Apollo Gray Team Lunar Landing Design

- Final Report -

May 16, 2007

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Acknowledgements

The Apollo Gray Team would like to thank Payload Systems Incorporated (PSI) in Cambridge for their generous support in printing a 3-D model of our final lander design; the model greatly enhanced our capability to communicate our lunar landing concept.

1. Introduction

This document represents the final report of the ESD.30/16.895 Gray Team class project. The goal of the class project was "to design a lunar landing". The Gray Team supplemented this high-level goal with additional, more detailed, objectives:

- To design a near-term, feasible, affordable, and safe mission architecture (including design of the lander vehicle, the descent trajectory, and mission operations) that is compatible with NASA's current lunar exploration strategy as outlined by the Lunar Architecture Team (LAT) at the 2nd Exploration Conference, December 4-6, 2006, in Houston.
- To carry out detailed analyses of the GN&C, human factors, and operations areas of the overall mission architecture to create design solutions and assess their feasibility
- To analyze ways to make the lunar landing more capable through use of innovative design, technology or operational choices
- To provide systematic comparisons of all aspects of the Gray Team landing design to Apollo in order to understand similarities and differences and assess their impacts

The high-level goal and these detailed objectives summarize the philosophy that the Gray Team followed throughout their design; the philosophy is reflected in all analyses presented throughout this document.

The objective of being compatible with NASA's lunar strategy as outlined by LAT specifically requires analysis of and design for three individual scenarios:

- Scenario 1: transport of crew and cargo to a lunar outpost, most likely located at one of the lunar poles (South pole is the current reference location)
- Scenario 2: delivery of a large amount of cargo to an outpost location without crew. This use case extends the problem significantly because it requires automatic lunar landing capability.
- Scenario 3: transport of crew and cargo to an unprepared lunar surface site anywhere on the Moon for a mission of exploration (sortie mission, much like the Apollo J-class missions)

While a high-level analysis of the overall mission architecture was necessary to provide context for the lunar landing phase, detailed design of the entire lunar landing mission architecture was clearly beyond the scope of the project. The Gray Team therefore decided to limit the in-depth analysis of the lunar landing to all mission events occurring after separation from other mission assets in a 100 km lunar staging orbit through landing and safing on the surface. Specifically, the following events were included:

- Insertion into a descent orbit (orbit that has a low pericenter located in the vicinity of the landing site)

- Coast in the descent orbit
- Powered descent down to low altitude and associated re-designation
- Final landing, hovering, and associated re-designation
- Touchdown and safing

The report is organized with an introduction first, which has provided context for the project and an overview of the report. The introduction is followed by four sections which outline the thought process and major insights and results from the four subteams: systems architecture, GN&C, human factors, and operations. The subteam sections are followed by conclusions and an annotated bibliography. Detailed results and work that was not included in the subteam sections are provided in the appendices.

2. Systems Architecture

The goal of systems architecture activities in designing the Gray Team lunar landing was to carry out an analysis of lunar landing concepts and select a reference concept for more detailed analysis with regard to lander design, GN&C, human factors, and detailed operations. The architecture team provided overall mass properties for this reference concept. In addition, the architecture team provided a more detailed geometrical lunar lander design and associated visualization, and worked closely with other teams on the design of the reference trajectory and the nominal operations plan.

2.1 Review of Lunar Landing Concepts

In the Apollo era, lunar landing was a novelty which had never been seriously analyzed before, let alone been attempted. In our time, lunar landing has been accomplished a number of times, and a plethora of concepts for lunar landing have been proposed over time (e.g. for Apollo or the 1989 Space Exploration Initiative). The Gray architecture team therefore decided to focus on a systematic review of a number of proposed lunar landing architectures that provide a representative sample of the architectural space. Specifically, the following nine concepts were selected (more detailed descriptions of the individual concepts are provided in Appendix 8.1.1):

- The Apollo LM concept
- The Soviet lunar lander concept
- NASA's 1992 First Lunar Outpost (FLO) crew transportation system concept
- NASA's 1993 Lunox crew transportation concept (innovative in that it uses in-situ propellant production)
- The 2005 NASA Exploration Systems Architecture Study lunar lander concept
- The 2006 NASA Marshall Space Flight Center lander concept
- A 2006 Lockheed Martin lunar lander concept
- A 2006/07 MIT concept utilizing the Ares V upper stage for lunar orbit insertion
- The December 2006 lander concept of the NASA Lunar Architecture Team (LAT)

These nine concepts were studied in detail with regard to the mission mode employed (for description of the different mission modes see Appendix 8.1.2), the assignment of propulsive maneuvers and habitation functionality to lander elements, and the overall lander geometrical layout. Based on this analysis they were then mapped out in a Morphological Matrix (Table 1), which is a product development tool that allows for analysis of existing and synthesis of new concepts based on design variables. For each design variable (shaded left-most column) an assignment in the corresponding row is chosen, thereby creating a path through the matrix. The full matrix with all nine concepts outlined is provided in Appendix 8.1.3, Table 1 shows the Apollo concept and the "EDS for LOI" concept.

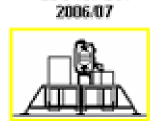
Table 1. Gray Team reference lander architecture in comparison to Apollo

Morphological matrix of lunar landing (lander and mission design):

Mission mode	LOR	EOR/LOR	Direct	EOR
Crew on surface	Entire crew	Crew left in orbit		
Lunar orbit insertion (LOI)	Stage 1	Stage 2	CEV / CSM	EDS
DOI + initial descent	Stage 1	Stage 2		
Final landing	Stage 1	Stage 2		
Lunar ascent	Stage 1	Stage 2		
Lander orientation	Vertical	Horizontal	Hybrid	
Crew compartment 1 usage	Descent + surface stay + ascent	Descent + ascent	Earth-Moon + descent + ascent + Moon-Earth	
Crew compartment 2 usage	No 2nd crew compartment	Surface stay	Airlock	Airlock + surface stay
Airlock	No airlock	4-person airlock	2-person airlock	Suit-lock
Lander stage 1 propellants	N_2O_4 / Aerozine-50	LOX / LCH_4	N_2O_4 / MMH	LOX / LH_2
Lander stage 2 propellants	N_2O_4 / Aerozine-50	LOX / LCH_4	N_2O_4 / MMH	LOX / LH_2

Apollo LM

EDS for LOI
2006/07



The nine concepts were then evaluated (ranked) with regard to a number of metrics (proximate metrics for development & operational cost, and development & operational risk) for carrying out transport of four crew and six megatons of cargo to a lunar polar outpost (LAT use case 1). Specifically, the metrics used for evaluation were (see Table 2):

- Mission and launch mode required to accomplish the mission (operational cost & risk)
- The number of different crew compartments required (development & operational cost, development risk)
- The number of different lander & CEV propulsion stages required (development & operational cost, development risk)
- The number of rendezvous and docking operations required (operational risk)
- The degree of difficulty of balancing cargo for all use cases outlined by LAT (see above)
- The degree of difficulty for crew egress and cargo unloading on the lunar surface
- Whether In-Situ Resource Utilization for propellant production is required (development and operational risk)

Table 2. Evaluation of lunar lander concepts for carrying out a crew and cargo mission to a lunar polar outpost (4 crew and 6 mt of cargo); ranking order (worst to best: red, yellow, light green, dark green)

Concept	Mission & launch mode	Crew compartments	# of propulsion stages	# of rendezvous and dockings	Cargo balancing	Crew egress and cargo offloading	ISRU required?
Apollo LM	2 Ares V, EOR/LOR	CEV + ascent	3	2	Easy	Medium	No
Soviet lander	2 Ares V, EOR/LOR	CEV + ascent	3	2	Difficult	Medium	No
NASA FLO	2 Ares V, EOR	CEV	2	1	Easy	Difficult	No
NASA Lunox	1 Ares V, Direct	CEV	1	0	Hard	Easy	Yes
NASA ESAS	2 Ares V, EOR/LOR	CEV + ascent	3	2	Easy	Difficult	No
NASA MSFC 06	Ares V + Ares I, EOR/LOR (Ares V + LOR possible)	CEV + ascent	3	2 (1 possible)	Easy	Medium	No
Lockheed 06	Ares V + Ares I, EOR/LOR (Ares V + LOR possible)	CEV + ascent	3	2 (1 possible)	Hard	Easy	No
EDS for LOI	Ares V + Ares I, EOR/LOR (Ares V + LOR possible)	CEV + ascent	3	2 (1 possible)	Easy	Medium	No
NASA LAT	Ares V + Ares I, EOR/LOR (Ares V + LOR possible)	CEV + ascent	3	2 (1 possible)	Easy	Medium	No

Based on the results of this evaluation (shown in Table 2), the EDS for LOI concept was chosen for the following reasons:

- It outperforms the Apollo LM, Soviet lander, and NASA ESAS concepts in all metrics
- The two concepts which bring the CEV to the lunar surface (FLO and Lunox) both have advantages in certain areas, but disadvantages in others:
    - Lunox requires ISRU propellant production on the lunar surface. This removes abort-to-orbit options for the landing after a certain threshold; this was deemed to risky and the concept therefore discarded.
    - The main advantages of FLO are the reduced number of rendezvous & docking operations, the use of only one type of launch vehicle (Ares V), and the need to design and produce only one crew compartment (the CEV CM). However, the CEV Block I (without lunar surface capability) is currently under development; it would therefore be quite costly to change to an architecture with the CEV going to the surface. Having an extra crew compartment for the lunar surface excursion decouples the CEV development from the lunar mission architecture (not unlike the LM did in the Apollo program).
- The NASA MSFC and NASA LAT concepts were discarded because they have the ascent stage off the centerline of the lander; this creates additional design and/or operational complexity due to the need to provide a docking adapter for the CEV on the lander centerline.
- The Lockheed concept with its horizontal configuration offers advantages in terms of cargo offloading, but introduces challenges with regard to balancing during descent for the different LAT use cases. It was therefore discarded.

The following section provides a more detailed description of the reference lander configuration.

2.2 Reference Lunar Lander Design

The lunar landing system architecture is overall very similar to the system used by Apollo and the system that NASA proposed in ESAS, but a few key differences must be noted. These key differences and reasoning behind them will be described in this section.

First, the Gray Team design descent stage is much shorter than the current ESAS design in order to make it easier for the astronauts to access the moon surface. This is made possible due to the use of CEV engines rather than LSAM engines in order to perform LOI. Therefore, the height of the descent stage is only 2.5m. In order to facilitate access to the surface, a system of two ladders is used. One short (1.6m) vertical ladder facilitates access via the crew hatch on the ascent stage to the top of the descent stage. A second (3.7m) ladder is placed along one of the landing legs at an angle of 40 degrees in order to provide access from the top of the descent stage to the lunar surface. The ladder interface can be seen in Figure 1.

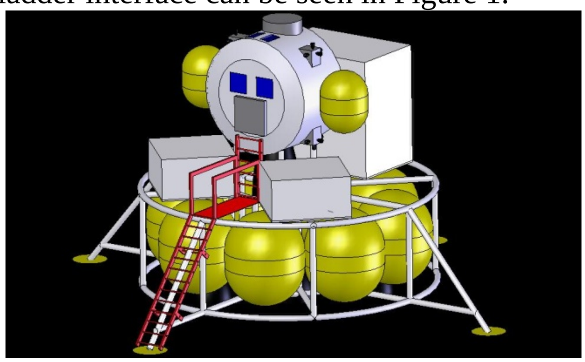


Figure 1. Ladder

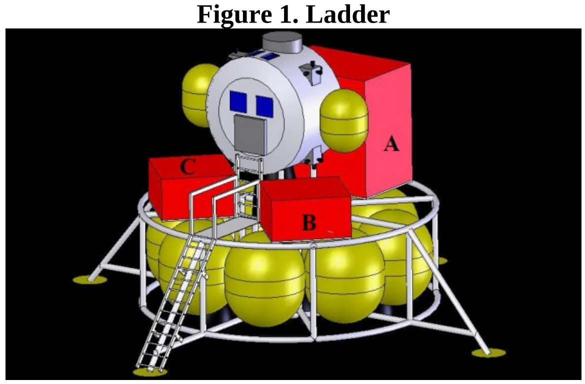


Figure 2. Cargo

In the Gray Team design, cargo is placed in one of three payload modules on the top of the descent stage. Payload Module A is placed behind (opposite the access hatch and window) the ascent stage so that the payload will not obstruct any view. Payload modules B and C are placed in front and to the sides of the ascent stage and flank the pathway between the two ladders that were described above. The payload modules are labeled in Figure 2.

The propulsion system consists of eight fuel tanks of the same design that carry both the liquid hydrogen and oxygen for the three RL-10 engines. As compared to ESAS, the Gray Team design requires only three engines. The motors are mounted to a plate at the bottom of the descent stage truss in order to provide support and mount points for the engines and auxiliary engine equipment, as seen in Figure 3.

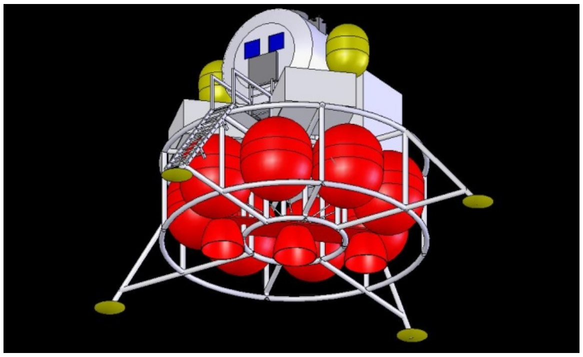


Figure 3. Propulsion System

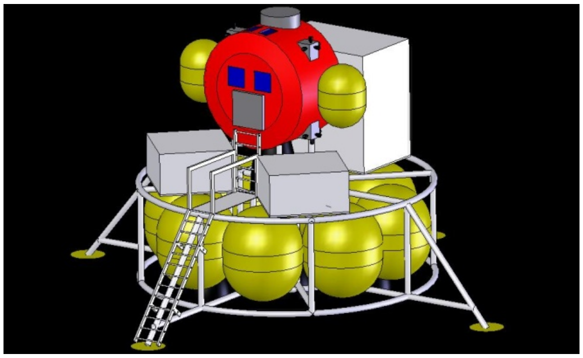


Figure 4. Ascent Stage Structure

The docking interface is on the top of the ascent stage in a location such that the docking ring is concentric with the centerline of the entire LSAM, for stability reasons. In order to facilitate the docking procedure, one window is placed in front of the docking ring such that both the commander and pilot can observe the docking procedure. The structure of the ascent stage is essentially a tube placed on its side, with the top of the cylindrical tube facing the ladder system, as seen in Figure 4. The tube is then chamfered on the ends. This structure is used since it is a standard module shape that can be easily and cheaply manufactured. Four sets of four RCS thrusters are then placed along the midline on either side of the ascent stage. This is shown in Figure 4.

The structure of the descent stage is a series of trusses that are arranged in a series of rings. Both the top and the bottom of the truss contain two concentric rings. Four horizontal struts are used to connect the inner to outer ring. Landing strut supports are placed angularly between the horizontal struts. Vertical strut supports are placed in 16 locations connecting respective locations between the horizontal rings. This is depicted in Figure 5.

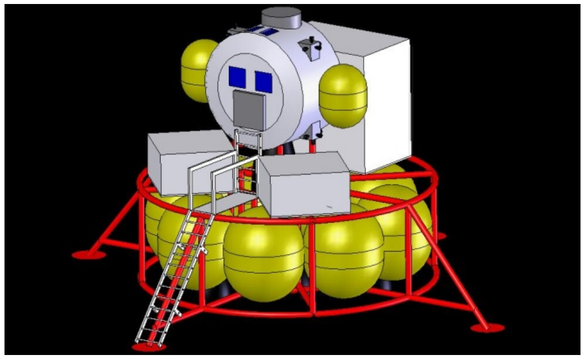


Figure 5. Descent Stage Structure

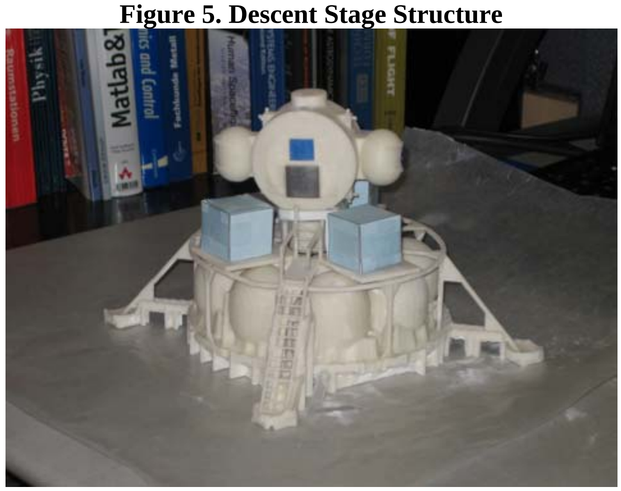


Figure 6. 3D Printout of Lander

Figure 23 and Figure 24, in Appendix 8.1.4, show the reference lander configuration in direct comparison to other configurations. A small-scale 3-dimensional of the printout of the reference lander was prepared to verify the concept and enhance inter-team communications. A photograph of the printout is shown in Figure 6.

As mentioned above, the reference lunar lander concept has to support the three different use cases required by the NASA ESAS and the LAT-1 campaigns: transport of crew and cargo to a lunar outpost (i.e. to a site with previously habitation infrastructure available), delivery of only cargo to a lunar outpost (uncrewed mission), and transportation of crew and cargo to a sortie site (unprepared site with no pre-deployed assets available). Figure 7 provides an overview of the lander configurations for these use cases:

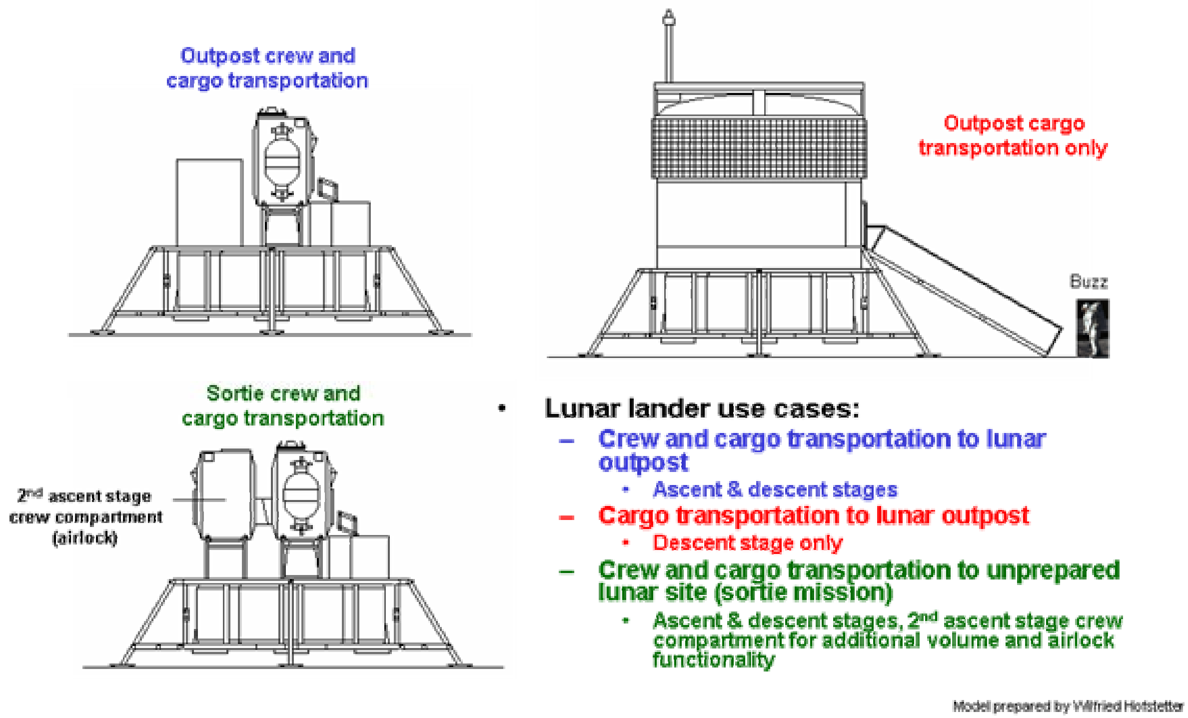


Figure 7. Gray Team reference lander design configurations for different use cases

For crewed outpost missions, the configuration outlined in Figures 1-6 is used; the ascent stage is used for crew habitation. For uncrewed outpost cargo transportation, only the descent stage is used (with added GN&C and avionics capability for automatic landing). The sortie mission is based on the crewed outpost mission configuration, but with an additional ascent stage crew compartment for extended pressurized volume for the crew; the 2nd compartment could also be used as an airlock if so desired. Note: the astronaut is shown to scale to emphasize that the top of the descent stage is close to the ground.

