for their function: Aitken Basin Lunar Excavator (ABLE 1 and ABLE 2). Each lander visits a different landing site within the SPA with different regolith and risk characteristics. This accomplishes the goal of achieving samples that are diverse in physical nature and site location. Having a second lander also introduces a high degree of redundancy in the system architecture. A single communications relay is stationed to provide constant communications between the Earth and the landers. An autonomous emergency software sequence is programmed into each lander in the rare case of relay failure. The pseudocode for this risk mitigation process is included in the appendix (see Figure 22).

The final landing sites are selected based on the availability of data from the Lunar Reconnaissance Orbiter (LRO) with a planned launch in October 2008. Current data sets from SMART 1 and SELENE provide moderate resolution of the lunar SPA Basin, but LRO data is needed before definitive decisions can be made about potentially high risk landing sites. The landing site for ABLE 1 is likely in the iron and thorium rich Olivine Hill area (160°W, 58°S). Current data suggests that Olivine Hill contains the type of samples specified in the mission requirements. The area contains some small craters and rocks, so this has been labeled a medium-risk landing site. ABLE 1 and ABLE 2 have been designed with the requisite software, propulsion, and attitude determination and control system (ADACS) to maneuver autonomously in the local area of the selected landing site. This provides an additional level of risk mitigation. The landing site for ABLE 2 is likely closer to the lunar South Pole itself, where the terrain is much more flat and volatiles are expected to exist.

A primary goal of our systems engineering process is to minimize programmatic risk. We have discussed how this important mission aspect has been considered throughout the systems engineering process. Table III demonstrates how the Yamato mission can mitigate programmatic risk in case of schedule or cost overruns. The performance floor is defined based on descope options for the primary science requirements. Table IV shows the full traceability of the Yamato mission architecture traits to key mission requirements. The requirements listed in these two tables are consistent with the full requirements breakdown in the Appendix of this document.

Table III. Several baseline-to-performance floor descope options greatly reduce programmatic risk.

Science Requirement	Baseline	Descope Option	Performance Floor
Recover Volatiles	Drill 1-2m for core sample	Remove drill	No volatiles
Maintain Sample Integrity	Cryocooler with separate compartments	Replace with unpowered container	No sample cooling
Recover Regolith	Two arms/scoops per lander	Remove one arm/scoop	One arm/scoop
Sample Diversity	Two identical landers	Remove instruments on one lander and/or remove one lander completely	One lander with arm/scoop

Table IV. Yamato mission requirement traceability. The mission baseline and performance floor exceed the stated mission requirements.

Key Mission Requirement	Yamato Mission Architecture Trait
Primary payload on ELV	Atlas V (541)
Cost < $700M (FY08)	7.1% cost margin
Recover at least 1kg of lunar samples	2-4 kg of regolith, rocks, core samples
Ensure sample diversity	Two unique landing sites, 2m radius of each scoop arm, drill, small rover for local exploration
Address lander safety	ABLE 2 to low risk landing site, autonomous local landing site selection
Minimize programmatic risk	Many descope options for instrument suite and architecture
Insure human safety	All Planetary Protection Requirements complied with
Advanced technology consideration	Novel orbit design, SRGs, PICA heatshield, Mid-Air Retrieval (MAR)

Yamato is a novel mission design that mitigates risk from end-to-end while simultaneously maximizing science benefit. The operational view depicted in Figure 12 demonstrates the entire lunar sample return mission from launch to sample recovery.

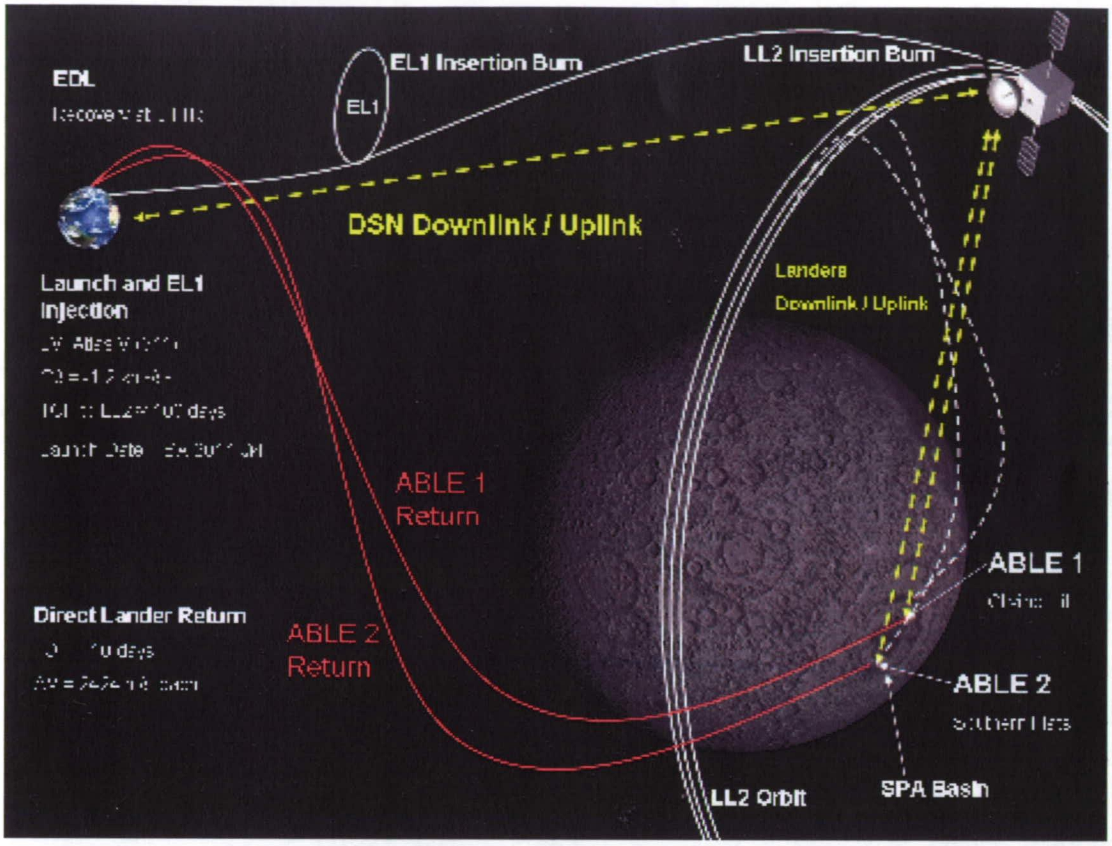


Figure 12. Yamato mission operational view. The redundant mission architecture mitigates mission risk while delivering a diverse set of samples.

The two ABLE landers and the communications relay are launched as the primary payload on a single Atlas V 541 launch vehicle from Kennedy Space Center (KSC). The system is configured in the Atlas V fairing so that the communications relay is in between ABLE 1 and ABLE 2 in the stack. The rocket engines of the ABLE 1 and ABLE 2 landers point in opposite directions such that thrust is directed along the axis of the system. The Atlas V injects the entire configuration into a 100 day journey to a Lissajous orbit at the Lunar Lagrange point 2 (LL2) via the Earth Lagrange point 1 (EL1). ABLE 1 is detached from the configuration once the system is stabilized in LL2 and the lunar day is set to begin in approximately two Earth days. Six hours later, ABLE 2 detaches from the orbiter. Detaching the landers in series phases out the landing sequences of the landers and gives ground operators on Earth enough time to analyze each landing sequence. Science operations begin as soon as Earth communications is established.

Science operations terminate twelve hours before the end of the lunar day (approximately 14 Earth days after landing) and pre-launch operations will begin. A Lunar Ascent Vehicle (LAV) containing the Earth Entry Vehicle (EEV) is integrated into each lander for the Earth-return phase. The LAV makes use of most of the lander bus for the subsystems, but does not include any of the science instruments or landing structure from the lander. Leaving these items on the Moon drastically reduces LAV mass and significantly reduces the propellant required throughout each phase of the mission. Both LAVs leave the Moon on a direct orbit back to Earth that lasts

10 days. Choosing a direct orbit back to Earth minimizes time of flight (TOF) in order to reduce the chances of sample contamination. The EEV is jettisoned from the LAV once Earth is reached and Mid-Air Retrieval via helicopter is initiated at the Utah Test and Training Range (UTTR). The payload is flown via helicopter to the Michael Army Airfield (MAAF) and immediately sent to the curatorial facility at NASA Johnson Space Center (JSC).

2.5.3 Computer Aided Design (CAD)

The lander configuration is depicted in Figure 13. A three-legged design is selected based on past and future mission heritage and its inherent stability. A low center of gravity is maintained to improve the vehicle slope tolerance. This is necessary due to uncertainty of the lunar SPA slope characteristics available from lunar data sets. Crushable mass is included in each of the legs to attenuate the loads from impact. Ground sensors are installed in the legs that trigger termination of the main engine immediately after touchdown.

The central module contains most of the spacecraft subsystems. A communications dish is placed on top of the lander for connectivity with the relay satellite. The EEV is recessed into the side of the central module. The EEV backshell is hinged to grant access to the storage container during sample retrieval. The scoop arms are placed on opposite sides of the lander to maximize the available area for regolith collection. The small rover is secured on the opposite side of the two arms and it exits the vehicle via a deployable ramp. The central module also acts as the LAV during Earth return. The triangular lander base and all of the science instruments are left on the lunar surface as the LAV returns to Earth.

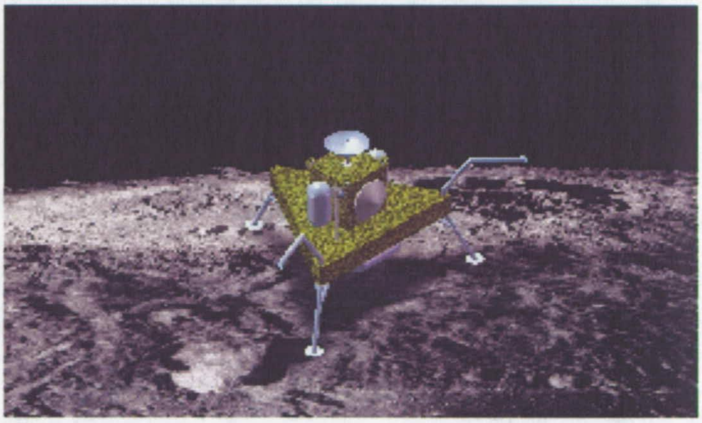


Figure 13. Notional CAD drawing of one of the ABLE landers. The structure and layout is optimized for stability and soil accessibility.

2.5.4 Detailed Architecture Trade Studies

Mission characteristics determined during functional analysis identify the key trades that must be conducted. Selection of a launch vehicle early in the design process is essential to ensure cost control and configuration management. Preliminary mass estimates and required launch energy (C3) to EL1 define the candidate launch vehicles. The Expendable Launch Vehicle Performance

Tool (ELVPERF) provided by NASA Kennedy Space Center (KSC) is employed. An Atlas V (541) is selected based on the performance results and mass margin shown in Figure 14.

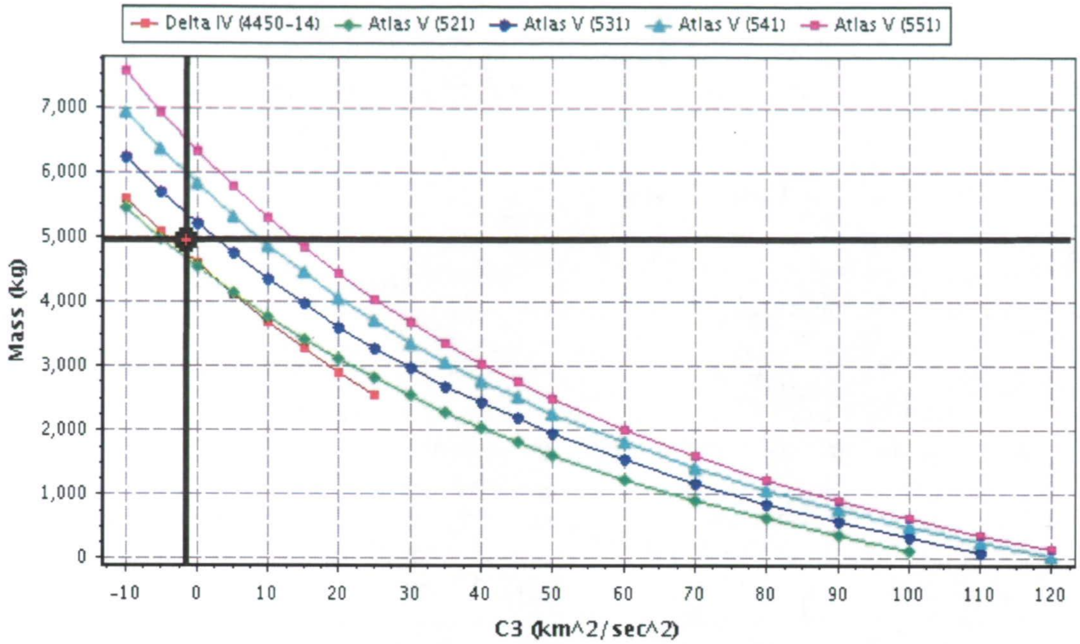


Figure 14. Candidate launch vehicle performance chart. The Atlas V (541) launch vehicle is selected based on its mass capability to EL1 and mass margin.