2.3 Systems Architecture Summary

In summary, the reference lunar lander concept chosen by the Gray Team features a number of similarities with both the ESAS lander and the Apollo LM (Table 3, lander sizes are to scale):

- Lunar Orbit Rendezvous (LOR) is used in order to decrease the overall injected mass requirements by leaving the Earth return propulsion and entry crew compartment in lunar orbit; for both the ESAS and the Gray Team lander Earth Orbit Rendezvous (EOR) was chosen to increase the mass that could be injected towards the Moon and allow for launch of the crew on the same vehicle as used for missions to the ISS.
- All three concepts utilize a ~100 km Low Lunar Orbit for staging in lunar vicinity
- All three concepts have a clear split of functionality with one module serving as an ascent stage and another module providing propulsion for descent and landing. This causes significant operational commonality between these designs, in nominal as well as contingency operations.
- The Apollo LM and the Gray Team lander designs are both exclusively used for descent to the surface, the surface stay, and the ascent.

- The ESAS lander and the Apollo Gray team design both utilize LH2/LOX propulsion for all maneuvers prior to descent, and in both cases the entire crew goes to the lunar surface.

Table 3. System architecture comparison between the Apollo, ESAS, and Gray Team landing concepts



Category	Apollo	ESAS	Gray Team Design
Mission mode	Single launch / LOR	EOR / LOR	EOR / LOR
Crew members	2 on lunar surface, 1 in lunar orbit	4 on lunar surface, none in orbit	4 on lunar surface, none in orbit
Surface stay	1-3 days	7 days	3-7 days (7 days with second compartment)
Lunar orbit insertion (LOI)	With SM engine	With descent stage	With Earth Departure Stage
Lunar staging orbit	LLO, 110 km	LLO, 100 km	LLO, 100 km
Lander maneuvers	DOI & descent & ascent	LOI & descent & ascent	DOI & descent & ascent
Lander stages	2 stages, separation at lift-off from lunar surface	2 stages, separation at lift-off from lunar surface	2 stages, separation at lift-off from lunar surface
LOI propulsion	N_2O_4 / Aerozine-50	LOX / LOH_2	LOX / LOH_2
Descent propulsion	N_2O_4 / Aerozine-50	LOX / LOH_2	LOX / LOH_2
Ascent propulsion	N_2O_4 / Aerozine-50	LOX / LCH_4	N_2O_4 / MMH
Lander airlock	None	4-person airlock	None / 2nd crew compartment

The Gray Team lander design is also different in many respects:

- It utilizes the Ares V upper stage (Earth Departure Stage or EDS) for lunar orbit capture
- It utilizes the space shuttle N2O4/MMH propellant combination for ascent (this enables utilization of the shuttle OME and the shuttle RCS thrusters)
- It can be utilized in different configurations with and without an airlock
- Depending on the configuration, it can provide lunar surface stay capabilities ranging from 3-7 days, thereby bridging the Apollo and ESAS durations

While not mentioned in Table 3, it should be noted that the Gray Team design also provides the option to conduct single-launch lunar cargo only missions (i.e. LOR missions like Apollo) utilizing the Ares V launch vehicle only; this could potentially enable significant reductions in operational cost and risk once an outpost is established.

Overall, the Gray Team architecture is similar to Apollo in many respects, mainly because the physics of propulsion and orbital mechanics are invariable. Some new technologies such as LH2/LOX propulsion lead to higher performance, while new operational constraints such as EOR/LOR and the three use cases mentioned above drive the design to more capability and flexibility.

3. Guidance, Navigation, and Control (GNC)

The GNC subteam is responsible for the guidance, navigation, and control of the spacecraft. The design includes a baseline fully automatic mode to support the proposed cargo missions and a manual intervention mode for crewed missions to increase reliability and safety. The scope of the GNC subteam was to define a trajectory, design the control architecture, identify hardware candidates for sensors and actuators, and combine the previous three areas into a simulation to predict GNC performance. A GNC goal is to provide global landing capability with specific access to the South Pole, the proposed location of the lunar base in NASA exploration plans as of December 2006.

3.1 Trajectory

The trajectory is designed to take the lander from the lunar parking orbit to the surface of the moon. Major design considerations include: minimizing fuel usage, variability of terrain, visibility, and abort contingencies. The trajectory is divided into three phases: lunar orbit phase, transfer orbit phase, and powered descent phase. The lunar orbit phase is a 100 km circular parking orbit. The transfer orbit phase uses a 75 ft/s Hohmann transfer to enter an elliptical orbit with a periapsis of 15.24 km. The final, powered descent phase is the most critical phase, and is thus the discussion of the remainder of this section.

The powered descent phase begins at the periapsis of the elliptical orbit at an altitude of 15.24 km. This altitude was chosen as a compromise between effects of initializing powered decent at an altitude that is either too low or too high. The PDI altitude should be low to minimize gravitational losses. However, if the altitude is too low, the high thrust to weight ratio would cause the lander to crash. Therefore, the 15.24 km initialization altitude compromises between the two adverse effects and provides good performance.

The powered descent trajectory consists of three phases: two gravity turns[1] and a final hover phase. The first gravity turn begins at PDI at 15.24 km, has a throttling ratio of 0.8, and ends with an altitude of 11.9 km. The second gravity turn has a throttling ratio of 0.23 and continues until the lander is at an altitude of 100 m. The hover phase begins at the 100 m altitude with a velocity of 1.23 m/s. The vertical and horizontal velocities are nulled, and there is ample amount of remaining propellant (110 seconds of hover time) in order to land in a desirable location. The entire powered descent trajectory is shown in Figure 8. Then, Figure 9 shows the end of the trajectory, so the hover phase is visible. The Apollo 12 trajectory is also plotted. The comparison with the Apollo trajectory highlights the difference in the two trajectories; our trajectory is much steeper. The steepness provides greater fuel efficiency due to fewer gravitational losses as compared with Apollo. This is possible because we do not need to pitch-up early for visibility, as was necessary for Apollo. Visibility is a major driver of the landing trajectory design. Pitching up early can give the crew out-of-the-window visibility of the landing site, but results in a large mass penalty due to inefficiencies. Our design makes use of external cameras to visualize the landing site, which eliminates the need for an early pitch over. More details on the camera and visibility design are given in the Human Factors section of the report.

[1] A gravity turn requires the thrust to remain parallel to the velocity vector. These maneuvers are extremely fuel-efficient.

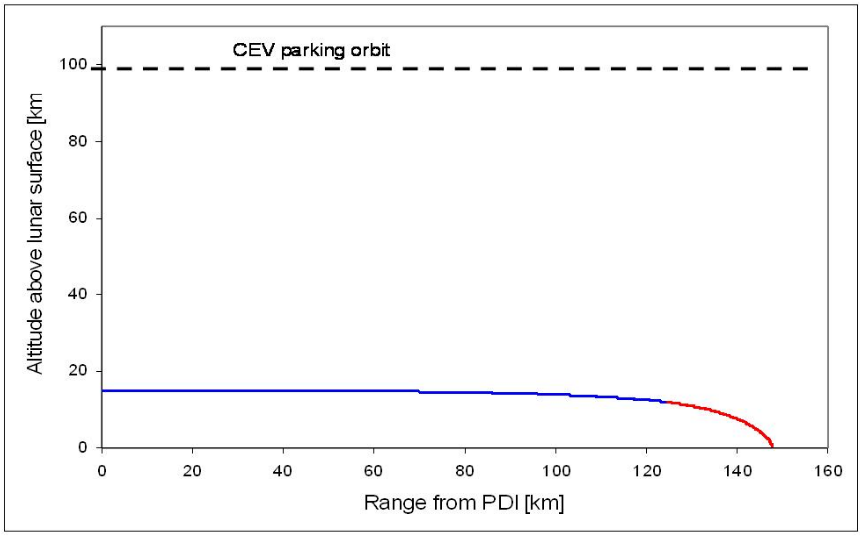


Figure 8. Descent trajectory: Attitude versus Range

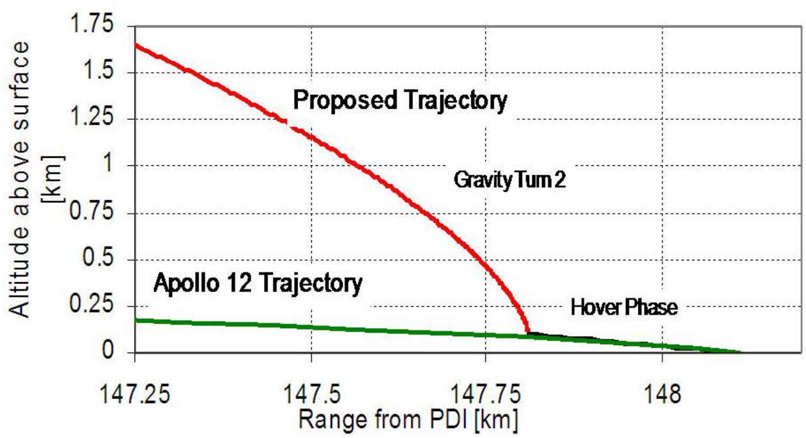


Figure 9. Final Trajectory Phase: Altitude versus Range

Another important point on the trajectory is the critical-descent altitude of about 10 m. This is the minimum altitude where the lander can still abort with the ascent stage. Below this altitude, the engines do not have necessary time to reach the thrust levels to ascend to a higher altitude. Although a hard landing from this height would cause significant damage to the lander, the crew would be able to survive, especially as they are equipped with either full EVA suits or rapidly-sealable pressure suits. Therefore, in the manual intervention mode, the astronaut would be instructed to remain above the critical altitude until he or she is ready to land, to reduce the risk of hard landing and concurrent loss of mission.

Table 4 summarizes the key similarities and difference of the final descent trajectory to that of Apollo. Overall, the trajectory is a much more efficient one, which is made possible largely due to the visibility decisions. Note that landing site is visible in the camera for the entire descent; at 5.3 km, the resolution becomes sufficient to allow re-designation.

Table 4. Trajectory Comparison with Apollo

	Apollo	Gray Team Design
Number of trajectory phases	3	3 (2 for cargo)
Total Delta V (m/s)	2150	1900
Descent profile	Shallow	Steep
PDI initialization height (km)	15.25	15
Pitch up for visibility	Yes	No
Altitude where landing site is visible (km)	2.7	5.3 (camera), 0.3 (window)
Altitude of final landing stage initialization (m)	152	100
Hover capability	Yes	Yes

3.2 Hardware

The GNC sensor suites and actuators must also be chosen. Our goal is to utilize hardware that enables completely autonomous navigation. However, during normal operations, ground-tracking updates would also be utilized.

3.2.1 Sensors

We performed trade studies of various types of navigation sensors. Consistent with the desire for autonomous navigation, the main navigation unit must be an on-board sensor which continuously tracks the spacecraft's position and velocity. Inertial measurement units (IMUs) provide such a capability, and contain three orthogonal accelerometers and three orthogonal gyroscopes. These devices have good accuracy and reliability, but must be integrated, so errors build over time. Therefore, it is also necessary to update the IMU; star trackers provide the necessary updates to account for the drift. Sun sensors could also provide such a capability, but are much less accurate than the star trackers, so were not chosen. Additionally, ground tracking via the Deep Space Network (DSN) will be used. DSN provides ~1 m position accuracy and 1 mm/s velocity accuracy at Neptune; we can expect better performance due to the proximity of operations. A third update option is to have a ground beacon. A beacon would improve the measurement accuracy. However, sortie missions and initial missions would not have such a beacon, and we determined that while a ground beacon would be useful and could be included in later missions, it should not be part of the primary GNC architecture.

In addition to the IMU and ground tracking, it is also desirable to have a ground-truth measurement. Altimeters can take multiple forms; two promising types are radar and LIDAR. Radar is proven technology, and has no problems with dust. LIDAR can provide better accuracy, especially at higher altitudes. However, the reflectivity of the lunar regolith can cause a decrease in accuracy of the LIDAR in comparison to radar. Therefore, the proven radar technology is chosen, and existing LIDAR maps are utilized in the computer algorithms.

The on-board GNC baseline sensor suite consists of an IMU, landing radar, and star trackers. We researched each type of sensor to identify individual components for the mission. The Honeywell MIMU and LN200 are two high performance IMUs that are space-rated. The comparison between the two models is summarized in Table 11. The Honeywell MIMU is chosen due to its superior performance, despite its higher mass; since this is the primary navigation sensor, accuracy is the priority. A wide range of star trackers were also examined. A comparison of seven star trackers and one sun sensor are compared in Table 12. The SED26 is selected for high accuracy and large field of view, allowing for more robust utilization. Two aircraft and space-rated landing radars were considered, and the comparison can be seen in Table 13. The HG9550 is chosen due to its higher accuracy. As it will be shown in the simulation results, the higher landing radar accuracy can greatly improve landing accuracy.

3.2.2 Actuators

The descent engine plays a critical role in the landing, so its selection is vastly important to the design. A modified RL-10 engine was selected because of its heritage and NASA's current plans to use such an engine. The propellants of liquid oxygen and hydrogen provide increased specific impulse and much lower mass than other bipropellant systems, including the propellants used on Apollo. Two models of the RL-10 were considered: the RL-10-B2, and the RL-10-A4-2. As can be seen in Table 14, they offer similar performance but the RL-10-A4 is significantly smaller in size. The shortened length of the A4 allows our lander to sit lower to the ground to allow better cargo off-loading capability as well as to accommodate a shorter ladder for the astronauts. There are known reliability issues with the RL-10, but they are expected to be remedied by the time of the mission.

For roll maneuvers and fine attitude control, we determined that reaction jets are the preferred method to produce our required angular velocity (estimated to be approximately 1 deg/sec) with the necessary precision and speed. Reaction wheels and control moment gyros are too slow and more massive. For the RCS thrusters, a number of models were considered (compared in Table 15). The RS-28 was selected for its high thrust capability and its heritage on the Space Shuttle.

During the landing, the descent engine will be gimbaled to ensure the thrust vector goes through the center of mass. Given this capability, the engine can also be used for pitch and yaw control. This is desirable since the decent engine is more efficient than the RCS thrusters. The RCS thrusters are pulsed fast enough to provide fine-tuned attitude control.

3.2.3 Apollo Hardware Comparison

The Apollo LM GNC hardware included an IMU, landing radar, RCS thrusters, and a descent engine. As shown in Table 5, our hardware design offers considerable improvements over Apollo in terms of capability and especially in terms of size. This improved capacity will aid in making our landing more accurate and efficient.

Table 5. Selected comparisons between Apollo and the Gray Team hardware

	Apollo	Gray Team
IMU		MIMU
Accelerometer Bias (μ-g)	200	100
Gyro Drift (deg/hr)	0.08	0.05
Size (in.)	12 dia. (sphere)	9.17 dia. x 6.65 (cylinder)
Weight (lb)	60.2	9
Landing Radar		HG9550
Vertical accuracy	4%	2%
Weight (lb)	42	9.75
Electronics Size (in)	15.75x6.75x7.38	3.5x6.3x8.75
Power (W)	132	35
Startracker		SED26
Pitch, yaw accuracy	N/A	3 arcsec
Roll accuracy	N/A	15 arcsec
RCS Thrusters		RS-28

Propellants	N2O4/UDMH	N2O4/MMH
Specific Impulse (sec)	290	295
Thrust (N)	445	2667
Descent Engine		RL-10-A4-2
Propellants	N2O4/Aerozine 50	LOX/LH2
Specific Impulse (sec)	311	449
Thrust (kN)	45	99

3.3 Control and Estimation

3.3.1 Control Architecture and Comparison to Apollo

The baseline control architecture consists of a minimum-time / minimum-fuel LQR controller and an Extended Kalman filter. This differs significantly from Apollo, especially in the controller. Apollo used a non-linear 3rd order minimum time controller, selected because of the limited computational power. Minimizing the time for execution minimized computation time as well, leading to better performance. An LQR controller is optimal, leading to significant performance improvements over Apollo. The basic Kalman estimator is the same from Apollo. However, it has been improved to the Extended Kalman filter and includes non-linear states. By including the ability to propagate non-linear states, this enables incorporation of a more robust and more accurate dynamics model. The state vector consists of the position, velocity, altitude, attitude (expressed in quaternions), angular rate, mass, and inertia. The ability to propagate non-linear states allows for the inclusion of the mass and inertia in the state vector. These states are continuously updated to account for the expulsion of propellant. Since the dynamics model accounts for the varying mass and inertia, the optimal gains calculated are also based on the mass and inertia at the moment of actuation. This leads to better performance over Apollo, where the gains were pre-scheduled to account for the varying mass. Table 6 shows a comparison between Apollo and our control architectures.

Table 6. Comparison of Control Architecture design between Apollo and the Gray team

	Apollo	Gray Team Design
Controller	Non-linear 3rd order min-time	Min-time/Min-fuel LQR
Estimator	Kalman Filter	Extended Kalman filter (EKF)
Propagate	Gain Scheduling	Mass/Inertia as states in model

3.3.2 Control Architecture Flow

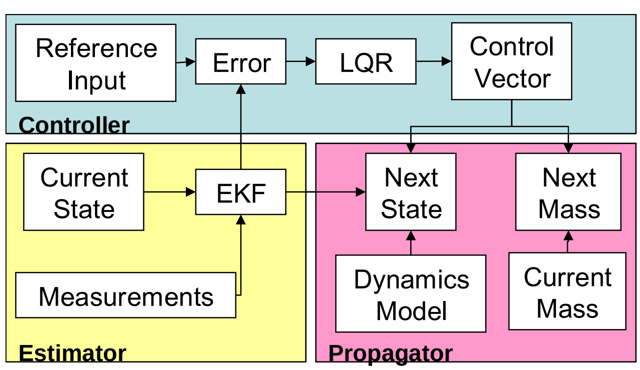


Figure 10. Block Diagram of Control Architecture

The basic flow of the control architecture is given in Figure 10. The reference input is differenced with the current state (as estimated using sensor measurements) to obtain error. Gains are applied to the error to obtain the control vector. The control vector is then converted to thruster firing times (not shown in Figure 10). On the estimation side, the Kalman filter generates an estimate at each time step using the sensor measurements and the predicted state. The estimated state and the control vector are propagated to obtain the state at the next time step. This propagation includes the calculation of the mass at each time step (shown separately for emphasis).

3.4 Simulation and Results

The aforementioned trajectory, hardware, and control system are combined into a simulation to obtain a quantitative analysis of the overall GNC subsystem. The purposes of the simulation include: comparing hardware options, ensuring control system performance, and performing a landing accuracy analysis.

3.4.1 Simulation Formulation

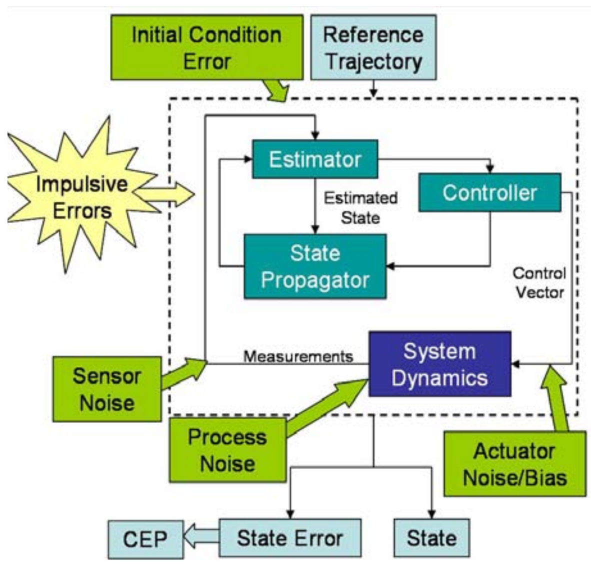


Figure 11. Simulation Overview

The simulation is a discrete-time, state-space model. A summary of the overall simulation structure can be seen in Figure 11. The inputs are the desired trajectory and various sources of noise and error, while the outputs are the state vector and state vector error as functions of time. The interior loop simulates both the computer (controller, estimator, state propagator) and the spacecraft dynamics. The computer takes in the noisy sensor measurements, estimates the current state, determines the control vector, and propagates the state vector. The control vector (thruster firings) is input to the spacecraft, which then outputs the sensor measurements at the end of the time-step. There are multiple opportunities for noise to enter the system; the noises included in the simulation are: initial condition error, actuator noise/bias, sensor noise/bias, process noise, and impulsive errors. The initial condition error arises when the PDI burn begins at the incorrect location. The sensor and actuator noise and biases are due to installation and hardware noise, and are included as Gaussian random variables with statistics based on the hardware specifications. The process noise includes other errors such as map error, gravitational effects, and computer error. The impulsive errors account for events such as stuck thrusters. The simulation accounts for the major aspects of the landing GNC.

3.4.2 Simulation Control Architecture

The control architecture modeled in the simulation is a Proportional Derivative (PD) controller with a Kalman filter. The state vector is simplified to be position only, and linearized about the current point. A PD controller simulates the behavior when the crew manually intervenes, since a PD controller is the maximum a human can enable. Only position control is considered because it has significant target change due to the trajectory. It is assumed that the attitude is maintained about the initial attitude, only damping out perturbations for the majority of the landing duration. The spacecraft was ideally modeled in the simulation with the following assumptions: rigid-body, holonomic motion, discrete time linearization, and sufficient attitude control. Constraints on maximum fuel, maximum thrust, and varying mass of the spacecraft based on propellant used are included in the model.

3.4.2 Simulation Results

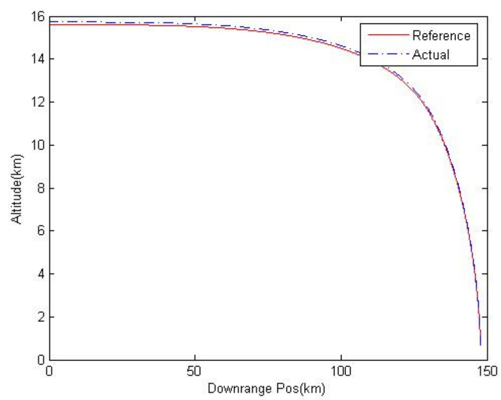


Figure 12. GNC Simulation Results

Figure 12 shows the trajectory from the simulation. The red line is the reference input trajectory. The blue is the actual trajectory. As visible in the figure, the actual system follows the trajectory extremely well. A close-up view of a small portion of the trajectory, seen in Figure 13, shows the discrepancy between the desired and actual trajectory. The left plot shows the errors when there is a large sensor noise, and the right plot shows the error resulting from a strong sensor bias. In both cases, the error is within the acceptable error range. The acceptable error range is considered to be ±50 m, which is the horizontal knowledge of the lunar terrain map.

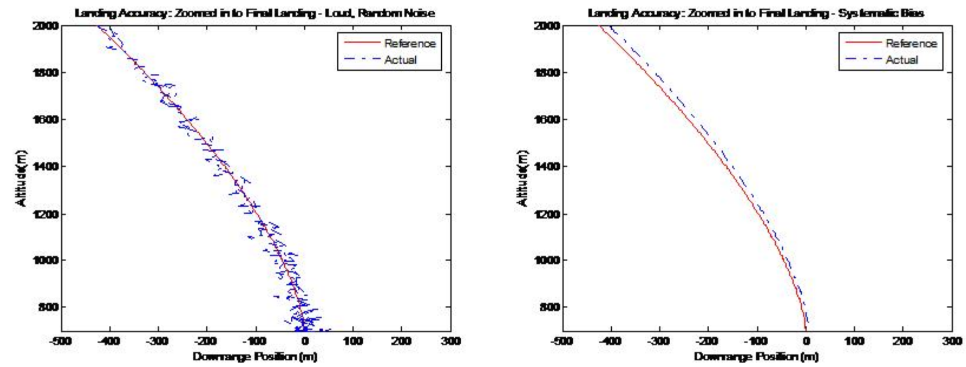
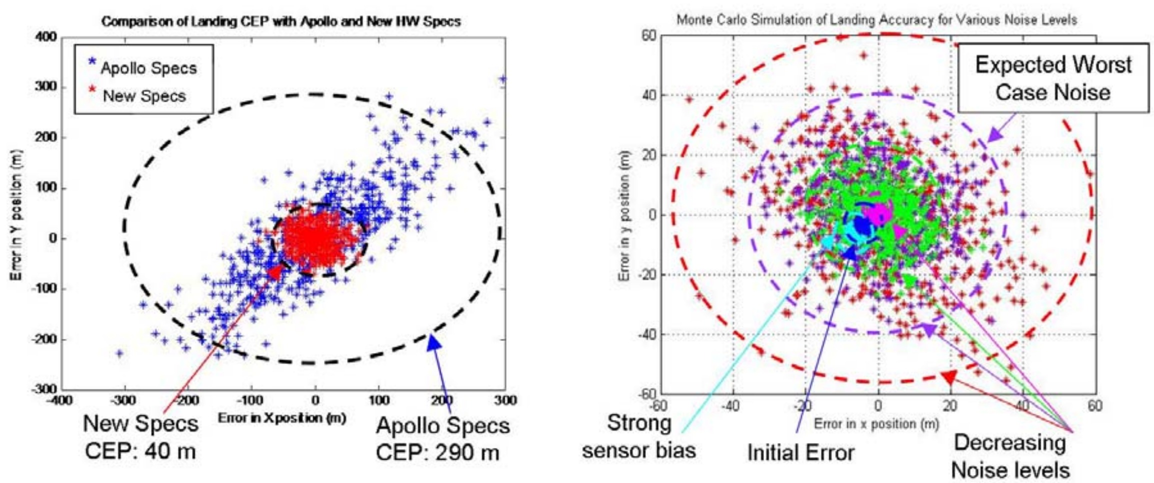


Figure 13. Trajectory Simulation Close-up: left is strong random noise, right is strong bias

3.4.3 Monte Carlo Analysis and Results

A Monte Carlo simulation is run with the randomly distributed noise inputs. The landing position error is recorded for each simulation, and plotted to obtain a circular-error-probability (CEP) of the landing accuracy. The CEP is defined as the radius of the circle that encloses the region where the where the system will be 99% of the time.



(a) (b)

Figure 14. Landing CEP, (a) comparison with Apollo, (b) various noise levels

Figure 14.a shows an example of two landing-CEPs. The smaller, red CEP is what we expect to see with the chosen configuration. The larger, blue CEP is found using the Apollo hardware specifications. As visible in the figure, the new landing CEP is about one seventh of the CEP from Apollo. The comparison is not completely accurate, as the Apollo CEP is obtained using our trajectory; only the hardware is changed. Also, the simulation is for the automatic landing and does not account for the manual intervention in Apollo. However, in a comparison with the actual Apollo landing accuracies (Table 7), the 290 m accuracy is in the correct range, giving confidence in the accuracy of the simulation.

Figure 14.b shows CEPs for various noise levels. As expected, decreasing the noise-level results in better landing accuracy; strong initial errors or biases result in CEPs that are off-set from center. The expected worst case CEP, based on the chosen hardware, is the purple line of 40 m in Figure 14.b. This is within the accuracy of the maps obtained from Lunar Reconnaissance Orbiter, Lunar Prospector, and Clementine, and thus is acceptable for the automatic landing capability.

Table 7. Apollo Landing Accuracy Comparison

Mission	Apollo 11	Apollo 12	Apollo 14	Apollo 15	Apollo 16	Apollo 17
Landing Accuracy (m)	6440	163	18	600	230	200

3.5 GNC Summary and Conclusions

The GNC subteam, collaboratively with the other subteams, determined a fuel-efficient, multi-phase trajectory for high performance and safety. Additionally, the GNC subteam did trade studies to determine appropriate GNC hardware and designed a control and estimation scheme. Finally, these three areas were combined into a quantitative discrete-time, state-space simulation. Many error types were included to determine landing CEPs. The simulation of a simplified system shows that the controller and hardware perform well, and result in an expected landing CEP of about 40 m, which is within the horizontal accuracy of the available lunar maps, and meets our desired capability. This performance could be further improved with manual intervention during the hover phase in crewed missions.

4. Human Factors

The Human Factors (HF) design for the lunar lander project concentrated on four main areas: lunar lander control, display design, interior design and anthropometry, and crew selection and training. Our study began with a comprehensive literature review that encompassed many areas including the Apollo program, the Space Shuttle and ISS, as well as several recent studies and papers related to designing a new lunar lander and its associated technologies. One of the results of our survey was the development of a design philosophy which served as a set of rules and guidelines for making many of the decisions related to the HF design. These guidelines are summarized as follows:

- Build on lessons learned from Apollo: make the best use of the extensive technical knowledge as well as feedback from the astronauts and engineers that were directly involved in the Apollo program.
- Take advantage of the numerous technologies developed since the Apollo era.
- Do not rely on un-proven technologies. There are many state-of-the-art technologies with promising benefits to aviation and space engineering, yet we chose to only rely on technologies with which we have significant operational experience. The added risk due to lack of experience and "unknowns" in a system is not acceptable in the high-risk and high-cost environment that is human space exploration.
- Optimize the balance between humans and automation. Computers and humans are best at performing different types of tasks, so the two's distinct strengths and weaknesses, as well as the balance between them, should always be taken into consideration when deciding how to allocate them.

Out of our four main focus areas we necessarily begin with the lunar lander control and some baseline decisions to establish our starting point. It is important to note the scope of the HF design in relation to what we are considering as our reference mission. Our reference mission is one in which the goal is to transport a crew between a lunar parking orbit and a lunar station where other infrastructure such as a habitation module is already in place. Thus, HF issues associated with a sortie-type mission (airlocks, dust contamination issues, and interior arrangement of the habitable modules), where the lander also serves as a temporary living space for the crew, were not considered, as they fall beyond our scope.

4.1. Lunar Lander Control

There are two main innovations in terms of the lunar lander control design. The first is that the nominal operating mode does not require any manual control inputs from the crew. Under normal conditions, the lander will land automatically. The second innovation is the reliance and use of the external cameras as the main tool for visualizing the outside environment. These two decisions are major changes from the Apollo design, yet they are a fundamental part of our concept and result in significant design improvements. This section will explain the rationale behind several of our decisions.

4.1.1 Design Requirements

The lunar lander must satisfy several design requirements. The two most important requirements, in terms of how they affect the control design, are that the lander must: 1) have the capability to operate in a fully autonomous mode without a crew onboard, and 2) be designed to carry a crew of four from a lunar parking orbit to the lunar surface and back. Although not strictly a design requirement, we develop our system so that communications with ground control are not essential for a successful landing. This is done, in part, to lay the groundwork for future Mars operations where the time delay associated with communications from Earth would render any design that depends on ground communications ineffective.

4.1.2 Number of Crew Members in the Control Loop

The lander is capable of transporting a crew of four. However, this does not mean that all crew members must be active in the control loop. If we design a system so that only two crew members are actively part of the control loop, this gives us the option to only take two crew members in any given future mission. Furthermore, the Apollo program has already demonstrated the feasibility of a two person crew. There have been numerous technological improvements since the 1970s which can reduce the operator workload and make their tasks simpler and safer. This leads us to our decision to only use two out of the maximum of four crew members as active elements in the control loop. We also refrain from reducing the crew to one person, as this would add unnecessary risk to the system and severely reduce its redundancy. Additionally, an assessment of the feasibility of only having a single person active in the control loop can only be accurately performed at a later stage in the design process.

4.1.3 Supervisory Control

The primary role of the crew in controlling the lander is of supervising the automation. In the nominal operating mode, the automation would automatically land the spacecraft at a predetermined spot, without any of the astronauts having to use manual control. The astronauts still retain the option of reverting to manual control in the final stages of the landing trajectory, and they also have the capability to re-designate the landing site. However, switching to manual control is considered an off-nominal procedure and would only be performed if, for some reason, the automation were not working properly, or if some unexpected situation were to arise that required human intervention. Such a strong reliance on the automation is in line with our design philosophy and is not relying on unproven technologies. In the later Apollo missions, the lunar module had an automatic landing capability (although it was never used, perhaps because in the 1970s this type of technology was still unproven). In today's world commercial airliners rely significantly on automatic landing functions when operating in very low visibility conditions. The recent growth of UAV technologies has also driven the development of many autonomous and automatic controllers, as well as increased our experience and confidence in such technology. Since autonomous control is one of the design requirements, there is no technical reason why manual control should be used as the normal operating mode for crewed missions. Here, we further improve the system's reliability by complementing the automation with human supervisory control, allowing the human operator more time to focus on the tasks for which they are more suited, such as dealing with any unusual situation by using their judgment and reasoning skills.

4.1.4 Task Areas and Crewmember Responsibilities

Three main task areas for which the crew is responsible have been identified as landing control, situational awareness (SA), and systems status monitoring. These three task areas are basic design drivers which eventually lead to our display design, as discussed in section 4.2. Landing control is the primary responsibility of the commander. The commander is in charge of making sure that the automation is performing its assigned task and that the vehicle is following the designated trajectory accurately at the right velocities within an acceptable error margin. If the automation is not working as expected, then the commander has the ability and obligation to take over using manual control and finish the final landing phases. Re-designation of the landing site is also part of the commander's responsibility. The pilot's primary task is to monitor all of the subsystems using the system status display. If there are any anomalies in the subsystems, the pilot should be the first to notice them and act accordingly by following the relevant procedures and checklists. All crew members are expected to maintain good SA at all times. Having a crew that is aware of the current state of the spacecraft and that fully understands what the automation is doing at all times is important because it improves the overall reliability of the system and reduces the likelihood of operator errors.

4.1.5. External Cameras

The crew's SA and manual control capabilities are greatly improved by the use of externally mounted cameras. Three cameras to be mounted on the lower structure of the lander will provide the crew with excellent visibility of the external environment. All three cameras are mounted on an actuated platform which gives 360° azimuth and ±90°elevation rotation capability. Under nominal conditions, when the lander is under automatic control, the cameras can be used to help the crew supervise and monitor the automation. Throughout the landing trajectory, by comparing the visible terrain features with the computer displays (see section 4.2 for details on the different displays), the cameras allow the crew to view the lunar surface and hence check that their current position matches the displayed trajectory from the computer. In the final stages of the descent trajectory, as the lander gradually descends to touchdown, a camera pointed downwards will also allow the crew to view the landing spot directly underneath the vehicle. In case of any undesirable terrain features at the landing spot, the crew can re-designate the landing to a nearby location. Basing the camera's capabilities on similar technology used in UAVs and aerial surveillance, focal distances of over 8km are possible. With the camera pointed in the direction tangent to the trajectory, at 5.3 km above the lunar surface (approximately 180 seconds before touchdown), the cameras acquire full focus of the landing site such that a vehicle-sized feature on the lunar surface is discernable, thereby allowing for landing site re-designation if necessary. This eliminates the need for an early pitch-over and leads to a steeper trajectory, which results in significant fuel and, therefore, cost savings. If the commander decides to take manual control of the lander, then the cameras will also provide the astronaut with a view of the exterior, which will be one of the tools used for controlling and navigating the lander during the final stages of the landing. Decreased reliance on traditional out-the-window views are becoming more commonplace in commercial aviation, where low-visibility conditions force pilots to rely solely on instrumentation to guide the landing, and in military aviation, where operators of remotely controlled UAVs rely extensively on external cameras for landing. Apollo astronaut John Young agrees that primary dependence on synthetic vision would be acceptable as long as there were also windows for backup purposes. Even if the cameras were to fail, and the view from the window to become obstructed due to the dust, the crew would still be able to land by using the instruments presented on the displays. There are several issues related to the use of the external cameras which still require further study. The use of infra-red or other spectra to be able to see through the lunar dust during the final seconds of the landing, or the possibility that dust would stick to the lens and deteriorate the view, are some issues that warrant further investigation.

4.2. Display Design[2]

To develop the displays for our lunar lander, we studied the displays of the Apollo lander, Space Shuttle, and the MIT-Draper Lunar Access Vehicle (LAV) (Table 16) Considering the advanced and proven technologies to date, we adopted the MIT LAV displays as a baseline to start designing our cockpit. Color selections are based on the Shuttle color code and Human Factors Engineering lecture notes as shown in Table 17, in Appendix 8.3. The necessary information for astronauts described in 4.1.4 was split into three displays: Landing Display (LD display), Situational Awareness Display (SA display), and Systems Status Display (SS display).

4.2.1 Landing Display

Figure 15 shows the landing display, based on the MIT-Draper study for the Lunar Access Vehicle (LAV). On the right-hand-side of the screen is a vertical altitude and velocity indicator (VAVI), which displays the current altitude (in white), reference altitude (in magenta), current descent rate (white arms), and reference descent rate (magenta arms). Optional pursuit information for altitude and descent rate (green) has been added to the VAVI display. In the center of the LD display, the roll and pitch angles are shown. On the left-hand-side of the screen, the tabbed menu provides the thrust and fuel levels of each engine, as well as capable hovering time and remaining delta V. Fuel gauge and thruster icons are placed in tandem to provide intuitive recognition. Lastly, the heading direction, horizontal velocity, and distance from the designated landing site are indicated in the graphic on the bottom right of the screen. All of this

[2] Figures and Tables available in Appendix 8.3

is super-imposed on a background image which is a camera view of the exterior; the designated landing site (60-meter radius) is also shown.