Orbit determination and optimization is essential for the Yamato mission due to the high ΔV requirements to get to and from the Moon. The orbit must also provide for a communications relay that can link the landers to the DSN network. In-house orbital mechanics tools, lunar transfer literature, and the Satellite Took Kit (STK) are used to trade between different orbit methods. A low-energy EL1 to LL2 orbit is selected for the Earth-Moon trajectory, and a direct return orbit is selected from the Moon-Earth trajectory. The optimized ΔV for each phase of the orbit is given in Table V. Graphics from our STK orbit simulation are shown in Figure 15.

Table V. Optimized ΔV Budget for selected orbit.

	Combo ΔV (m/s)	Lander ΔV (m/s)
Translunar Injection	3185	
EL1 Insertion Burn	30	
LL2 Insertion Burn	10	
Lander Departs LL2		35
Lander Landing		2335
Lander Liftoff		2424
Total	3225	4794

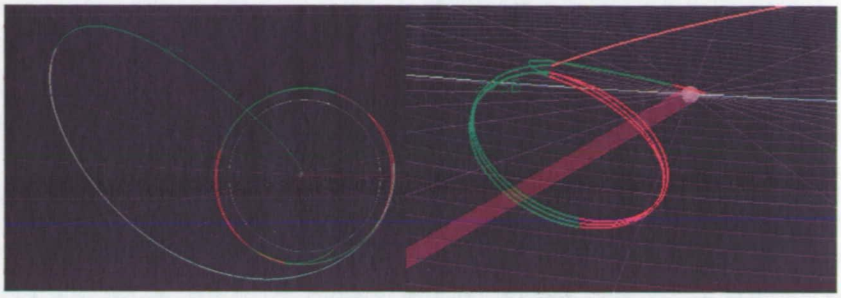


Figure 15. Earth to EL1 orbit (left) and LL2 Lissajous orbit (right). The Yamato orbit is optimized to minimize ΔV where necessary using low energy transfers.

The exact nature of the EEV is an additional required trade identified during functional analysis. In particular, it is necessary to decide whether it is preferable to bring the EEV all the way to the Moon and back, or instead leave the EEV in Low Earth Orbit (LEO) and perform a rendezvous after Earth return. Rendezvous is an appealing option because it does require have to expend propellant carry the extra EEV mass all the way to and from the lunar surface. However, performing a LEO rendezvous introduces significant complexity into the design and is a single point failure mode. A trade study is performed to examine how the extra propellant required to carry the EEV all the way to and from the moon propagates throughout the various phases of the trajectory. This mass is compared to the total estimated propellant mass if the EEV is left in LEO. The results of the trade study as a function of the architecture component dry mass ratios is depicted in Figure 16. The precise masses of each component were not known at the time of the trade study, so the probable design space is outlined in the graphic. It is clear that rendezvous is never more than 10% cheaper in terms of propellant than carrying the EEVs to the Moon and back. It is determined that this minor cost savings is not worth the added risk and complexity of a LEO rendezvous.

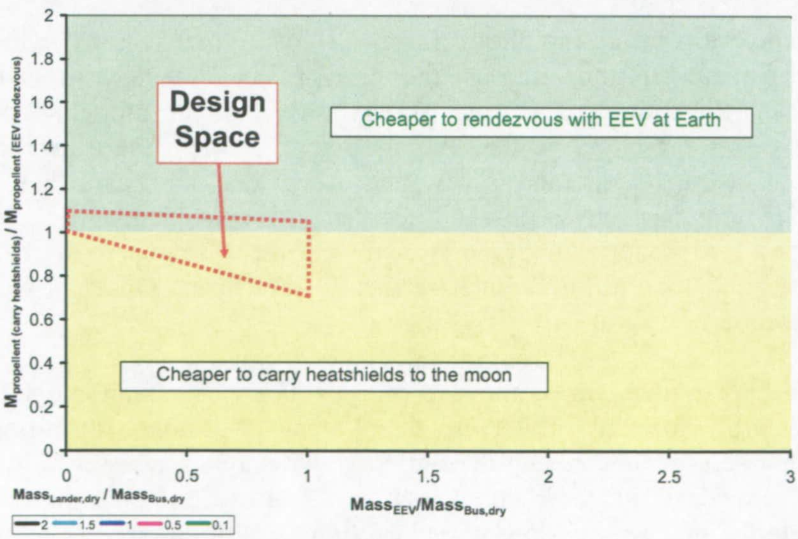


Figure 16. Trade study to determine feasibility of LEO rendezvous. It is never more than 10% cheaper in terms of propellant to perform rendezvous.

The lander power system presents an interesting challenge to the design team. The SPA is inherently dark and can recieve little to no direct sunlight. This eliminates solar cells as a feasible power option for the landers. Solar panels with eclipse batteries are still employed for the orbiter and LAV power systems, but non-solar power must be used for the landers themselves. Fuel cells and nuclear Radioisotopic Power Sources (RPS) are considered. The fuel cells considered are desirable due to their relatively low cost compared to RPS. Concerns about non-storable fuel boil-off during the transfer to the Moon led the design team to choose Stirling Radioisotopic Generators (SRGs). This form of nuclear power has the best combination of cost and performance compared to other nuclear power sources, such as Multi-Mission RTGs (MMRTGs) and General Purpose Heat Source Radioisotopic Thermoelectric Generators (GPHS-RTGs). LEXco has accurately accounted for all extra nuclear launch operations costs and other RPS provisioning costs.

3 Transitioning Critical Technologies

An ancillary goal of the New Frontiers program is to encourage the use of advanced technologies to reduce mission cost and achieve performance enhancements. These technologies should only be used provided that appropriate risk mitigation measures are included. The Yamato mission incorporates critical technologies in the power subsystem, the recovery system, and the trajectory design. Development testing and evaluation is conducted and accounted for in order to mitigate the risk of implementing these novel ideas.

Stirling Radioisotope Generators (SRG) are currently under development by Glenn Research Center and Lockheed Martin. Electrical power is created in these radioisotope thermoelectric generators by harnessing the heat produced by a radioactive material. The development of this technology is being supported by the Yamato team in order to achieve the power needs of the landers in the harsh conditions of the SPA. The Yamato team is working closely with Glenn Research center in assuring size and interface capabilities with a real spacecraft system.

An innovative choice of orbit design allows for a low-cost, efficient transfer from the Earth to the Moon for the Yamato Mission. Because the mission is unmanned this transfer is optimized for cost rather than flight time. This cost is a byproduct of the propellant fuel required to complete the transfer, represented by the ΔV required. Although the utilization of the Earth Lagrangian point as a transfer method is new, the use of this point as a final destination has already been realized for space observing satellites such as the Wilkinson Microwave Anisotropy Probe, the Solar and Heliospheric Observatory, and Genesis. Missions to the Lagrangian points have also been planned for the future such as the Hershel Space Observatory and the James Webb Space Telescope.

Unlike the Earth Lagrangian Points, the use of the Lunar Lagranian point has never been exploited. This orbit for the orbiter of the Yamato Mission will allow for constant communication between the landers and the Earth, which is a critical asset to the success of the mission. Extensive research has been completed for the use of this point to allow for constant communications during manned missions with the Orion/CEV program.

By leveraging the research already completed for the use of the Lagrangian points as a transfer method and as a final orbit method, a more refined time of flight and fuel cost can be obtained. Risk is mitigated further through the use of high fidelity modeling and simulation software created by experts in the area. This software includes the LTool package designed by NASA JPL specifically for the optimization of Lagrangian orbits. Also, more developed simulation of the orbit has been accomplished with the help of Mike Loucks of Space Exploration Engineering Corp., an expert in the Satellite Tool Kit (STK) and Astrogator software package. These methods of transitioning the orbit design will allow the Yamato mission to maintain a low cost without the risk of failure.

Mid-Air Retrieval (MAR) is an EEV recovery method that has been used for decades by the Department of Defense (DoD) to recover payloads from spy satellites. The most recent attempt at Mid-Air Retrieval would have been performed during recovery of the Genesis Sample Return Capsule. Unfortunately, the switch to trigger the parachute sequence was installed upside down, so the Genesis SRC crashed in the Earth at UTTR.

The LEXco team is working with Vertigo, Inc., a Lake Elsinore, CA based company that specializes in MAR operations and Parachute Recovery System (PRS) design. They have extensively tested and demonstrated the success of MAR on several sample payloads. The MAR system has been optimized for safety and reliability. After the EEV deploys a parafoil at approximately 10,000 ft to give the capsule a 2:1 glide ratio a helicopter then flanks from the side of the EEV glide slope and captures a trailing cable with a grappling hook. This process is depicted in Figure 17.

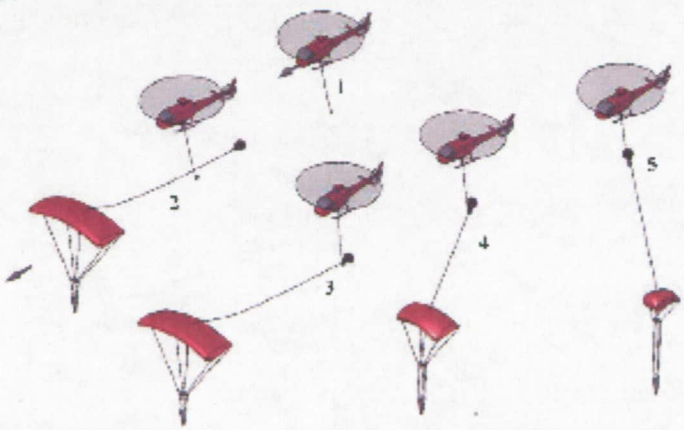


Figure 17. Vertigo 3G Mid-Air Retrieval process. MAR provides the fastest response time of any recovery option and eliminates the need for crushable mass.

4 Integration of System Engineering Effort

4.1 Team Organization

The system engineering effort is led by the LEXco team in Atlanta, GA. LEXco is an independent engineering firm focusing in space design and lunar excavation missions. The team organizational flow for the Yamato mission can be seen in Figure 18 and the group members and responsibilities are as follows:

- Scott Martinelli, Project Manager, responsible for mission validity; science, cost, and risk balance; internal and external scheduling; and team leadership.
- Brandon Smith, Lead Systems Engineer, responsible for requirement flowdown, system engineering planning and allocation, major architecture trade studies, and programmatic margins.
- Neal Patel, Mission Systems Engineer, responsible for mission and orbit trades and design, launch vehicle selection, navigation, and space environment.
- King Lam, Spacecraft Flight Systems Engineer, responsible for subsystem trades and design, CAD development, mass and power estimation, and communication architecture.
- David Powell, Science Payload Engineer, responsible for science payload design, landing sites selection, and literature review.

The Principal Investigator (PI) for the mission is Dr. Michael Duke of the Colorado School of Mines. Dr. Duke is an expert on lunar samples and lunar resource collection. He has worked extensively with Apollo moon samples and is developing methodology for future return samples and in-situ resource collection on the moon. Co-Investigators include Dr. Mike Loucks of Space Exploration Engineering Co., an expert in orbital navigation and propagation, Dr. Charles Shearer of the Institute of Meteoritics, a leading planetary geologist, and Mr. Craig Peterson of the Jet Propulsion Laboratory, a senior mission design engineer.

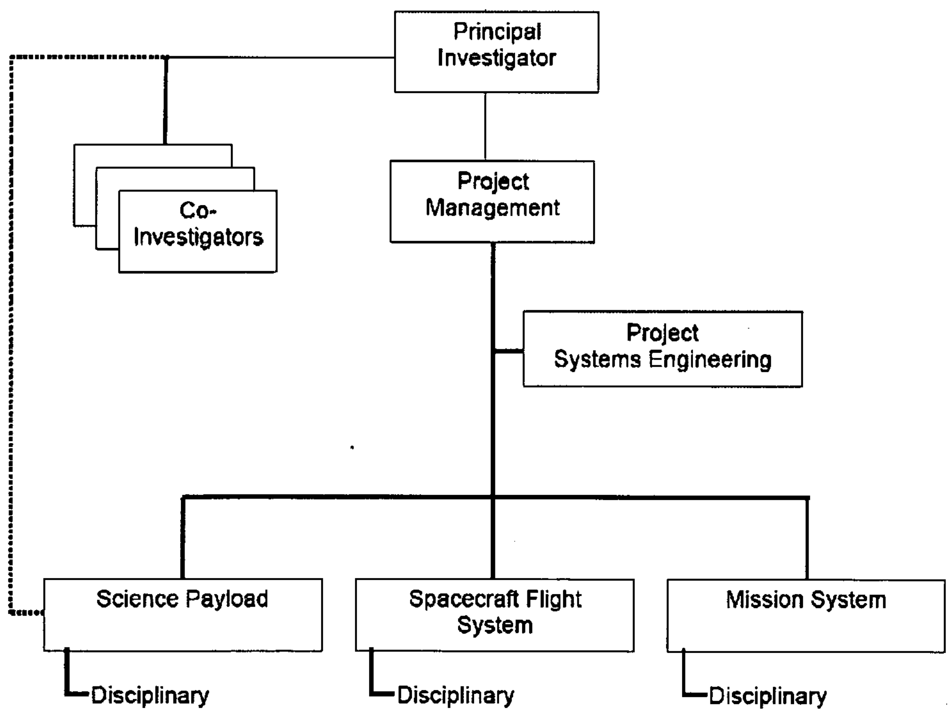


Figure 18: LEXco team organizational flow designed to specialize in integration of system design.