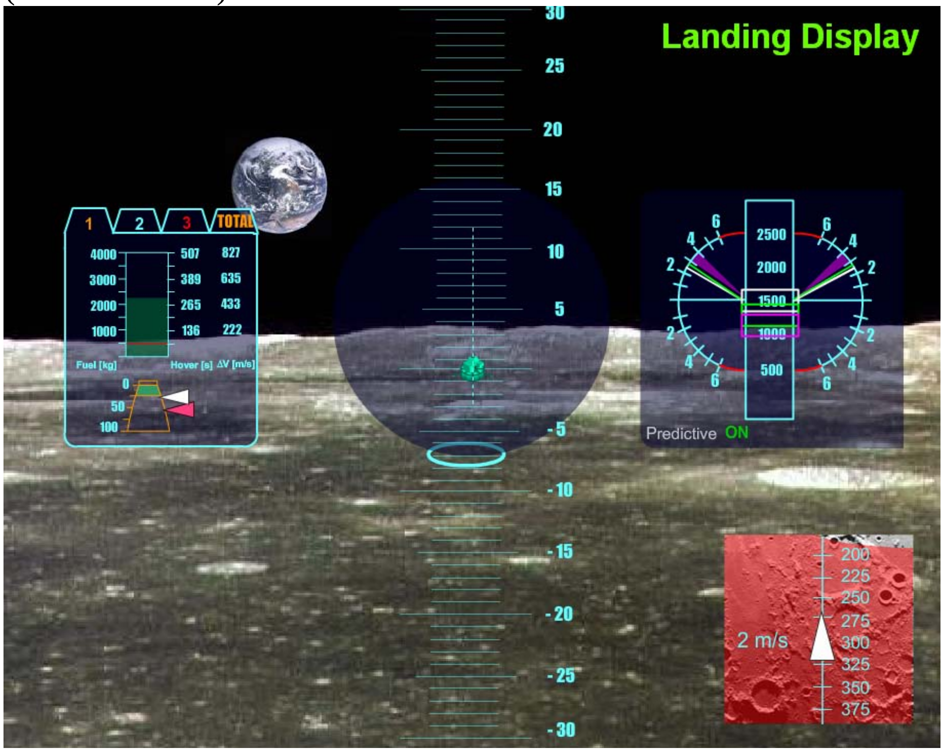
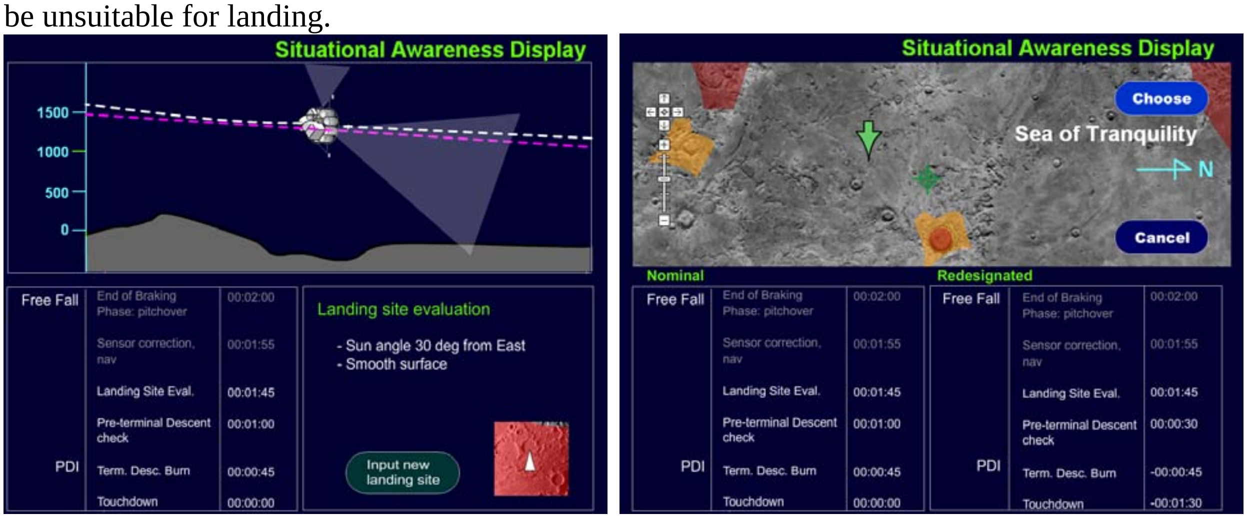


Figure 15. Landing Display.

Based on MIT LAV landing display [1]. The each tabs of the left item can show information of each engine. The red area on the item of the lower right indicates undesirable landing zones

4.2.2 Situational Awareness Display

Figure 16 shows the situational awareness display, which provides horizontal display, timeline for the landing, and landing site re-designation for the commander and pilot. A scrollable and zoom-able map interface of the lunar surface is provided to re-designate a new landing site. Improved knowledge of the lunar surface would be used to highlight areas on the map that would be unsuitable for landing.



(a) Nominal Display (b) Landing Site Redesignation Mode

Figure 16. Situational Awareness Display

On the nominal display, the upper half shows the horizontal velocity, reference trajectory and current trajectory. Below are the checklist and landing site redesignation link. On the landing redesignation mode display, the green arrow is the currently designated landing site, while the green crosshair is the new landing site the commander or pilot is designating.

4.2.3 Systems Status Display[3]

Figure 25 shows the systems status (SS) display which provides the following information: subsystem status, the root cause of a failure, the sequence of failures due to the root cause, repair procedures, and mission abort scenarios. The main display (Figure 25.a.) shows the overall subsystem status. Clicking the alerts brings you to the subsystem alert displays (Figure 25.b). The blinking alarm light colors are based on the color codes shown in Table 17. The alert displays include root causes of the failures detected by the Intelligent Cockpit System; the concept of the Intelligent Cockpit System is adopted because it is important for the crew to know whether it is the root cause or an effect. The SS display also shows consequences of the root causes, and repair options to help astronauts troubleshoot the problems or make a decision to abort a mission. At the bottom of the SS display is a direct link to the abort displays (Figure 25.c.), which provide checklist(s) of possible abort scenario(s). The Macromedia Flash™ movies which demonstrate the interactive three displays are available at: http://apollo-gray.mit.edu/wiki/index.php/Human_Factors.

4.2.4 Window

The cockpit design also includes a window located as illustrated in Figure 27, which gives a field of view of approximately 50˚ down as measured from the horizontal. This window is located in the middle of the cockpit, allowing both astronauts to make use of the view. It is important to understand that the window is not designed to serve as the primary tool for navigating and controlling the lander under manual control. Under manual control, the commander makes use of the landing display, optional additional camera views and the SA display to control and navigate the lander. There is also a small window on the top of the vehicle designed to give a direct view of the docking mechanism that can be used during the rendezvous and docking operations.

4.3. Interior Design and Anthropometry

4.3.1. Total Volume

Although there is no accepted model of relation between total habitable module volume per astronaut and mission duration, NASA suggests the curves shown in Figure 28. The right figure is an enlarged version of the 0 to 1 month period of the left figure. The net cabin interior volume for the astronauts is 11.5 m^3 (depth 2.5m, width 2m, height 2.3m), which should be sufficient because the landing mission itself is shorter than seven days.

4.3.2. Cockpit Anthropometry

Given that the maximum gravity load experienced by the crew during ascent and descent is low (~1G), a standing position was adopted for the lander cockpit. Backrests and seat belts are

[3] Figures and Tables are available in Appendix 8.3

provided for comfort and safety during the landing. According to NASA's Man-Systems Integration Standards (MSIS), the cockpit layout should be provided for the 5th percentile Asian Japanese female and the 95th percentile American male. Based on this standard, the cockpit layout for the lander was designed, as shown in Figure 27. Considering this anthropometrically wide range, the seats, backrests, footholds, and keyboard heights are adjustable to individuals while the positions of the displays and the window are fixed in the cabin, as shown in Figure 27.a. The detail of the display layouts is shown in Figure 29. LD display and SS display are in front of the commander and the pilot, respectively, and SA display is between the LD and SS displays. The interchangeable displays adjacent to the window can show any of the aforementioned displays or external camera views on demand. For example, the commander and/or pilot can display the SS display on the interchangeable screen as well, or a camera view to see the detail of a landing site. All the displays are placed such that head and eye movements are minimized, and all the displays are visible from both astronauts on the front seats.

4.3.3 Input devices

Several design options were considered when defining the human-machine interface. For controlling the displays and general input and output from the computers, a keyboard will most likely be necessary. Additionally, we need some way to interface with the graphical display on the screens. The two main options are either to use touch-screens or have a mouse-type controller which moves a cursor on the screen. When considering touch-screens, the obvious problem is that of inadvertently commanding inputs. A general override switch would need to be added to the cockpit which would enable or disable all of the touch-screen functionalities. The other option is to have a moving cursor on the screen which can be controlled by some mouse-type device. Based on previous spaceflight experiences, a small joystick-type of controller, similar to the trackpoint (the red dot) on IBM ThinkPad notebooks, seems to work better than other devices in a weightlessness environment. However, a final decision on which type of interface works best will, to a large extent, depend on the improvements of space suit technology during the next few years. Manual dexterity is compromised when wearing an EVA suit and this has an important impact on the design of the computer interface. A touch-screen system might be the best option if dealing with significant reduction in manual dexterity. However, if significant advances are made, such as the development of mechanical counter-pressure astronaut gloves, then perhaps a mouse-type device would be the best choice if the astronauts are wearing pressurized suits. For manual control of the lander, a system similar to that used on the Apollo's LEM is considered. A joystick with three degrees of freedom allows for controlling the pitch angle with forward-aft movements, bank angle with left-right movements, and roll angle with clockwise-anticlockwise rotations of the control column. Additionally, a second input device would allow the commander to control the descent rate of the lander. Obviously, this form of "manual" control is not entirely manual. Several features would be incorporated into the control system to make the piloting task easier. For example, as the commander changes the attitude of the lander in order to move it along the horizontal plane, the computer's autopilot would adjust the engine's throttle to match the descent rate to the rate that has been commanded.

4.3.4. Life Support Systems

As shown in Figure 27.b, the astronauts in the back seat will wear full EVA suits, while the commander and the pilot will wear emergency space suits to maintain dexterity and field of view. The emergency space suits have additional gloves and helmets that can be worn if necessary, and are connected to the life support systems of the full EVA suits via umbilical cables to support emergency egress. Having pressurized suits and gloves and helmets on is desirable from a safety point of view, but can limit the operator's field of view and manual dexterity. Thus, the final decision on whether or not the astronauts will be wearing gloves and helmets and whether or not the suits will be pressurized depends largely on the technology incorporated into newly redesigned EVA suits which will be developed by the 2020 timeframe. Additional life support considerations include crew metabolic needs, waste management, and the cabin environment. Based on the operational conditions of the International Space Station, the following conditions will be maintained onboard the lander (

Table 18). Assuming an average metabolic rate of 2677 calories/person/day, Table 19 lists crew necessities and corresponding outputs. The basal metabolic rate was calculated from the calorie requirement for an average adult American man weighing 79 kg that sleeps 8 hours a day and spends the remaining 16 hours sitting. For scenarios that require higher levels of physical activity, the caloric intake will be increased; including 1 hour of heavy work and 2 hours of walking (i.e. in lunar operations) results in roughly a 25% increase in caloric requirements. Subsequently, drinking water requirements can also be expected to increase with increased physical activity. These processing rates will govern the design of onboard waste management systems such as CO_2 removal units, and water processing assemblies, as well as appropriate liquid and solid waste management units.

4.4. Crew Selection and Training

4.4.1 Crew Selection

Crew selection will follow current NASA basic requirements for physically fit, mentally sound, and intelligent pilots, scientists, engineers, and doctors. All crewmembers shall have a science or engineering background. The Commander, who is in charge of the piloting of the lunar lander, shall have previous test piloting experience, which is necessary because of the similar situations and requirements between test flying and commanding the lunar lander. The Pilot, who has responsibility for coordinating on-board operations and monitoring subsystems, shall be a pilot with flying experience and would preferably have experience in systems engineering. The other two crew members will be scientists, engineers, or medical doctors depending on the specific mission requirements. The crew roles are further detailed in Table 20.

4.4.2 Crew Training

The overall crew training goals include technical training (design and operations, failure modes and corrective actions), spaceflight training (simulator, instrument training, parachute and survival training), biomedical training (space physiology, medical equipment), and scientific training (space and the Moon). These tasks will be taught over five main phases, as seen in Figure 17.

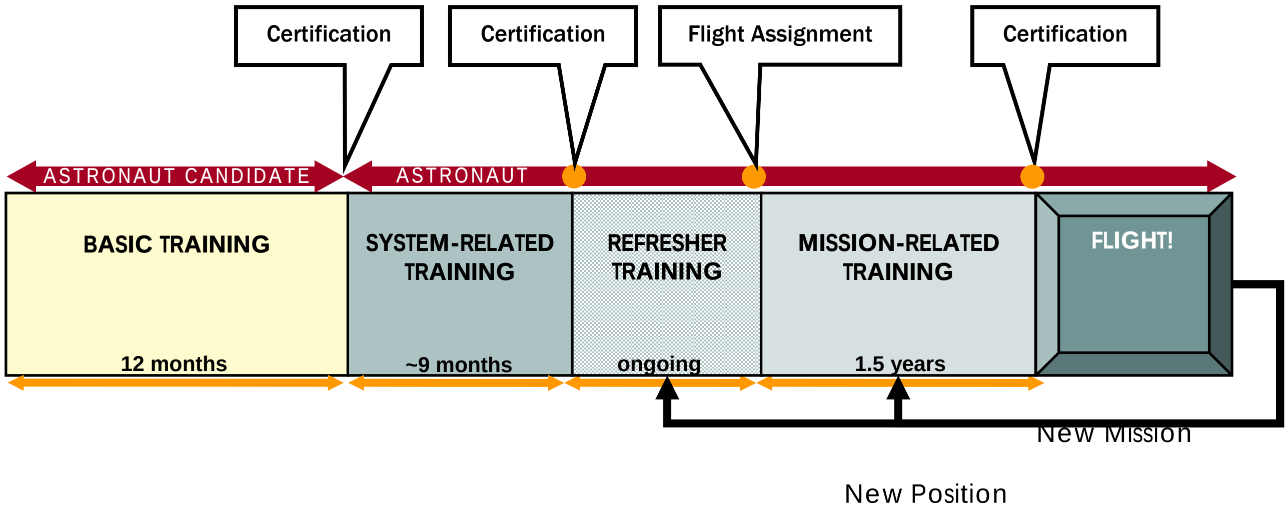


Figure 17. Crew Training Timeline

Newly selected astronaut candidates will spend the first 12 months in basic training, which covers basic knowledge of entire CEV system and operations required for Moon missions. The basic training includes the following: short courses in aircraft safety, situational awareness, and parachute, escape, scuba-diving, and survival training; ongoing training of piloting and language skills; basic science and technical courses; CEV overview courses including knowledge of the CEV system through lectures, briefings, textbooks, mockups, and flight operations manuals; single system trainer with simulations to become familiar with the system, to develop work procedures, and to react to basic malfunction situations; weightless training with "Neutral buoyancy" water tank and modified KC-135 flights; moon operations training at the Mars Desert/Arctic Research Station; biomedical training.

At the end of basic training, there will be a certification process to ensure competency in the aforementioned areas, which will include written and simulated tests, an interview, and a review by a board. Upon certification, the candidates are members of the astronaut corps but not eligible for flight assignment until one year after the basic training program due to additional training requirements. After basic training, the commander and pilot will undergo the same training so that the pilot is capable of taking over the commander's role if necessary. Additionally, one of the extra crewmembers will be fully trained to take over piloting duties. At this stage, system-related training commences to train for specific roles while further increasing familiarity with orbiter, lander, and outpost systems. This stage uses medium fidelity trainers for individuals and teams to become familiar with single- and multi-system operations in nominal mode. In addition, single system staged malfunctions as well as situational awareness (SA) training will be included, involving higher order cognitive training (such as attention sharing, information filtering, etc.) and simulator feedback based on the Situational Awareness Global Assessment Technique (SAGAT). The commander and pilot will train for lunar landing in a vehicle that is configured to simulate the handling characteristics of the lander, such as a modified helicopter, or a new and safer version of a LLTV. The commander and pilot should perform about 100 hours of training (throughout the whole training program) in a motion-based simulator or modified aircraft, which is similar to current requirements for the Shuttle. Physical and virtual simulations will also be run to practice Moon operations. Certifications will be done by a NASA instructor to test a deeper understanding of systems and repeatability of critical tasks through simulations. After that, refresher training must be done until assigned to a flight crew. Once the crew member has a flight assignment, they begin mission-specific training (recommended 1.5 years) that is highly tailored to the astronaut's assigned job. This involves practicing all phases of the mission in high-fidelity simulators as a team; multi-segment training to test mission rules and flight procedures in a full system mockup; multi-system failure modes to learn corrective actions for combined systems. Certification is necessary before flight to make sure that the crew is capable of all their assigned tasks and that they are physically in good condition.

4.4.3 Workload and Situational Awareness Testing

New systems, such as displays, require testing both from engineering and human factors viewpoints. To better design for humans, two tests are critical: workload and situational awareness (SA) assessments. Once a system becomes operational and astronauts have had basic training, the design will be tested in the loop with the astronauts so that the engineers can get feedback both from the subjective reports of the astronauts and the results of workload and SA assessments, allowing for subsequent refinements in the design.

A variety of workload tests will be performed to ensure proper workload balance. The first is the embedded secondary task technique. Here, a required (but less important) secondary task is imposed on a primary task to measure residual resources, such as responding to an air traffic controller (secondary task) while flying (primary task). Secondary tasks will be tested on normal operation and manual control with and without abort scenarios. This test has a long history in the field of workload research and has high face validity. The second test is visual scanning. This is a diagnostic index for the source of workload, although it can be physically obtrusive. The last test is the NASA Task Load Index, which is a subjective measure of workload done after the primary task is completed. These two other workload tests will be utilized to determine if any one screen, or part of a display, requires too much attention/workload and to test perceived workload. When using two or more tests, dissociation often occurs (i.e. conditions that are compared have varying effects on different workload measures), so the system designer must consider dissociation and then decide which workload assessment is more accurate for the specific circumstances.

Situational Awareness will be tested with the Situation Awareness Global Assessment Technique (SAGAT). This was the first popular and standardized procedure and now is the typical measurement technique for SA. This test collects SA data by pausing the simulations and asking the users a random set of SA-related questions. The SAGAT is useful because it is an immediate objective measurement that covers the whole span of SA issues.

5. Operations

5.1 Introduction to Operations

In developing the operations for a lunar landing, the Operations subteam embraced a philosophy of safety and simplicity. Simple operations plans increase mission safety by reducing the number of potential error points.

The operations team developed nominal procedures, based on simulated trajectories, vehicle capabilities, and inheritance from the Apollo and STS programs. Failure modes and effects analysis (FMEA) was also carried out, with the results informing the development of abort procedures and flight rules. Training the crew in these abort procedures and flight rules helps ensure safe operations during both nominal and off-nominal flight conditions.

5.2 Nominal Landing Operations

The nominal operations are designed for optimal crew attention on the landing situation. The pilot and commander work with the computer to coordinate the landing. Mission Control is updated periodically, but interaction between the ground and the crew is minimized, and the mission is designed to be completed without input from Mission Control. This design stems from a desire to have decreased ground-to-space communications volume. Decreasing the need to split attention between communications and flying tasks was chosen as a route to simplifying the landing by decreasing crew work load.

Procedures, in the form of printable timesheets, were developed. A sample section of a procedure appears in Figure 18.

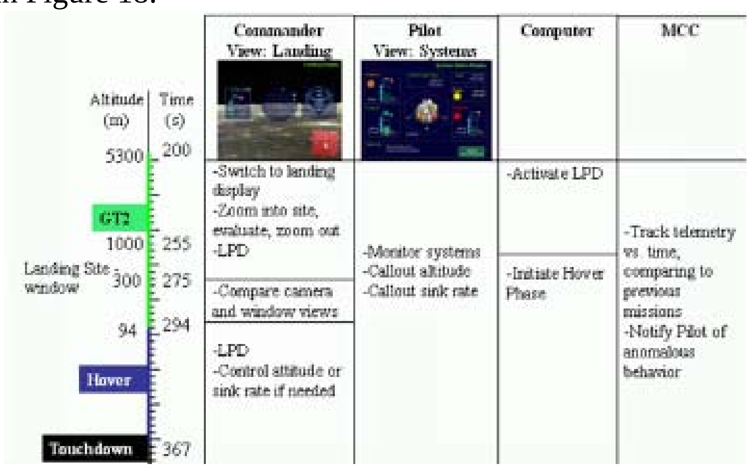


Figure 18. Sample selection of flight procedure

The procedure tasks are arranged so that timelines for one operations element can be constructed from columns, while rows indicate all activities occurring at the labeled time. Major events and altitude also appear in the time column. A full procedure appears in Appendix 8.4.1 of this report.

The pilot, serving as a systems engineer, utilizes the Systems Interface, as shown in Figure 16 to monitor systems, confirm that the critical systems are nominal, troubleshoot failures, and determine abort options. This interface decreases the need for mission control communications during critical failure situations, allowing the crew to respond more quickly to failures. The commander's main display will be the Landing Interface, as shown in Figure 15. However, during the first and second gravity turns of the trajectory, the commander will mainly utilize the situational awareness display to monitor altitude and sink rate and the camera displays to evaluate landmarks. The commander will be able to use external camera views to analyze the quality of the landing site and perform redesignations if needed.

The computer will be designed such that the entire landing can be performed autonomously, without human input. During a nominal landing, redesignation and manual hover-stage control will not be needed. However, these options have been included as a final layer of safety, which uses the human strengths of analyzing situations and making informed decisions.

5.3 Failure Modes and Effects Analysis

Because the FMEA was conducted as part of the operations plan, the focus was on the operational procedures required to recover from failures, instead of on hardware modifications required to mitigate failures, as is done in traditional FMEA. For this reason, only top-level failure modes and their effects were considered without analysis of the underlying "hardware" causing the problem. The analysis procedure used is as follows:

- Identification of critical subsystems based on knowledge of architecture
- Identification of major failure modes
- Evaluation of the effects of failure modes for each phase of powered descent
- Evaluation of criticality of failure based on knowledge of architecture
- Development of procedures for recovery from single-point failures
- Development of procedures for repeated failure (after initial recovery, if applicable)
- Recommendation of design changes required to aid in recovery from failure and avoid major losses

Table 8 is a listing of the some of the failure modes identified using the process described above. A full listing of failure modes considered can be found in Appendix 8.4.2 Failure Modes & Effects Analysis Results.

Note that the punctuation used in the "Operational Procedures for Recovery" field is meaningful. Commas separate a sequence of steps for a single procedure. Semicolons separate different procedures that can be used for recovery. These procedures are listed in order of preference, i.e. if the first one cannot be done or is unsuccessful when attempted, the crew should move on to the next procedure. Finally, the procedures described in “Operational Procedures for Repeat of Failure after Recovery” are to be used if the problem recurs after it was rectified using the first of the procedures from the previous column. This column is meaningless if the second or third procedure from the previous column was used to recover from the initial failure.

Table 8. Some failure modes and associated recovery procedures.

Item	Failure mode	Potential Effects of Failure	Phase of Failure	Critical Failure?	Operational Procedures for Recovery	Operational Procedures for Repeat of Failure after Recovery	Design Recommendations
Descent engine	Descent engine flameout	Not enough thrust to land, LOM, LOC	1st burn	Yes	Attempt restart to land; compensate with other engines/RCS; ascent stage abort to orbit	Attempt restart to descent stage abort; ascent stage abort to orbit	
	Descent engine flameout	Not enough thrust to land, LOM, LOC	2nd burn	Yes	Attempt restart to land; compensate with other engines/RCS; ascent stage abort to orbit	Attempt restart to descent stage abort; ascent stage abort to orbit	
RCS	RCS motor on	Reduced landing safety and accuracy, LOM	1st or 2nd burn	Yes	Compensate for loss; ascent stage abort to orbit		
	RCS motor on	Reduced landing safety and accuracy, LOM	Hover	Yes	Compensate for loss; ascent stage abort to orbit		Check maximum rate of pitch/roll/yaw to minimize sudden crash risk
Ascent engine	Ascent engine fire	LOM, LOC	1st or 2nd burn	Yes	Abort to orbit, depressurize; abort to landing, bailout		Minimize this risk with a high-performance fire suppression system
	Ascent engine fire	LOM, LOC	Hover	Yes	Abort to landing, bailout		Wear pressure suits on landing
Command and Control System	Command software failure	LOM, LOC	1st or 2nd burn	Yes	Attempt software patch; descent stage abort to orbit; ascent stage abort to orbit		Have backup software available, use completely different set of software in abort situation
	Command software failure	LOM, LOC	Hover	Yes	Attempt software patch; ascent stage abort to orbit; abort to landing		

The FMEA results presented in Table 8 affect both the operations and hardware design of the lunar lander in several important ways. By identifying design and operational issues that increase operational risk, the FMEA aids in increasing crew safety and mission success. Recovery procedure results are used to identify failures that lead to aborts. This aids in the development of flight rules for the mission. Additionally, this information is used to develop abort checklists by specifying the sequence of events that must take place in case of a failure. The "Design Recommendations" field lists items relevant to hardware design that greatly decrease operational risk. After this initial analysis, the major recommendations are as follows:

- The crew should wear pressure suits during landing to avoid complications in the event of Environmental Control and Life Support System (ECLSS) failures or hard landings causing depressurization
- There must be redundancy of wiring in the electrical system to bypass signals in the event of open or short circuits
- The wiring and electronics in the cabin must be able to survive increased humidity due to ECLSS failures
- The cabin should be equipped with a high-performance fire suppression system to prevent loss of crew in the event of a cabin fire
- The command and control system should be able to switch to backup software programs providing essential functionality in the event of command software failure
- The abort software system should be completely independent from the nominal mission software.

By incorporating the design recommendations suggested by the FMEA and training the crew in abort procedures for critical failures identified by the FMEA, crew safety and the likelihood of mission success can be greatly enhanced.

5.4 Flight Rules

Flight rules affect a significant portion of operations and play an important role in the success of the mission. They are used to provide Mission Control or crew personnel with guidelines to expedite decision making in time-critical situations. In developing flight rules for the proposed lunar landing, the Apollo rules were used as a stepping stone. While Apollo flight rules were customized for each mission, there was a common core of rules, as the majority of each flight was the same. These rules adhered to three main themes: appropriate responses in case of a specific failure at different points during the mission, determining authority in different situations, and general rules for each flight segment. Failure flight rules were determined from a Failure Modes and Effects Analysis, and as such include separate sub-rules for application during each phase of flight.

A second flight rule category is labeled as General Rules. This type of rule is designed to be a broad guideline encompassing a number of situations. However, it is superseded by any specific rules that apply. An example of this type of rule is the following:

> *A mission segment cannot begin if communication with Mission Control is lost, but shall continue in the event of such a loss during execution.*

This rule is in place to allow Mission Control the ability to confirm onboard trajectory calculations and do a final check on all systems prior to a burn. However, halting a burn due to communications loss is unnecessary, as technological advances since Apollo enable a significant amount of autonomy for the lander.

A final set of flight rules addresses who has final authority to make decisions during different mission segments. These rules are in place to allow smoother decision making processes and avoid confusion with who is responsible for specific tasks. An example of this type of rule is the following:

> *The Mission Commander may take over manual control of the spacecraft at any time without prior approval from Mission Control.*

This rule is important because there may be instances when there is no time to notify Mission Control and get approval for taking over manual control before action is required.

In general, flight rules do not have any direct impact on the public appeal of the lunar landing, but they do represent a framework upon which the guidelines for safety and efficiency in operation are laid, and as such have direct impact on the safety and simplicity of a landing. A full listing of developed flight rules appears in Appendix 8.4.3 Flight Rules.

5.5 Abort Procedures

Abort procedures are developed from nominal procedures, with the specific aborts detailed based on information from the Failure Modes and Effects Analysis. The crew workload is typically increased during an abort scenario, and the level of interaction with Mission Control also increases, but, again, the level of onboard autonomy on the lander allows for maximum flexibility in responding to an abort. A substantial number of abort procedures are possible, and it is envisioned that nearly all will be programmed into the onboard computers of the lander, ready for immediate use when a failure occurs. Abort scenarios may also be formatted as needed, for use by the crew during an abort or for providing a template for Mission Control. A sample section of an abort timeline-style procedure (for use by crew) appears in Figure 19.

290	Begin Hover				Begin Hover
291					
292	Begin Final Landing Phase	Control rate of descent if needed			
293					
294					
295			Monitor Rate of Descent	Initiate Final Landing Phase	
296				Display that currently in Final Landing Phase	
297		Control attitude if needed	Callout Rate of Descent		
298					
299					
300		Control attitude if needed	Callout attitude		
301		Confirm Hover Phase to MCC			
302					
303					
304				Initiate Attitude Hold	Acknowledge Hover Phase
305	RCS Motor On Failure Mode				
306	Roll increase	Verbally note roll increase	Confirm roll increase		
307	Roll increase	Verbally note RCS thruster error			
308	Roll increase	Verbally note RCS thruster error	Call up failure display		
309	Roll increase	Attempt manual compensation			
310	Roll increase	Attempt manual compensation	Confirm RCS failure	Display failure information	
311		Attempt manual compensation			
312		Verbally note nulling of error	Note failure response options		
313					Acknowledge RCS failure
314		Order automatic compensation	Program automatic compensation		
315		Compensate as needed	Analyze failure	Automatically compensate for attitude	
316		Compensate as needed			
317		Compensate as needed	Note available aborts		
318					
319			Advise abort		
320		Make abort decision	Call up abort options	Show abort options	
321			Describe abort options		
322			Describe abort options		
323			Describe abort options		Confirm abort decision
324		Select abort option	Confirm abort option		
325	Ascent Stage Engine warmup	Prepare ascent stage engine	Ready undock/abort charges	Prepare to jettison descent stage	
326	Ascent Stage Engine warmup			Arm undock/abort charges	
327	Ascent Stage Engine warmup		Verbally note abort ready	Calculate abort trajectory	
328	Ascent Stage Engine warmup				
329	Ascent Stage Engine warmup	Announce abort burn		Automatically fire abort charges	Advise on abort
330	Ascent Stage Abort to Orbit	Fire ascent stage engine	Confirm ascent stage abort burn	Fire ascent stage engine	Acknowledge abort burn

Figure 19. Sample section of RCS-Stuck-On abort procedure.

5.6 Impact of Technological Developments

Among the primary enabling technological developments, which allow for safer, simpler, and more publicly appealing lunar landing operations, are advances in computer capabilities and display interfaces. Increased computer memory capacity and decreased processing time allows extensive onboard error analysis, problem resolution, and abort calculation. Advances in user interfaces allow for multi-function displays and rapid presentation of key flight information, as well as increased situational awareness without the need for the commander to continuously take his eyes away from the landing display. Although the Apollo-era radio link to Mission Control and a simple window allow for a last level of fail-safes, the new advancements will allow landings with decreased fuel use and decreased pilot attention demands but increased ability to respond to failures.

5.7 Mission Control and Public Impact

Mission control support will be useful for analyzing systems status. For this design, each mission controller will monitor telemetry for several critical subsystems. The telemetry will be graphed versus time, and monitors will also include system limitations and plots from previous missions, so aberrant trends will be quickly apparent to controllers (Figure 20). This system, currently used for satellites of several kinds, will take full advantage of the amount of operational time that will rapidly accumulate over the course of a few lunar landings.

With increased onboard digital storage capacity and great improvements in video technology since Apollo, high-definition video footage of the landing transmitted to mission control can be distributed via NASA and the news media, which has the potential to greatly increase the public appeal of the lunar landings.

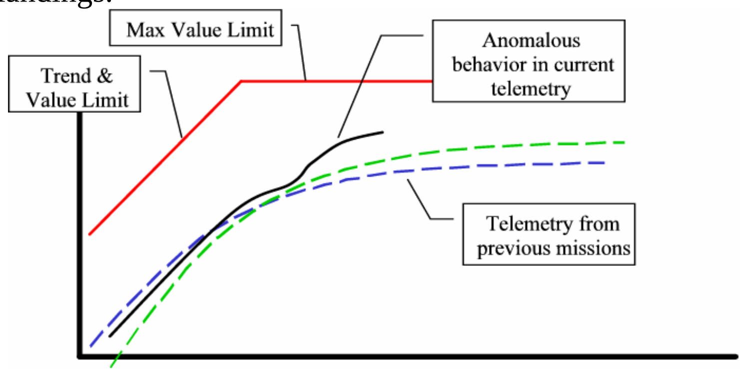


Figure 20. Mission control display of telemetry data

6. Conclusions

The Apollo Gray Team created a flexible, safe and capable lunar landing design that is compatible with NASA's current lunar exploration plans as outlined by LAT in December 2006. Through detailed analyses, the Gray Team established the feasibility of geometrical design, lunar landing trajectories, control of the vehicle along the descent trajectory, human-machine interface, mission operations procedures, and provided an integrated baseline for the lunar landing phase which was visualized using solid modeling, 3-dimensional printing, and animation of the trajectory control and the final landing as seen on displays in the cockpit.

A culminating part of the project can be seen in the final stages of the landing, primarily the last 100 feet (the hover phase). This portion of the landing is a result of collaboration between all four subteams. This critical portion of the landing can be seen in Figure 21. The Figure shows the trajectory, as well as the attitude and orientation through key parts. These orientations are chosen to give the astronauts the best possible scenario in terms of operations and human factors. A hover trajectory, with ample additional hover time available, is made possible by the GNC and architecture decisions on the beginning trajectory phases and propulsion types. Therefore, this final, critical phase truly captures the culmination of four subteams' work. It can also be seen in a video at: http://apollo-gray.mit.edu/touchdown.swf

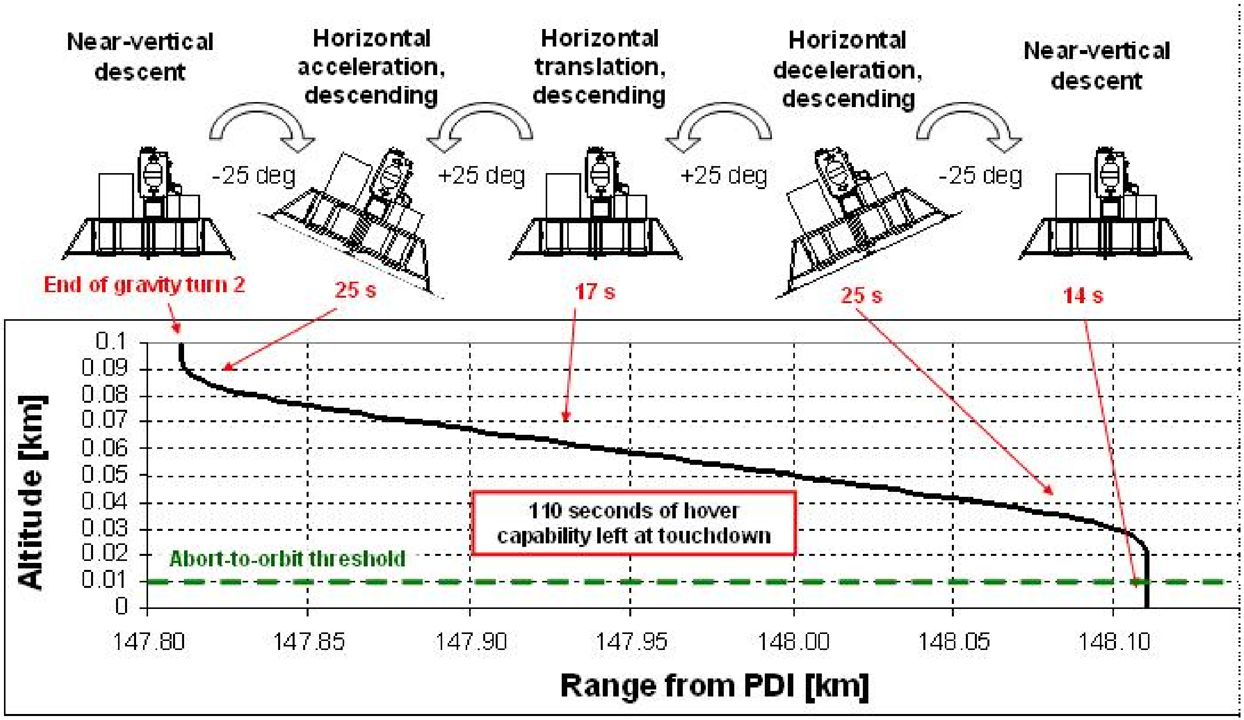


Figure 21. Landing Hover Phase

While the Gray team landing design is similar to Apollo and the NASA ESAS and LAT concepts in some respects (mainly due to the invariance of the physics of propulsion and astrodynamics); it is innovative in other respects:

- Reliance on the Earth Departure Stage of the Ares V for lunar orbit capture, which leads to a smaller lander than most NASA concepts with the same capabilities
- Reliance on automation that can perform the entire descent trajectory autonomously, without requiring any manual control from the operator
- Utilization of redundant cameras for surface visibility; this allows for landing site visibility much earlier in the descent than would be possible with windows only, while also maintaining a steeper and more fuel-efficient trajectory towards the end of the landing. It is very important to note that the crew can at any time pitch up the vehicle to get direct eyes-on visibility of the landing site through the window if so desired.
- For crewed missions, pitch-over is carried out at 100 m altitude providing eyes-on landing site visibility through the window and a very slow approach to the landing site which allows for additional extensive re-designation; in addition, over a minute of hover time is provided at touchdown for increased safety in case of an unsuitable landing site.

Based on these innovations, the Apollo Gray Team design would enable a very capable, flexible, and safe return to the Moon within the national lunar exploration strategy.

7. Annotated References

7.1 Systems Architecture References

1. President G. W. Bush, A Renewed Spirit of Discovery – The President's Vision for Space Exploration, The White House, Washington, January 2004.
    - Provides background / context for the overall exploration program and its objectives

2. NASA, Exploration Systems Architecture (ESAS) Study Final Report, www.nasa.gov, November 2005.
    - NASA overall transportation architecture; provides information on launch vehicles, Crew Exploration Vehicle, and ground infrastructure
    - Important reference because we intend to design a lunar landing that works with the NASA overall architecture (relevance)

3. Project Constellation, http://www.nasa.gov/mission_pages/constellation/main/index.html, NASA, September 2006.
    - Reference website on NASA exploration program; provides the latest information on launch vehicles etc.

4. Pahl, G., Beitz, W., Engineering design, Berlin / New York, Springer, 1995.
    - General reference on methods for systematically designing and varying concepts (mostly focused on hardware design)

5. NASA, Apollo Program Summary Report, NASA-TM-X-68725, JSC Houston, TX, 1975.
    - Reference information on the Apollo spacecraft and launch vehicles designs and their operations

6. NASA Public Affairs Office, Apollo 11 Press Kit, NASA, 1969.
    - Specific information on the operations of the Apollo 11 first lunar landing mission

7. NASA Request for Information (RFI) on Lunar Lander Concept Studies Solicitation Number: NNJ06LSAM05L, NASA, 2006
    - This document contains some of the reference assumptions and constraints for NASA's lunar lander.

8. Mark Wade, www.astronautix.com. May 2006.
    - Excellent source of general information on past, present, and planned spacecraft designs and missions; contains a lot of reference information on rocket engines, propellant combinations, etc.