4.2 Program Schedule

The implementation of the system engineering plan is governed by the master program schedule and monitored by various design and program reviews to ensure quality, cost and schedule are met at all phases of design and implementation. The detailed schedule, seen in Figure 21 in the Appendix, is given in phase condensed form in Figure 19. The schedule includes margins in all phases and takes into consideration the long lead time of the nuclear launch permissions. Mission Definition Review (MDR) occurs at the culmination of Phase A on October 1, 2009. System Definition Review (SDR) and the Preliminary Design Review (PDR) occur during Phase B with PDR governing the transition from Phase B to Phase C on January 15, 2010. The Critical Design Review (CDR) in between Phase C and D occurs on April 15, 2012. The System Acceptance Review (SAR) and Flight Readiness Review (FRR) occur in Phase D in preparation for launch in mid-November 2014, within the 47 month time constraint on Phase C to launch timing. A nominal 125 day mission is then commenced with dedicated operations and post flight science analysis in Phase E

<table>
<tr><td>FY09</td><td>FY10</td><td>FY11</td><td>FY12</td><td>FY13</td><td>FY14</td><td>FY15</td><td>FY16</td><td>FY17</td></tr>
<tr><td>Phase A</td><td>Phase B</td><td colspan="2">Phase C</td><td colspan="2">Phase D</td><td colspan="3">Phase E</td></tr>
</table>

Figure 19: Condensed schedule demonstrating adequate schedule timing to facilitate mission success.

5 Implementation Tasks

LEXco engineering processes and implementation procedures are consistent with the standards outlined in the ISO 9000 series. LEXco employees are thoroughly trained in quality control procedures and LEXco management conducts process proofing at regular intervals. Sufficient time is allocated in the program schedule for all technology verifications, manufacturing of engineering test articles, and development tests and evaluation. These activities are all geared toward ensuring the quality of the final product and maximizing value to the customer.

6 Cost Engineering and Estimation

Cost estimation techniques are used at every phase of the design process. LEXco insures that all development and hardware costs are accounted for and clearly shown in the full cost breakdown (see Figure 23 in the Appendix). Several different sources of information are used to estimate overall mission LCC. New Frontiers cost documentation is used wherever available. Instruments and subsystems are costed primarily using mass-based Cost Estimation Relationships (CERs) and bottom-up numbers where available. Table VI shows the cost estimation method for various mission cost elements.

Table VI. Cost estimation method for various mission cost elements.

Cost Element	Estimation Method
Launch Vehicle + Ops	NF Launch Services Information Summary
DSN	NASA Mission Ops. and Comm. Services
Instruments	Mass CER / Bottom-up
Subsystems	Mass CER
RPS (SRGs) + Provisioning	Specifications for Space RPS for NF

All cost requirements from the AO are verifiable by tracing the requirements to the full cost breakdown. A summary of the mission LCC is given in Table VII. Notice that 25% reserves are included for all costs through phase D (not including the launch vehicle and RPS) to allow for cost growth. Included in these costs are the contingencies for each cost component. The total NASA cost of $651M (FY08) provides an ample 7.3% margin from the cost cap of $700M (FY08).

Table VII. Yamato mission cost summary. Contingencies are includes for each cost contribution and ample margins are included at each phase.

Cost Element	Amount (FY08)
Phase A-D (no reserves)	$297.3M
Phase A-D reserves (25%)	$74.3M
Phase E (no reserves)	$22.5M
Phase E Reserves (15%)	$5.6M
E/PO (included in Phase A-E costs)	$5.5M
Launch Vehicle (+ nuclear ops)	$159.4M
RPS + Provisioning Costs	$94.9M
Total NASA Cost	**$651.9M**
LSR Mission % Margin	**7.3%**

The five-year forecast of NASA funding is provided by the NASA program office. This data determines the rate of funding available and helps delineate the LEXco spending chronology. Figure 20 shows the maximum NASA funding for each year, as well as the proposed spending for each year by the LEXco team. The green line represents the total rollover available. It is clear that design team has adequately planned the schedule so that sufficient funding is available at each phase.