9. Andrews / Andrews Technical Services, www.spaceandtech.com. 2006.
    - Similar to astronautix.com, but more focused on propulsion systems in general (liquid, solid, existing engines); has in particular good information on the RL-10 engine (which is baselined by NASA for lunar descent)

10. Larson, W. J., Pranke, L. K. (editors), Human Spaceflight – Mission Analysis and Design, McGraw-Hill, New York, 2000.
    - General reference textbook on human spacecraft and mission design; has introductions and tabular reference information on all subsystems and also some information on costing

11. Draper Laboratory/MIT NASA-CER Extension Period Final Report, MIT, Cambridge, MA, September 2005.
    - Final report of the Draper/MIT team for NASA's Concept Exploration & Refinement (CE&R) study; contains information on human lunar transportation and surface system architectures and associated analysis processes

12. Eckart, P., Lunar Base Handbook: An Introduction to Lunar Base Design, Development, and Operations, Space Technology Series, McGraw-Hill, New York, 1999.
    - Provides information on the lunar surface environment, as well as subsystems and costing; excellent introduction to the engineering aspects of landing and operating on the Moon

13. Humble, R. W., Henry, G. N., Larson, W. J. (editors), Space Propulsion Analysis and Design, McGraw-Hill, New York, 1995.
    - General reference on spacecraft propulsion system design; contains parametric models that can be used for conceptual design and sizing of propulsion system components

14. Connolly, J. F., "Kickin' up some dust" – Report on NASA LSAM design status, lunar lander project office, NASA, 2006.

- Description of NASA-internal human lander design efforts and results; excellent source for review of lunar lander concepts

15. 2006 study on LSAM architecture and design by Prof. Edward F. Crawley's group, Aeronautics and Astronautics, MIT, 2007.
    - Provides further information on LSAM design concepts and sensitivity analysis

16. Apollo – A Program Review, Society of Automotive Engineers, New York, 1964.
    - Provides various technical review documents regarding the LEM

17. Apollo Program Summary Report, National Aeronautics and Space Administration, Houston, 1975.
    - Full program summary report from Apollo, including change lists and testing and procedures for the LEM

7.2 Guidance, Navigation & Control

1. Ball Aerospace & Technologies Corp., "CT-602 Star Tracker [online listing]," http://www.ballaerospace.com/pdf/ct602.pdf [retrieved 18 April 2007]
    - Technical document with specifications for the CT-602 Star Tracker.

2. Ball Aerospace & Technologies Corp., "CT-633 Stellar Attitude Sensor [online listing]," http://www.ballaerospace.com/pdf/ct633.pdf [retrieved 18 April 2007]
    - Technical document with specifications for the CT-633 Star Tracker.

3. Bellcomm Inc. and LaPiana, F., "Touchdown Position Deviations Due to LM PGNCS IMU Error Sources." NASA CR-106885, 24 June 1969
    - Numerical analysis of Lm IMU errors' effect on landing distance. Used primarily for Apollo IMU technical specifications

4. Battin, Richard H. and Levine, Gerald M. "Application of Kalman Filtering Techniques to the Apollo Program," MIT Instrumentation Lab Report E-2401, April, 1969.
    - Includes disturbance sources accounted for in Kalman estimator. Also discuses state vector and state-vector updates.

5. Bennett, Floyd and Thomas Price. "Study of Powered-Descent for Manned Lunar Landings." National Aeronautics and Astronautics Administration report number NASA-TN-D-2426. http://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/19640018029_1964018029.pdf (accessed April 3, 2007).
    - This paper describes in detail Apollo's method of the powered-descent including the PDI. Will give the history behind how the Apollo trajectory was selected. It compares trade-offs leading to the design decisions they made.

6. Blarre, L. and Perrimon, N., "New Multiple Head Star Sensor (HYDRA) description and development status: a highly autonomous, accurate and very robust system to pave the way for gyroless accurate AOCS systems," AIAA Paper 2005-5932, August 2005.
    - Description of current status and development of the HYDRA Star Tracker; used for technical specifications

7. Brand, Tim et. All. "GN&C Technology Needed to Achieve Pinpoint Landing Accuracty at Mars." AIAA/ASS Astrodynamics Speceialist Conference, 16-19 August, 2004, Providence, RI.
    - This paper documents preliminary navigation research and trade studies involved in a theoretical mission to Mars. There is a useful section compiling IMU and Startracker data

8. Cheatham, D. C. and Bennett, F. V., "Apollo Lunar Module Landing Strategy", Apollo Lunar Landing Mission Symposium, June, 1966.
    - Useful for trajectory planning and setting up phases for the landing. Also contains some information on disturbance sources and numbers from Apollo (Class reading)

9. Chilton, Robert G., "Apollo Spacecraft Control Systems", Symposium on Automatic Control in Peaceful Uses of Space, Stavanger, Norway, 1965.
    - Details on the Apollo attitude control system (not guidance), including detailed block diagrams. Also describes the HW configurations

10. Clementine, http://nssdc.gsfc.nasa.gov/planetary/clementine.html [retrieved 1 May 2007]
    - Information of Clementine lunar maps

11. Crisp, R. & Keene, D., "Apollo Command and Service Module Reaction Control by the Digital Autopilot." MIT Instrumentation Laboratory. E-1964. May 1966.
    - Describes the design of Reaction Control system for Apollo CSM and LM, complete with control block diagrams and overall flow architecture. Also includes the implementation of RCS jet firing

12. Culler, Glen J. and Burton D. Fried. "Universal Gravity Turn Trajectories." *Journal of Applied Physics* 28, No. 6 (1957): 672-676.
    - Contains general information for a gravity turn, which is the first phase of our proposed trajectory. Also contains equations and analysis tools to calculate this part of the trajectory.

13. Eisenman, Allan Read, et al., "Realization of a Faster, Cheaper, Better Star Tracker for the New Millennium." IEEE 0-7803-3741-7, 1997.
    - Provided extensive information and technical specifications for the Advances Stellar Compass

14. Grayson, G. D., "Propellant Trade Study for a Crew Space Vehicle," AIAA 2005-4313, July 2005.
    - Extensive trade study performed for possible propellants to be used for a CEV-type vehicle; both descent engine and RCS propellants. Used as reference to decide fuels.

15. Grumman Aircraft Engineering Corp., "Lunar Excursion Module Familiarization Manual," NASA CR-129890, 15 Oct. 1965.
    - Overall description of the LM, with many technical specifications, including descent engine

16. Hoag, D., "The Guidance, Navigation, and Control of the Apollo Lunar Landing." *Space Technology and Science: Proceedings of the eighth international symposium*. Tokyo, Japan, 1969.
    - Describes the overall Functional design of Apollo GNC, along with performance from Apollo missions

17. Honeywell International Inc., "Honeywell HG8500 Series Radar Altimeter," DFOISR# 03-S-1426, 13 June 2003.
    - Technical document including specifications describing the Honeywell HG8500 Radar Altimeter

18. Honeywell International Inc., "Honeywell HG9550 Series Radar Altimeter," DFOISR# 03-S-0914, 2003.
    - Technical document including specifications describing the Honeywell HG9550 Radar Altimeter.

19. Kayton, M. and Fried, W. R., *Avionics Navigation Systems*, 2nd ed., Wiley-IEEE, New York, 1997.
    - Overall description of naviagation instrumentation concepts with details of radar altimeters and inertial measurement units.

20. Klumpp, Allan. "A Manually Retargeted Automatic Landing System for Lunar Module." MIT Instrumentation Laboratory. R-539 Rev 1. Aug. 1967.
    - Contains a block diagram of lunar landing approach phase guidance. Also has information on attitude and rate of descent limits for Apollo LM and coordinate frames used in Apollo LM GNC

21. Lunar Prospector, http://lunar.arc.nasa.gov/ [retrieved 1 May 2007]
    - Data on Lunar Prospector maps and measurements

22. Lunar Reconnaissance Orbiter, http://lunar.gsfc.nasa.gov/ [retrieved 1 May 2007].
    - Data on accuracy of LRO performance predictions

23. NASA Guidance and Control Systems Department, "Project Technical Report Task E – 38D: Apollo XIII Guidance, Navigation, and Control Systems Performance Analysis Report," NASA 9-8166, 24 July 1970.
    - Detailed account of GN&C performance on Apollo 13 but contained overall technical accuracy specifications for Apollo IMUs

24. Optech, Inc., "Optech Inc. – About Lidar [online]," http://www.optech.ca/aboutlaser.htm [retrieved 28 April 2007]..
    - Technical description of Lidar and Optech's Lidar performance, used mainly general comments about Lidar technology

25. Paschall, S. C., "Mars Navigation Performance Analysis using Monte Carlo Techniques." Master's Thesis, Massachusetts Institute of Technology, June 2004. pp. 38-40.

- Master's thesis describing a Mars mission; provided technical specifications and performed trade study between Honeywell MIMU and Litton LN200 IMU.

26. Rozas, P. and Cunningham, A. R., "Apollo Experience Report – Lunar Module Landing Radar and Rendezvous Radar." NASA TN D-6849, June 1972..
    - Excellent account of development and performance of LM landing radar. Provided detailed specifications and performance metrics

27. Schmidt, U., "ASTRO APS – The Next Generation High-Rel Star Tracker Based on Active Pixel Sensor Technology," AIAA Paper 2005-5925, August 2005.
    - Description of current development of the ASTRO APS Star Tracker; used for technical specifications

28. Shelton, H., "Apollo Experience Report – Guidance and Control Systems: Lunar Module Stabilization and Control System." MIT Instrumentation Laboratory. NASA TN D-8086. Nov 1975
    - Includes functional description of LM stabilization and control subsystem, with flight results, abort modes, and hardware assemblies

29. SODERN, "Low Cost Digital Sun Sensor [online listing]," http://www.sodern.fr/site/docs_wsw/fichiers_communs/docs/DSS%20.pdf?PHPSESSID=b124b5ab07df [retrieved 22 April 2007].
    - Technical document with specifications for Sodern's DSS sun sensor.

30. SODERN, "SED16 Star Tracker [online listing]," http://www.actus-multimedia.com/sodern/site/docs_wsw/fichiers_communs/docs/SED16.pdf [retrieved 20 April 2007].
    - Technical description including specifications for SED16 Star Tracker

31. SODERN, "SED26 Star Tracker [online listing]," http://www.sodern.fr/site/FO/scripts/siteFO_contenu.php?mode=&noeu_id=56&lang=EN&PHPSESSID=b124b5ab07df [retrieved 20 April 2007].
    - Technical description including specifications for SED26 Star Tracker

32. Sostaric, Ronald R. "Powered Descent Trajectory Guidance and Some Considerations for Human Lunar Landing." Paper presented at the annual international meeting for the AAS Guidance and Control Conference, Breckenridge, Colorado, February 3-7, 2007.
    - Contains information for recent analysis of the lunar landing. Describes in detail the modes for different trajectories and specifics on their design.

33. Stubbs, G., Penchuck, A., and Schlundt, R., "A Digital Autopilot for Thrust Vector Control of the Apollo CSM and CSM/LM vehicles," MIT Instrumentation Lab Report R-670, November, 1969.
    - Describes the method of controlling roll, pitch, yaw by both gimbaling the engine and the reaction control jets.

34. Tamblyn, S., Hinkel, H., and Saley, D. "NASA CEV Reference GN&C Architecture." 30th Annual AAS Guidance and Control Conference.
    - Good general description of GN&C proposal for NASA CEV program. This provided ideas and insight for our own design

35. Tooley, Craig. "Lunar Reconnaissance Orbiter Spacecraft & Objectives." 2006 AIAA- Houston Annual Symposium, May 19, 2006. [http://lunar.gsfc.nasa.gov/library/tooley-scobjectives-51906.pdf last accessed 4/8/07]
    - This is a presentation of the spacecraft details and mission objectives of LRO. There are detailed specifications for the LOLA laser altimeter, which is being considered in the GNC subteam design.

36. Vallado, David. *Fundamentals of Astrodynamics and Applications*. El Segundo, California: Microcosm Press, 2001.
    - This is a general reference for the Astrodynamics equations to design the trajectory and analyze it analytically. It contains information about basic equations and general methods for their use.

37. Vaughan, C.A., et. al. "Apollo Experience Report – Lunar Module Reaction Control System." NASA TN D-6740, March 1972.
    - Analysis of development and performance of LM Reaction Control system: used for technical specifications.

38. Wade, M., Encyclopedia Astronautica [online], http://www.astronautix.com/index.html [retrieved 24 April 2007].

- Large database of numerous spacecraft, rockets and space people. Used primarily for specifications of multiple descent engines and RCS thrusters.

39. Widnall, William S. “The Minimum-Time Thrust Vector Control Law in the Apollo Lunar-Module Autopilot,” MIT Instrumentation Lab Report E-2450, December, 1969.
    - Describes the attitude control law used in Apollo, specifically the minimum-time attitude controller using the descent engine.
40. Young, D.A., et.al. “*Artemis*: A reusable Excursion Vehicle Concept for Lunar Exploration,” AIAA Paper 2005-4010, July 2005.
    - Description of another concept architecture for lunar exploration, used as reference for GN&C ideas.

7.3 Human Factors

1. McCandless, Jeffrey, "Development of New Displays for the Cockpit of the Space Shuttle," *IBM 6th Annual Make Information Technology Easy 2002 Conference*, June 2002, http://human-factors.arc.nasa.gov/publications/McCandless_2002.pdf [retrieved 24 March 2007].
    - The author discusses the implementation of the displays in the cockpit of the Shuttle. Then, he describes how the Shuttle's displays were upgraded in 2001/2 to use new technology, decrease workload, and increase SA. The paper also includes color, graphics, and layout standards. Finally, the author compares the process for informing the astronauts of malfunctions under the current and proposed systems.

2. McCandless, Jeffrey, "Upgrades to the Caution and Warning System of the Space Shuttle," *Proceedings of the Human Factors and Ergonomics Society 47th meeting*, 2003, pp. 16-20, http://human-factors.arc.nasa.gov/publications/20051025103849_McCandless_HFES_2003%202.pdf [retrieved 27 March 2007].
    - The author describes the problems with the current Caution and Warning System (CAWS) and how the Shuttle wants to upgrade to a better system, making error detection easier. Only part of this plan was implemented when it was decided to retire the Shuttle in 2010, but recommendations are made for a full upgrade, utilizing the basis of the Shuttle's old system.

3. Knouse, Brad and Jean Zophy, " Intelligent Displays: Techniques to Support Safe and Autonomous Space Exploration," *Space 2006*, San Jose California, AIAA Paper 2006-7448, Sept 2006.
    - This paper is an overview of the following: data manipulation techniques, integrating data, detecting failures and root causes, generating action alerts, predicting critical data, automating procedures, user-interface techniques, and validation of techniques. The paper is based on the Space Shuttle.

4. Farkas, Andrew, "Apollo Experience Report – Lunar Module Display and Control Subsystem," NASA TN D-6722, March 1972, http://history.nasa.gov/alsj/tnD6722LMDisplayControl.pdf
    - The author describes, in great detail, the lunar module display and control subsystem, emphasizing major problems and solutions (many of which are outdated problems). Included is a description of each item, what it does, and how it's constructed.

5. Cummings, M., Wang, E., Smith, C., Marquez, J., Duppen, M., Essama, S., et al, “Conceptual Human-System Interface Design for a Lunar Access Vehicle”, Massachusetts Institute of Technology, Humans and Automation Lab [online], http://web.mit.edu/aeroastro/www/labs/halab/papers/HSI_interim_report_1.pdf [retrieved 3 April 2007].
    - This is an extensive report of the first phase of a Draper funded MIT study on how to design the next lunar lander vehicle. It gives a preliminary design of the landing displays needed based on the evolution of technology since Apollo and in lessons learned from the past.

6. Smith, C., “An Ecological Perceptual Aid for Precision Vertical Landings”, Massachusetts Institute of Technology, Humans and Automation Lab [online], http://web.mit.edu/aeroastro/www/labs/halab/papers/Smith-MIT-2006.pdf [retrieved 3 April 2007].
    - This is an MIT SM thesis that provides a detailed design presentation of a new display, the VAVI, which is particularly useful for vertical landing operations. It builds on Apollo program experience, and this display is integrated in the overall system described in [5].

7. “Apollo Program Summary Report,” NASA-TM-X-68725, April 1975, [online] http://history.nasa.gov/apsr/apsr.htm [retrieved 3 April 2007].
    - This is an extensive document that gives a synopsis of the Apollo program. It contains details on many of the technical aspects of the LM and all of its subsystems as well as on the overall systems integration.

8. Smith. C. A., Cummings, M. L., Forest, L. M., and Kessler, L. J., "Utilizing Ecological Perception to Support Precision Lunar Landing," Proceedings of HFES 2006: 50th Annual Meeting of the Human Factors and Ergonomic Society, San Francisco, CA, USA, 2006.
    - This paper proposes an integrated flight instrument display component "Vertical Altitude and Velocity Indicator (VAVI)" for use during vertical landing and hover operations of a future Lunar lander.

9. Norman, D. A., "The Design of Everyday Things," Currency; Reissue edition, 1990.
    - Written by the director of the Institute for Cognitive Sciences at University of California, investigates the psychological processes needed in operating and comprehending devices.

10. Kelly, T. J., "Moon Lander: How We Developed the Apollo Lunar Module," Smithsonian, Washington D.C., 2004.
    - Detailed documentary of the Lunar Module development told by the author, a former Grumman engineer who experienced from top level designing to manufacturing of the LM.

11. Jones, Harry., "Design Rules for Space Life Support Systems", *33rd International Conference on Environmental Systems (ICES),* Vancouver, Canada, SAE Paper 2003-01-2356, July 2003.
    - Provides engineering rules of thumb for life support system design such as: human metabolic needs, hygiene water requirements, atmosphere losses

12. Klaus, D.M., et al., "Spacecraft Life Support System Design Guidelines for Human Exploration of the Moon and Mars", *35th International Conference on Environmental Systems (ICES)*, Rome, Italy, SAE Paper 2005-01-3008, July 2005.
    - Provides specific human consumable and throughput values in kg/crewmember/day for atmospheric, water, and waste considerations needed in life support systems design.

13. Eckart. P., "Spaceflight Life Support and Biospherics," Space Technilogy Library, 1994.
    - This is a comprehensive summary of current and future life support systems including from physico-chemical to bio-regenerative, from small-scaled to large scaled systems.

14. Clement. G., "Fundamentals of Space Medicine," Springer; 1st ed., Springer, 2003.
    - Describes space experiments of Salyt, Mir, Spacelab, and the Space Shuttle missions, covering fundamental space life sciences.

7.4 Operations

1. Wertz, J. R., and Larson, W. J. (eds.), *Space Mission Analysis and Design*, 3rd ed., Space Technology Library, Microcosm Press, El Segundo, CA, 1999, Ch. 14.
    - Will provide model for data transport and delivery inclusion.

2. Larson, W. J., and Pranke, L. K. (eds.), *Human Spaceflight Mission Analysis and Design*, McGraw-Hill, New York, Ch. 26.
    - Will assist in developing a basic mission plan for crewed operations.

3. *Report of the Apollo 13 Review Board*, National Aeronautics and Space Administration, Washington, D. C., 1970.
    - Will show where possible abort scenarios may occur.

4. Brody, A. R., "Spacecraft flight simulation: a human factors investigation into the man-machine interface between an astronaut and a spacecraft performing docking maneuvers and other proximity operations," Space Systems Laboratory, Dept. of Aeronautics and Astronautics, Massachusetts Institute of Technology, Cambridge, MA, 1987.
    - Will provide information on key human-machine interactions during close landing phase.

5. Cooper, B., "Apollo Press Kits," URL: http://www-lib.ksc.nasa.gov/lib/presskits.html [cited 3 April 2007].
    - Will provide timed list of mission events, with dV's and rationale. Will also provide abort scenarios.

6. Dismukes, K., "Communications Transcripts: Mercury Through Apollo," URL: http://www.jsc.nasa.gov/history/mission_trans/mission_transcripts.htm [cited 3 April 2007].
    - Transcripts of landings will show actual events as they proceeded.

7. Mindell, D. A., “Connolly Presentation,” URL: http://stellar.mit.edu/S/course/STS/sp07/STS.471/courseMaterial/topics/topic19/resource/connolly.aiaa.02.20.07/connolly.aiaa.02.20.07.pdf [cited 3 April 2007].
    - Will provide basic information on lunar lander.

8. Feather, M. S., Cornford, S. L., and Moran, K., “Got risk risk-centric perspective for spacecraft technology making,” 4th National Symposium on Space System Risk Management, Manhattan Beach, CA, 2004.
    - Will provide information on risk modeling.

9. Feather, M. S., Cornford, S. L., and Moran, K., “Risk-based analysis and decision making in multi-disciplinary environments,” ASME International Mechanical Engineering Congress and Regional Expo, Washington, DC, 2003.
    - Will provide information on risk modeling and possible procedures.

10. Wertz, J. R., and Larson, W. J. (eds.), *Space Mission Analysis and Design*, 3rd ed., Space Technology Library, Microcosm Press, El Segundo, CA, 1999, Ch. 14.
    - Provided model for data transport and delivery inclusion.

11. Larson, W. J., and Pranke, L. K. (eds.), *Human Spaceflight Mission Analysis and Design*, McGraw-Hill, New York, Ch. 26.
    - Assisted in developing a basic mission plan for crewed operations.

12. *Report of the Apollo 13 Review Board*, National Aeronautics and Space Administration, Washington, D. C., 1970.
    - Showed where possible abort scenarios may occur.

13. Brody, A. R., “Spacecraft flight simulation: a human factors investigation into the man-machine interface between an astronaut and a spacecraft performing docking maneuvers and other proximity operations,” Space Systems Laboratory, Dept. of Aeronautics and Astronautics, Massachusetts Institute of Technology, Cambridge, MA, 1987.
    - Provided information on key human-machine interactions during close landing phase.

14. Cooper, B., “Apollo Press Kits,” URL: http://www-lib.ksc.nasa.gov/lib/presskits.html [cited 3 April 2007].
    - Provided timed list of mission events, with dV’s and rationale. Also provided abort scenarios.

15. Dismukes, K., “Communications Transcripts: Mercury Through Apollo,” URL: http://www.jsc.nasa.gov/history/mission_trans/mission_transcripts.htm [cited 3 April 2007].
    - Transcripts of landings showed actual events as they proceeded.

16. Mindell, D. A., “Connolly Presentation,” URL: http://stellar.mit.edu/S/course/STS/sp07/STS.471/courseMaterial/topics/topic19/resource/connolly.aiaa.02.20.07/connolly.aiaa.02.20.07.pdf [cited 3 April 2007].
    - Provided basic information on lunar lander.

17. Feather, M. S., Cornford, S. L., and Moran, K., “Got risk risk-centric perspective for spacecraft technology making,” 4th National Symposium on Space System Risk Management, Manhattan Beach, CA, 2004.
    - Provided information on risk modeling.

18. Teague, Kipp, “The First Lunar Landing,” URL: http://www.history.nasa.gov/alsj/a11/ [cited 12 May 2007].
    - Provided video and audio record of first lunar landing phase.

19. Neff, Gary, “A Visit to the Snowman,” URL: http://www.history.nasa.gov/alsj/a12/ [cited 12 May 2007].
    - Provided video and audio record of second lunar landing phase.

20. Ryba, Jeanne, “Kennedy Media Gallery,” URL: http://mediaarchive.ksc.nasa.gov/search.cfm [cited 12 May 2007].
    - Provided video and audio record of STS landings for comparison with Apollo lunar landings.

21. Feather, M. S., Cornford, S. L., and Moran, K., “Risk-based analysis and decision making in multi-disciplinary environments,” ASME International Mechanical Engineering Congress and Regional Expo, Washington, DC, 2003.
    - Will provide information on risk modeling and possible procedures.

22. Jones, Eric M., “Apollo 11 Flight Mission Rules,” URL: http://history.nasa.gov/alsj/a11/A11MissionRules.pdf [cited 12 May 2007].
    - Used as a baseline to compare Gray team mission rules

8. Appendices

This section provides additional material on the Apollo Gray Team lunar landing design.

8.1 System Architecture Appendices

This section provides additional material on the Apollo Gray Team lunar landing design. It is organized into four main subsections, corresponding to the appendices for each subteam: systems architecture, GNC, human factors, and operations.

8.1.1 Lunar Lander Concepts

Nine distinct concepts for lunar landing missions and lander designs were reviewed by the systems architecture team. In the following, a brief description of each of these architectures is provided, along with a reference:

- Apollo LM: the Apollo Lunar Module (or LM) was the lunar lander used during the Apollo program. It was transported to lunar orbit by the Saturn V launch vehicle and the Apollo Command and Service Module (CSM). Once in lunar orbit, 2 of the 3 crewmembers descended to the lunar surface and performed an exploration mission at a specific site (sortie mission). After ascent from the lunar surface, the LM docked with the CSM, crew and cargo were transferred to the CSM, and the LM was then discarded.
  - Reference: http://history.nasa.gov/alsj/alsj-JSC09423.html
- Soviet lunar lander: the Soviet lunar landing architecture involved two spacecraft, which were both intended to be launched towards the Moon using the N-1 launch vehicle: a modified Soyuz spacecraft as a lunar orbit and Earth return vehicle, and a lunar lander. Different from Apollo, the Soviet architecture used a single propulsion stage to provide LOI and (after undocking of the orbiter), the majority of the descent propulsion. Shortly before landing, this stage was jettisoned, and the lander provided the remaining landing delta-v. The lander stage also provided all of ascent and habitation on the surface. The Soviet design involved only two crew: one crewmember stayed on board the orbiter, while the other conducted the surface excursion.
  - Reference: http://www.astronautix.com/craft/lk.htm
- First Lunar Outpost (FLO) was a post-90-day-study lunar exploration architecture prepared by the Office of Exploration under Mike Griffin. It involved a direct lander lunar crew transportation architecture and a pre-deployed lunar outpost that would be visited by crews for stays of 45 days. The architecture was based on a 200 mt to LEO launch vehicle which would launch the crew transportation system and the lunar outpost (on separate flights). The architecture is very interesting, especially also the outpost concept (independent of transportation infrastructure).
  - Reference: http://www.nss.org/settlement/moon/FLO.html
- Lunox was another Office of Exploration concept for lunar exploration involving in-situ oxygen production on the lunar surface. It is in many ways similar to FLO, although ISRU leads to significantly reduced TLI mass requirements for crew transportation.
  - Reference: http://www.nss.org/settlement/moon/LUNOX.html

- The NASA ESAS lander concept was part of the Exploration Systems Architecture Study published by NASA in late 2005. ESAS is the foundation of all of NASA's current lunar exploration plans in the post-shuttle era. The lunar lander concept is based on an EOR/LOR mode, and is mostly designed around sortie missions (large ascent stage, reduced surface payload delivery). ESAS represents the first fully integrated and near-term feasible lunar exploration plan put forth by NASA in over 10 years.
    - Reference (in particular Chapter 4: Lunar Architecture (and lander)): http://www.nasa.gov/mission_pages/exploration/news/ESAS_report.html
- The 2006 MSFC lander concept is a representative of post-ESAS lunar lander design concepts studied at NASA during the year 2006. It involves the same general mission mode as ESAS, but features a much smaller and side-mounted ascent stage and a descent-stage design driven by outpost missions as much as sortie missions.
    - Reference: www.aiaa-houston.org/cy0607/event-22feb07/**Connolly**_AIAA_2-20-07.pdf
- The 2006 Lockheed Martin concept is another post-ESAS design that was prepared in response to a RFI by NASA. It is radically different in configuration because it features a horizontal lander with the ascent stage mounted on one side. This introduces significant challenges for delivering large amount of cargo to the lunar surface for uncrewed outpost transportation. The concept also utilizes MMH/N2O4 for ascent propulsion as opposed to LCH4/LOX.
    - Reference: pdf.aiaa.org/preview/CDReadyMSPACE06_1393/PV2006_7284.pdf
- AS part of their lunar lander and campaign architecture study, Prof. Edward Crawley's research team at MIT proposed a lunar lander design which utilized the Ares V EDS for lunar orbit insertion. This allows for a much smaller descent stage, facilitating crew access to the lunar surface and offloading of cargo. It may also allow a lower the number of engines required in the descent stage because of the reduced lander weight.
    - Reference: personal communication with research team
- In December 2006, the NASA Lunar Architecture Team (LAT) provided an updated lunar campaign plan and a lunar landing architecture, which featured a modified lunar lander with 2 crew compartments (one left on the lunar surface), and a more complex lander geometry to accommodate the different landing use cases.
    - Reference (see charts by Doug Cooke and Tony Lavoie for more detailed lunar architecture and lunar lander design description): http://www.nasa.gov/mission_pages/exploration/main/2nd_exploration_conf.html

8.1.2 Lunar Mission Modes

Four mission modes were considered for the review of lunar lander concepts (see Figure 22):

- Direct: in this mission mode, the crew launches in their Earth entry crew compartment using a single launch vehicle (stopover in LEO and LLO possible, but no rendezvous). This means that the entry crew compartment goes to the lunar surface, and the associated heat shield and Earth return propulsion must be brought down and then up again in the lunar gravity well. This usually leads to high mass requirements at Trans-Lunar Injection (TLI).

- Lunar Orbit Rendezvous (LOR): all mission elements are launched and inserted towards the Moon using one launch vehicle. After capture in lunar vicinity, however, only part of the stack descends to the lunar surface while the Earth return propulsion and the Earth entry compartment are usually left in orbit. This leads to reduced mass at TLI; however, two crew compartments and an additional propulsion stage are required, which leads to additional development and operational cost.
- Earth Orbit Rendezvous (EOR): this mission mode utilized several launch vehicles to transport the individual elements of the lunar stack into Earth orbit where they are mated. The stack then departs to the Moon, where the Earth entry compartment and the return propulsion are brought to the lunar surface (like in direct). This mode also only requires one crew compartment design, but multiple launches and a rendezvous in Earth orbit (increased operational cost and risk).
- EAR/LOR is a hybrid mission mode which is identical to LOR in lunar vicinity, but utilizes several launches to deliver the elements of the lunar stack to Earth orbit, where they are mated. Thus, a larger TLI payload can be assembled in Earth orbit prior to departure, enabling more capability on the lunar surface. Arguable, this mission mode represents a classic cost-risk-performance trade: by increasing both cost (more launches) and risk (more rendezvous and more launches), it is possible to increase performance

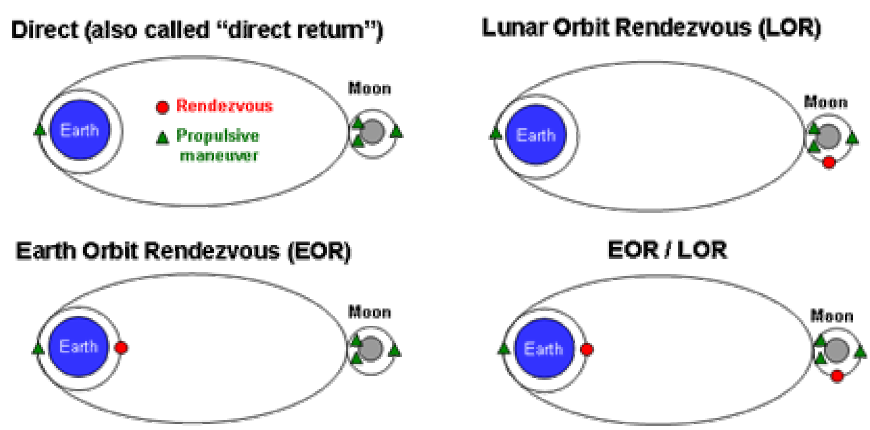


Figure 22. "Mission modes" for lunar missions

8.1.3 Lunar Landing Morphological Matrix

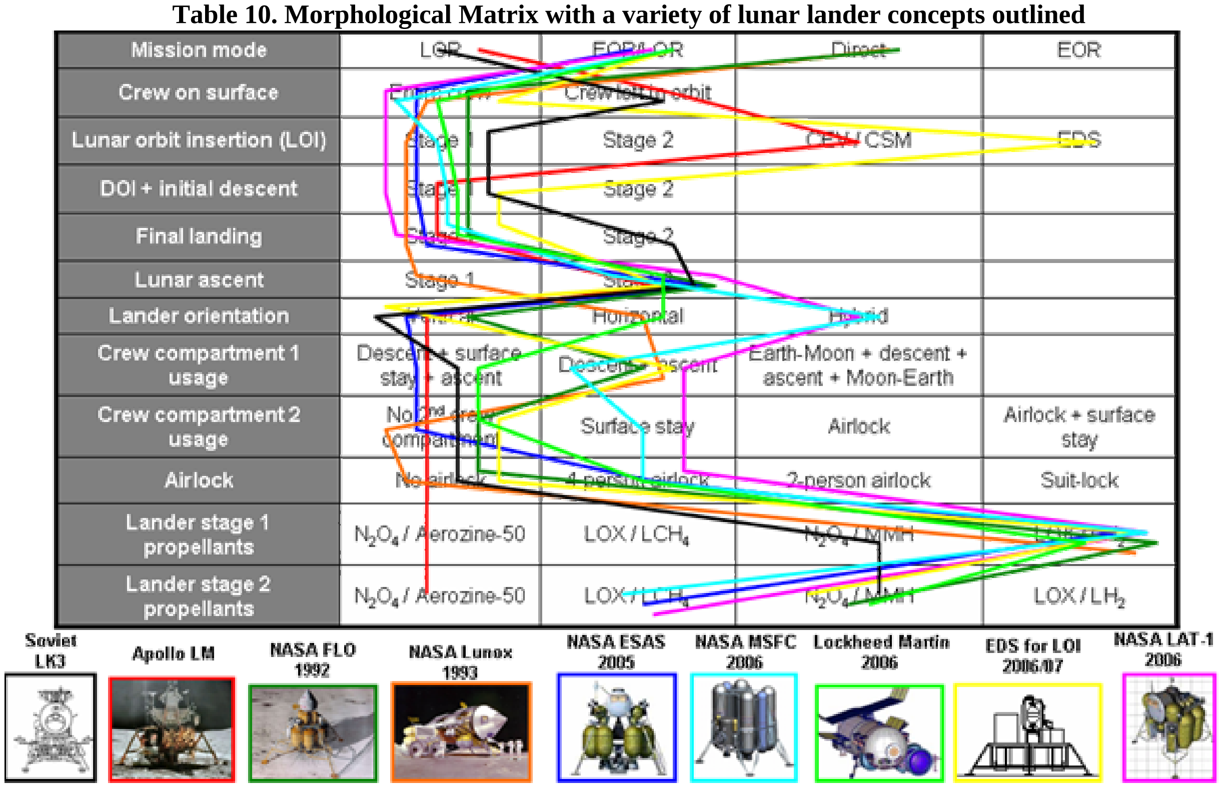
Table 9 provides an overview of the Morphological Matrix used to review lunar lander and mission concepts (see Systems Architecture Section in report); Table 10 shows the same matrix with the 9 lunar lander concepts outlined. Each colored path through the matrix represents one architectural concept; depictions of the associated lander configurations are shown below Table 10 with matching colored boxes around the pictures.

Table 9. Morphological Matrix for mapping lunar lander concepts

Morphological matrix of lunar landing (lander and mission design):

Mission mode	LOR	EOR/LOR	Direct	EOR
Crew on surface	Entire crew	Crew left in orbit		
Lunar orbit insertion (LOI)	Stage 1	Stage 2	CEV / CSM	EDS
DOI + initial descent	Stage 1	Stage 2		
Final landing	Stage 1	Stage 2		
Lunar ascent	Stage 1	Stage 2		
Lander orientation	Vertical	Horizontal	Hybrid	
Crew compartment 1 usage	Descent + surface stay + ascent	Descent + ascent	Earth-Moon + descent + ascent + Moon-Earth	
Crew compartment 2 usage	No 2nd crew compartment	Surface stay	Airlock	Airlock + surface stay
Airlock	No airlock	4-person airlock	2-person airlock	Suit-lock
Lander stage 1 propellants	N_2O_4 / Aerozine-50	LOX / LCH_4	N_2O_4 / MMH	LOX / LH_2
Lander stage 2 propellants	N_2O_4 / Aerozine-50	LOX / LCH_4	N_2O_4 / MMH	LOX / LH_2

Table 10. Morphological Matrix with a variety of lunar lander concepts outlined



8.1.4 Lunar Lander Concept Comparisons

Figure 23 shows the Gray Team lander reference design in a size comparison to other proposed and / or built lunar landers. Note: the top of the descent stage of the reference design is about at the same height above ground as the top of the Apollo LM descent stage; the ESAS lander had a much taller descent stage. This is mainly due to the fact that the Gray Team design utilizes the EDS stage for lunar orbit capture, thereby reducing the propellant mass (and volume) required in the descent stage.

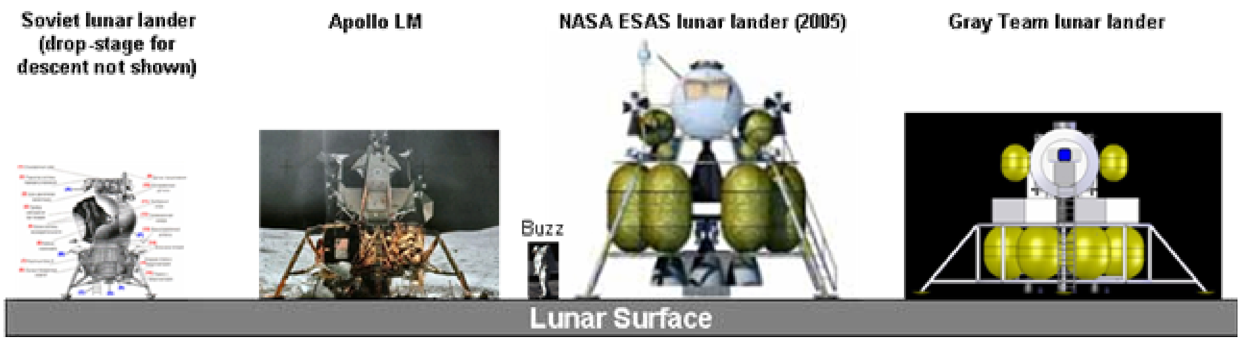


Figure 23. Size comparison of lander configurations

Figure 24 shows the Gray Team lunar mission stack in Low Lunar Orbit prior to separation of the CEV, descent orbit insertion, and descent to the surface in comparison to Apollo, NASA ESAS, and the Soviet lunar landing stack. Again, it can be seen that the Gray stack is comparable in size with Apollo.