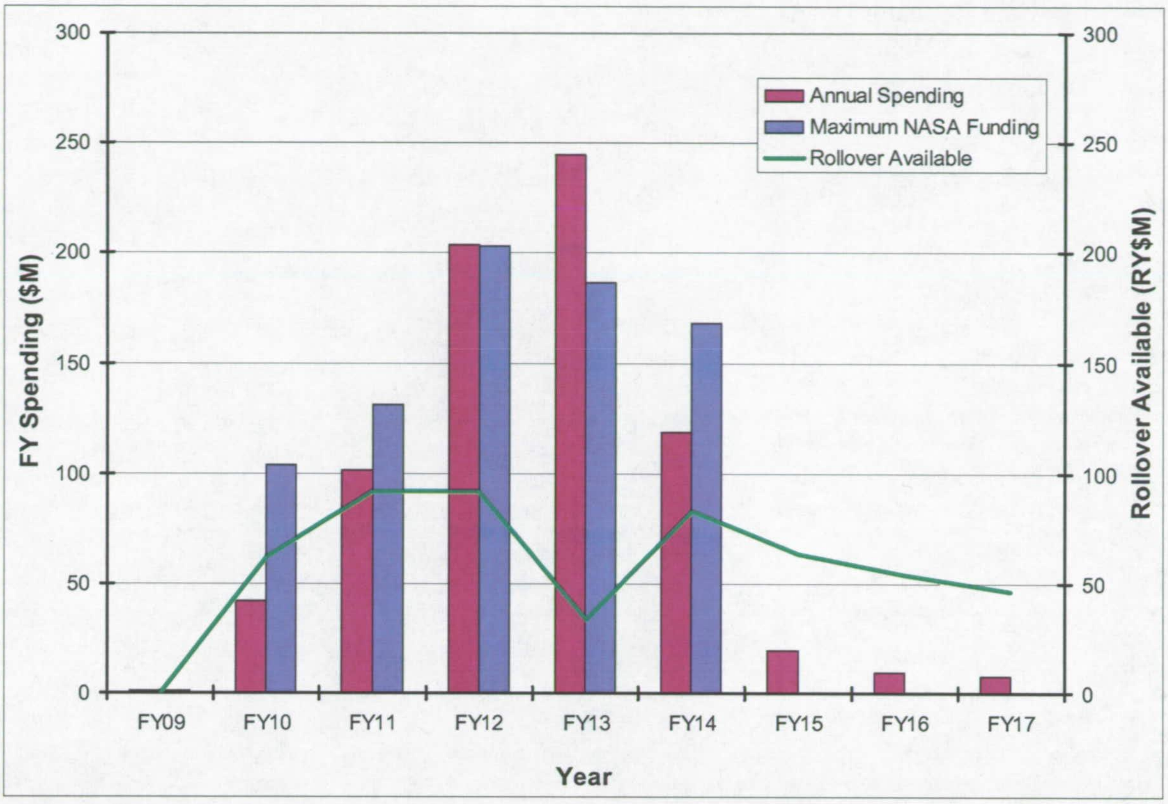


Figure 20. Mission spending profile and annual rollover available.

7 Concluding Remarks

Through a well defined systems engineering process the Yamato mission design has been optimized for a low-cost, high performance solution to the customer needs. This process, beginning from the NASA AO requirements through detailed cost budgets and scheduling, achieves the overall mission objectives and can be easily reiterated to address changes in the system architecture. The utilized methodology not only places a check and balance relationship on the feasibility of the design, but also synergistically combines a system of parts into a whole so as to create an advanced lunar sample return mission.

The LEXco team would like to thank the following people for their support during the Yamato mission design:

- Dr. Carlee Bishop
- Dr. Robert Braun
- Mr. Chris Cordell
- Dr. Juan Cruz
- Dr. Michael Duke
- Dr. Mike Loucks
- Mr. Craig Peterson

- Mr. Bala Radharamanan
- Dr. Joseph Saleh
- Dr. Charles Shearer

8 Appendices

8.1 Supplementary Tables and Figures

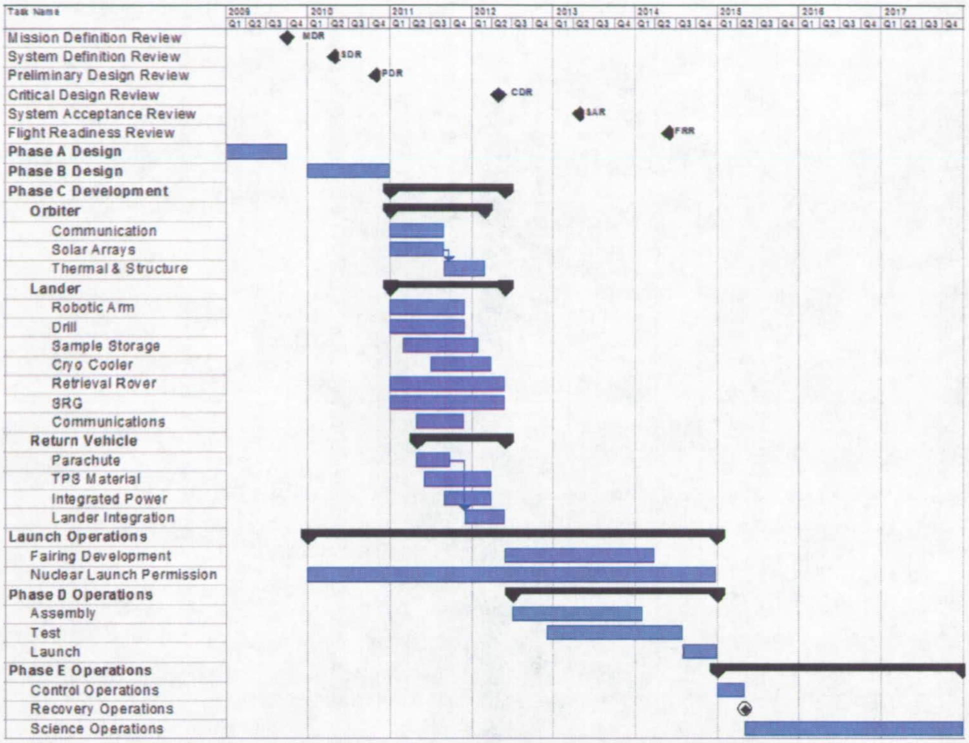


Figure 21: Yamato program schedule including adequate margins.

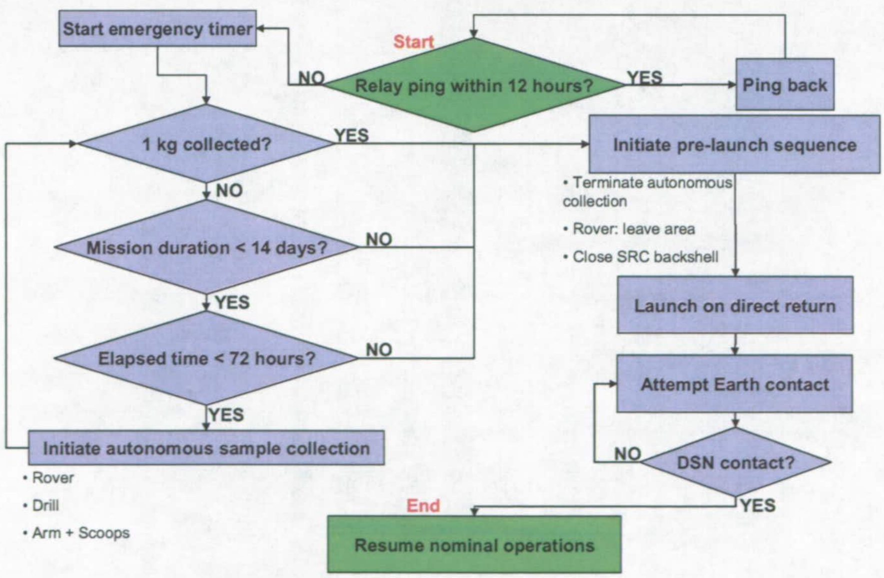


Figure 22. Software design to mitigate possible communications relay failure. The landers can autonomously meet the mission performance floor and return to Earth regardless of relay status.

	Formulation									Subtotal Formulation		Implementation									Subtotal Implementation		Total Life Cycle Cost	
Cost Element	FY09	FY10	FY11	FY12	FY13	FY14	FY15	FY16	FY17	RY$	FY08$	FY09	FY10	FY11	FY12	FY13	FY14	FY15	FY16	FY17	RY$	FY08$	RY$	FY08$
Start to Launch + 30 days (Phases A/B/C/D)																								
Phase A Concept Study	0.96	0	0	0	0	0	0	0	0	0.96	0.93	0	0	0	0	0	0	0	0	0	0	0	0.96	0.93
Proj. Mgmt/Miss. Analysis/Sys. Eng.	0	1	0	0	0	0	0	0	0	1	0.94	0	3	3	4	4	3	1	0	0	18.00	15.84	19.00	16.78
Scoop + Sieve	0	0	0.50	0	0	0	0	0	0	0.50	0.46	0	0	0	0.77	1.33	0	0	0	0	2.10	1.83	2.60	2.28
Drill	0	0	4.45	0	0	0	0	0	0	4.45	4.06	0	0	0	6.89	11.84	0	0	0	0	18.72	16.26	23.18	20.32
Arm	0	0	3.12	0	0	0	0	0	0	3.12	2.85	0	0	0	4.83	8.30	0	0	0	0	13.14	11.40	16.26	14.26
Mast	0	0	1.47	0	0	0	0	0	0	1.47	1.34	0	0	0	2.28	3.92	0	0	0	0	6.20	5.38	7.67	6.72
High-Res Camera with Light	0	0	1.02	0	0	0	0	0	0	1.02	0.93	0	0	0	1.57	2.71	0	0	0	0	4.28	3.72	5.30	4.65
Cryogenic Sample Container	0	0	2.36	0	0	0	0	0	0	2.36	2.15	0	0	0	3.65	6.27	0	0	0	0	9.92	8.61	12.27	10.76
Landing Camera	0	0	1.00	0	0	0	0	0	0	1.00	0.91	0	0	0	1.50	1.50	0	0	0	0	3.00	2.62	4.00	3.53
Small Rover	0	0	11.06	0	0	0	0	0	0	11.06	10.09	0	0	0	17.10	28.50	0	0	0	0	45.60	39.60	56.65	49.68
Instr. Integration, Assembly and Test	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	1	4	0	0	0	5	4.19	5	4.19
Subtotal - Instruments	0	0	24.99	0	0	0	0	0	0	24.99	22.80	0	0	0	38.59	65.36	4	0	0	0	107.95	93.59	132.94	116.39
Spacecraft Bus / Comm. Relay	0	5.71	11.77	12.13	10.42	0	0	0	0	40.03	35.79	0	0	0	3.14	6.47	6.67	0	0	0	16.27	13.88	56.30	49.67
Lander 1 (no instruments)	0	6.15	12.69	13.08	11.24	0	0	0	0	43.16	38.59	0	0	0	3.25	6.71	6.92	0	0	0	16.88	14.40	60.04	52.99
Lander 2 (no instruments)	0	0	0	0	0	0	0	0	0	0	0	0	0	0	3.25	6.71	6.92	0	0	0	16.88	14.40	16.88	14.40
S/C Integration, Assembly and Test	0	0	0	0	0	2	0	0	0	2	1.67	0	0	0	0	5	9	0	0	0	14	11.79	16	13.45
Other Hardware Elements	0	1	1	1	1	1	0	0	0	5	4.43	0	0	0	0	5	5	0	0	0	10	8.46	15	12.88
Launch Ops (Launch +30 days)	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	3.00	2.00	0	0	5.00	4.11	5	4.11
Subtotal - Spacecraft	0	12.86	25.46	26.21	22.66	3	0	0	0	90.19	80.48	0	0	0	9.64	29.88	37.50	2	0	0	79.02	67.02	169.22	147.50
Science Team Support	0	0	0	0	0	0	0	0	0	0	0	0	2	2	2	1	1	0	0	0	8	7.17	8	7.17
Pre-Launch GDS/MOS Development	0	0	0	0	2	1	0	0	0	3	2.55	0	0	0	0	2	1	0	0	0	3	2.56	6	5.10
DSN/Tracking	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0.80	0	0	0	0.80	0.67	0.80	0.67
E/PO	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0.00	1.61	1.66	0	0	0	3.26	2.76	3.26	2.76
Other	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Subtotal (A-D), before Reserves	0.96	13.86	50.44	26.21	24.66	4	0	0	0	120.14	107.70	0	5	5	54.24	103.84	48.95	3	0	0	220.04	189.59	340.18	297.30
Instrument Reserves	0	0	6.25	0	0	0	0	0	0	6.25	5.70	0	0	0	9.65	16.34	1	0	0	0	26.99	23.40	33.23	29.10
Spacecraft Reserves	0	3.22	6.36	6.55	5.67	0.75	0	0	0	22.55	20.12	0	0	0	2.41	7.47	9.37	0.5	0	0	19.76	16.76	42.30	36.88
Other Reserves	0.24	0.25	0	0	0.5	0.25	0	0	0	1.24	1.11	0	1.25	1.25	1.5	2.15	1.86	0.25	0	0	8.27	7.24	9.51	8.35
Total Phases A/B/C/D	1.2	17.33	63.05	32.77	30.83	5	0	0	0	150.17	134.63	0	6.25	6.25	67.80	129.81	61.19	3.75	0	0	275.04	236.99	425.22	371.62
Launch + 30 Days to End of Mission (Phase E)																								
Mission Ops./Data Analysis/Proj. Mgmt	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	10.88	6.31	6.50	23.69	18.67	23.69	18.67
DSN/Tracking	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	1.35	0	0	1.35	1.09	1.35	1.09
E/PO	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	1.71	1.76	0	3.47	2.76	3.47	2.76
Other	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Subtotal Phase E before Reserves	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	13.94	8.07	6.50	28.51	22.52	28.51	22.52
Reserves	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	2.09	1.21	0.98	4.28	3.38	4.28	3.38
Total Phase E	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	16.03	9.28	7.48	32.79	25.89	32.79	25.89
Launch Services (Non-Nuclear)	0	0	0	0	0	0	0	0	0	0	0	0	0	2.33	68.46	61.91	38.30	0	0	0	171.00	147.75	171.00	147.75
Nuclear Mission Additional Launch Costs	0	0	0	0	0	0	0	0	0	0	0			2.33	4.80	3.71	2.55	0	0	0	13.40	11.69	13.40	11.69
RPS(s)																								
NEPA Compliance / EIS	0	1.30	0.84	0.10	0.11	0.11	0	0	0	2.46	2.26	0	0	0	0	0	0	0	0	0	0	0	2.46	2.26
Nuclear Launch Safety Approval	0	1.01	1.91	3.24	2.69	1.41	0	0	0	10.25	9.04	0	0	0	0	0	0	0	0	0	0	0	10.25	9.04
Emergency Preparedness	0	0.12	0.12	0.13	0.13	2.17	0	0	0	2.67	2.26	0	0	0	0	0	0	0	0	0	0	0	2.67	2.26
Spacecraft Accom., Processing and Int.	0	0.29	0.67	1.46	5.32	5.48	0	0	0	13.21	11.30	0	0	0	0	0	0	0	0	0	0	0	13.21	11.30
Risk Communication	0	0.26	0.27	0.28	0.68	1.11	0	0	0	2.61	2.26	0	0	0	0	0	0	0	0	0	0	0	2.61	2.26
Delivered Hardware Costs	0	0	0	0	0	0	0	0	0	0.00	0.00	0	15.85	23.77	24.51	9.48	1.63	0	0	0	75.24	67.79	75.24	67.79
Subtotal - RPS	0	3.0	3.8	5.2	8.9	10.3	0	0	0	31.21	27.12	0	15.85	23.77	24.51	9.48	1.63	0	0	0	75.24	67.79	106.46	94.91
Total NASA Cost	1.2	20.30	66.87	37.98	39.75	15.29	0	0	0	181.39	161.74	0.00	22.10	34.68	165.57	204.91	103.67	19.78	9.28	7.48	567.48	490.13	748.86	651.87

Figure 23. Full breakdown of Yamato mission costs. The LEXco systems engineering process employs cost engineering at every phase of the design process to insure value to the customer

Table VIII. Detailed customer and lower level requirements based on the NASA AO

NASA AO Requirements and Derived Requirements

- **Cost**
  - Total NASA cost through mission completion shall not exceed $700M
  - The Concept Study (Phase A) shall cost no more than $1.2M in Real Year Dollars (RY$)
  - Proposed cost to NASA shall not increase more than 20% from the original proposal to the final report
  - Proposed missions shall maintain a reserve of at least 25 percent from Phase B to the end of Phase D
  - Proposed missions shall not rephrase Phase E funds to Phase C/D after confirmation of approved investigation
- **Schedule**
  - The Concept Study (Phase A) shall be no longer than 7 months
  - The launch date shall be no later than December 31, 2014
  - LSR Missions shall specify the desired launch date and indicate any flexibility
  - Launch shall take place within 47 months from start of Phase C
- **Management**
  - Proposed missions shall designate a single Principal Investigator
  - Proposed missions shall clearly define a management approach
- **Mission**
  - LSR missions shall be launched as primary payloads on ELVs
  - The design team shall consider at least the following when performing trades: performance, safety, cost, risk.
  - Proposed missions shall describe the risk mitigation policy
    - The design team shall identify the performance floor below which the investigation will not be considered
    - The design team shall present a plan for descoping in the event of cost or schedule growth
  - Proposed missions shall describe adequate backup plans for technologies with a TRL less than 7
  - Proposed missions shall have mission assurance program consistent with the ISO 9000 series
  - Proposed missions shall address their plan to comply with all planetary protection requirements
- **Science**
  - The design team shall design a LSR mission to and from the Moon's South Pole-Aitken Basin
  - The lander shall recover at least 1 kg of lunar basalts
  - The returned samples shall include both soil and diverse rock chips
    - The lander shall actively seek high thorium concentrations
    - The lander shall actively seek high FeO concentrations
    - The samples shall help differentiate between planetary bodies and Moon structure/composition
    - The samples shall help determine the effects of early impacts on the Moon's structure and dynamics
    - The samples shall help determine the depth of the impact
    - The samples shall help determine the composition and origin of the impacting object
    - The samples shall explain the nature of the Moon's lower crust and mantle
    - The samples shall help validate global, regional, and local remotely sensed data of the sampled site
    - The samples shall explain the sources of thorium and other heat-producing elements
    - The samples shall determine the ages and compositions of far-side basalts
- **Returned Samples**
  - The samples returned shall be delivered to the NASA Astromaterials Curatorial Facility at NASA JSC
  - The investigation team is responsible for the transport of the materials
  - The science team will be allocated no more than one quarter by mass of the returned sample
  - Proposed missions shall include analysis and publication of data to NASA's Planetary Data System (PDS)
- **E/PO**
  - Proposed missions shall describe the full implementation of an Education and Public Outreach (E/PO) program
    - Create unique tools that are compelling to educators and students
    - Provide opportunities for research awards and education programs for students
    - 1-2% of NASA cost (excluding LVs) will be dedicated to E/PO
    - Provide samples to museums
    - Create programming on NASA television

8.2 List of Acronyms

ABLE: Aitken Basin Lunar Excavator
ADACS: Attitude Determination and Control System
AHP: Analytical Hierarchy Process
AO: Announcement of Opportunity
C3: Launch Energy
CAD: Computer Aided Design
CER: Cost Estimation Relationship
CEV: Crew Exploration Vehicle
COTS: Commercial Off The Shelf
DDT&E: Design, Development, Test & Engineering
DoD: Department of Defense
DSN: Deep Space Network
EEV: Earth Entry Vehicle
EL1: Earth Lagrange Point 1
ELVPERF: Expendable Launch Vehicle Performance Tool
E/PO: Education and Public Outreach
FOM: Figure of Merit
FY: Fiscal Year
GPHS-RTG: General Purpose Heat Source Radioisotopic Thermoelectric Generators
JPL: Jet Propulsion Laboratory
JSC: Johnson Space Center
KSC: Kennedy Space Center
LAV: Lunar Ascent Vehicle
LCC: Life-Cycle Cost
LEO: Low Earth Orbit
LEXco: Lunar Excavation Company
LL2: Lunar Lagrange Point 2
LOI: Lunar Orbit Insertion
LV : Launch Vehicle
LRO: Lunar Reconnaissance Orbiter
MAAF: Michael Army Airfield
MAR: Mid-Air Retrieval
MMRTG: Multi-Mission Radioisotope Thermoelectric Generator
NASA: National Air and Space Administration
PRS: Parachute Recovery System
QFD: Quality Function Deployment
RPS: Radioisotopic Power Sources
SPA: South-Pole Aitken
SRG: Stirling Radioisotopic Generator
STK: Satellite Took Kit
TLI: Trans-Lunar Injection
TOF: Time of Flight
UTTR: Utah Test and Training Range

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