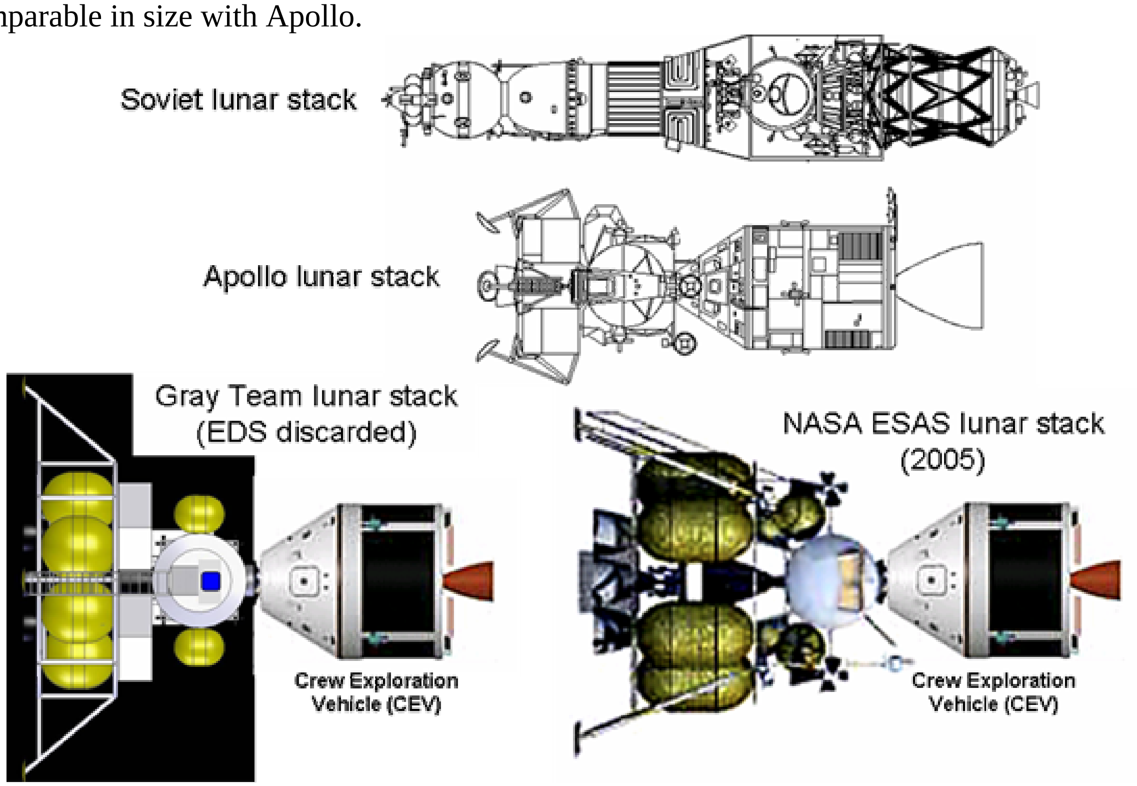


Figure 24. Comparison of vehicle stacks in lunar orbit prior to undocking and descent

8.2 GN&C Appendices

8.2.1 Hardware Comparisons

Table 11. IMU Comparison

		MIMU	LN200
	Size (in)	9.17dia. X 6.65	3.5dia. x 3.4
	Weight (lb)	9	1.65
Accelerometer Errors	Bias (μ-g)	100	300
	Scale Factor (ppm)	175	300
	Nonorthogonality (arcsec)	15	20
	Misalignment (arcsec)	15	20
	Random Walk (m/s/√s)	0.00015	0.00049
Gyroscope Errors	Bias (deg/hr)	0.05	1
	Scale Factor (ppm)	5	100
	Nonorthonality (arcsec)	25	20
	Random Walk (arcsec)	0.0001	0.0012

Table 12. Star Tracker and Sun Sensor Comparison

	SODERN Sun Sensor	ASC*	CT-602	CT-633	HYDRA APS*	SED16	SED26
Pitch and Yaw Accuracy (arcsec)	72	1.4	3	6	2	15	3
Roll Accuracy (arcsec)	36	8	5	30	16	55	15
Mass (kg)	0.3	1.53	5.4	2.5	2.2-3	3	3.1
Field of View	120x120	16x22	8x8	20x20	N/A	25x25	30x30
Power (W)	1	5.5	8	8	12	7.5	7.5
Size (mm)	130x120x45	100x100x100	250x180 dia.	140x135 dia.	115x115x135	170x160x290	160x170x290
*In early development							

Table 13. Available Landing Radar Comparison

	HG8500	HG9550
Weight (lbs)	3	9.75
Max Power (W)	16	35
Electronics Size (in)	3.4x3.4x5.6	3.5x6.3x8.75
Accuracy	3%	2%

Table 14. Reaction Control Engine Comparison

	R-4D	RS-52	RS-42	RS-2101A	RS-28
Manufacturer	Aerojet	Rocketdyne	Rocketdyne	Rocketdyne	Rocketdyne
Propellants	LOX/LH2	N2O4/MMH	N2O4/MMH	N2O4/MMH	N2O4/MMH
Specific Impulse (sec)	312	405	441	287	295
Thrust (N)	490	107	441	1333	2667

Table 15. Descent Engine Comparison

	RL-10-B2	RL-10-A4-2
Thrust (kN)	110 kN	99.1 kN
Propellants	LOX/LH2	LOX/LH2
Specific Impulse (sec)	462	449
Length (m)	4.1	2.3
Diameter (m)	2.2	1.2

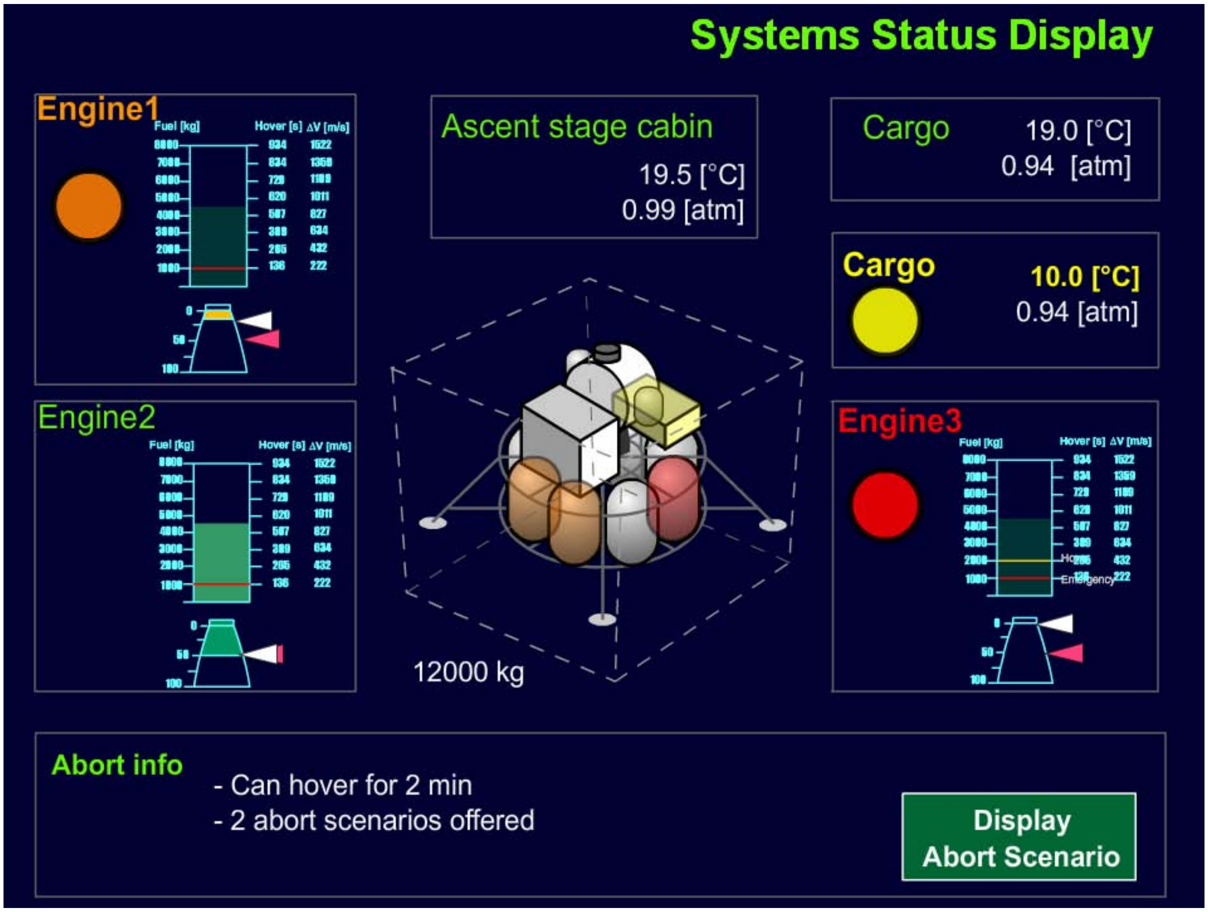
8.3 Human Factors Tables and Figures

Table 16. Cockpit display study of Apollo LM, Shuttle, and MIT Lunar Access Vehicle

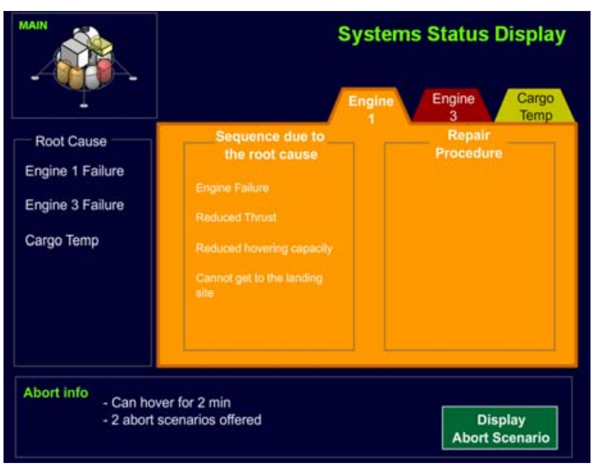
	FEATURES			ACTION ITEMS
	MIT LUNAR ACCESS VEHICLE	APOLLO LUNAR ACCESS MODULE	SPACE SHUTTLE	
1. Low Altitude		Altitude difficult to tell from terrain		System to inform astronauts of surrounding terrain
2,3. Spatial Disorientation	▪ Top view map with obstacles colored ▪ Lunar map assumed	Had difficulty in finding the landing site during the Apollo 12 mission	Half of astronauts get space motion sickness in first 3 days	Location of displays and controls to minimize need for head movement and frame switching
4. Situation Awareness	SA display	Systems engineer had to read out display information to commander	Error system had false alarms and too much unecessary information	Centralized error system with the root failure cause and overall vehicle health
5. Display Principles **5-1: Signal Detection** **5-2: Attention** **5-3: Graphical Perception** **5-4: Virtual Environments**	▪ No windows ▪ HUD type large displays instead ▪ Three displays: landing zone display (kind of basic T), SA display, and system status display ▪ Two crew members ▪ Intuitive integrated altitude (sink) and speed (sink rate) meter ▪ Not intuitive fuel & thrust gauge ▪ Time-stamped checklist provided	▪ Moving pointer, fixed pointer, numeric and status indicator ▪ Displays required one person to read and one to control ▪ Used just numbers and abbreviations ▪ Lights and error messages for warnings	▪ LCDs ▪ Make systematic and logical use of color ▪ Systems information "at a glance" ▪ Simple yet effective symbology ▪ Logical grouping, related information onto one display ▪ Color – suggestions vary from 6-12 color options ▪ Caution and warning: text fields should indicate only the source of the malfunction (root cause) and show only the information they really need at the time ▪ Display hierarchy should also be organized in a hierarchy that is easy to learn and remember	▪ Display based on basic T ▪ Ergonomics: sizes of the items on the displays should be determined by anthropometry ▪ Display available for the rest of the two crew members ▪ Status display illustrating lunar lander configuration ▪ Quickened displays during nominal, pursuit displays during off-nominal (both display modes available upon request by users, and the modes should be easily recognized by the users) ▪ Shuttle's color codes
6. Aviation Ctrl	Sink rate and altitude both displayed	Switches (toggle, rotary, pushbutton) and variable ctrls (potentiometer, synchro)	Partly automatic	▪ Manual control block diagram ▪ Input interface needed for astronauts (stick, yoke, switches, etc ▪ Determine allowed time lag
7. Reaction Time		Crew trained to reduce reaction time for engine cutoff during landing	▪ Fatigue increases reaction time ▪ Sleeping pills can increase reaction time	
8. Manual Ctrl	pp35 checklist of actions to be taken	Switches (toggle, rotary, pushbutton) and variable ctrls (potentiometer, synchro)	▪ Entry: only landing gear extension and braking action on the runway are required by the flight crew, but crew usually switches to manual once subsonic	
9. Time sharing	Two crew members involved in landing	Crew split tasks but they still had to work together to land		▪ Performance Resource Function (W&H Ch11) ▪ Make sure cross-modal sharing
10. Automation and Human Performance	In off-nominal situation: manual lander supervisory monitoring	Incorporate humans in the loop as much as possible – monitor, evaluate, and control	Automation: the danger is that the crew will have no insight into the basis for the C&W (caution and warning) decisions, and so no basis for troubleshooting these decisions and detecting possible errors.	▪ We should adopt function-mimicking displays (intuitive) ▪ How the other two crewmembers can help in what kind of emergency ▪ Use adaptive automation?
11. Memory	reduced	heavy		HUD type displays will help reducing memory load for astronauts
12. Decision Making	Abort when astronauts cannot find the place to land	Astronauts can abort based on uncertainty within physical constraints	▪ Launch scenarios have abort with human in loop (with ground control too) ▪ Landing like a glider - little choice	
13. Stress & Human Error			▪ Key error-prevention technique for CAU (cockpit avionics upgrade) was echoing keystrokes on a mobile scratchpad. As keystrokes were entered, the scratchpad highlighted the selected item number and, for data items, the data entered was shown in the box as it was typed	▪ Can we guess Yerkes Dodson Law curve? ▪ Physiological sensors? ▪ Can automation tell human error?

Table 17. Color codes

COLOR	HF NOTES	SHUTTLE	GRAY TEAM
Red	Danger	Warnings	Danger
Orange	Warning	Discrepancies between two software systems	Warning
Yellow/Amber	Caution	Off-nominal cautions	Caution
Green	Notice, piloted-selected		Pilot-selected data
White	Current status	Nominal, nominal data	Current status
Magenta	Target info		Reference status
Cyan	Background info		Background info
Dark blue	Optional	Background of display format	Background of display format
Dark gray	Optional	Lines separating regions of display format	Lines separating regions of display format
Light gray	Optional	Labels adjacent to data	Labels adjacent to data
Light green	Optional	Titles	Titles
Dark green	Optional	Optional	Optional



(a) Main Display



(b) Subsystem Alert Display

(c) Abort Display

Figure 25. Systems Status Display

The main view (a) shows the subsystem status. Clicking the alerts bring you to the subsystem alert displays which show the root causes of the failures, sequences due to the root causes, and repair procedures if any. Link to the abort display is also provided. The abort display shows checklist(s) of the provided abort scenario(s.)

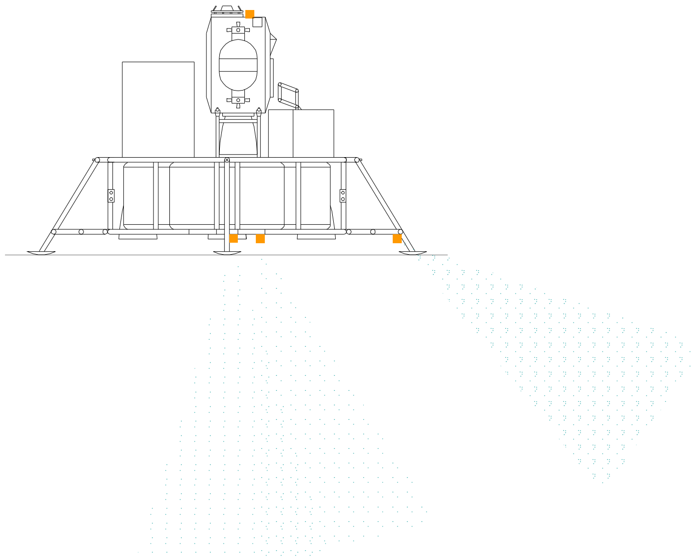
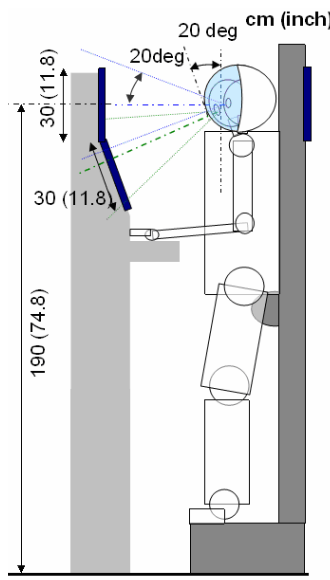
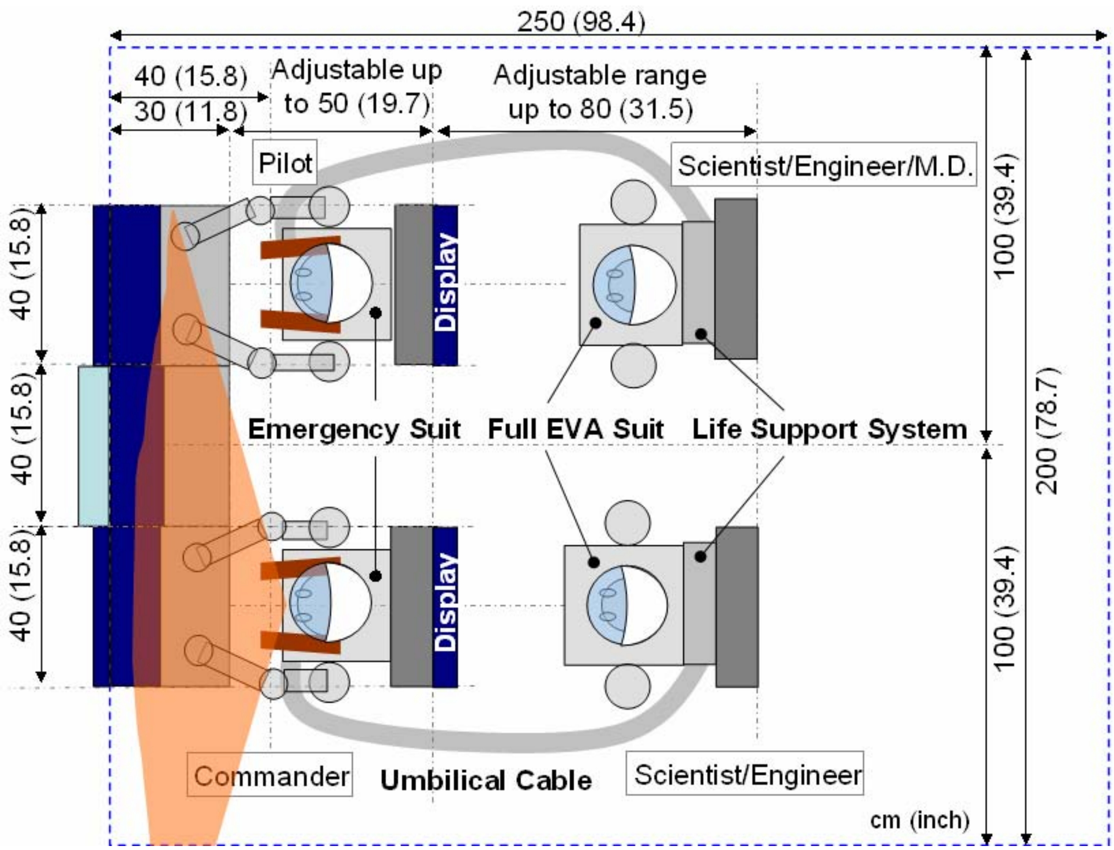


Figure 26. Lunar lander external cameras

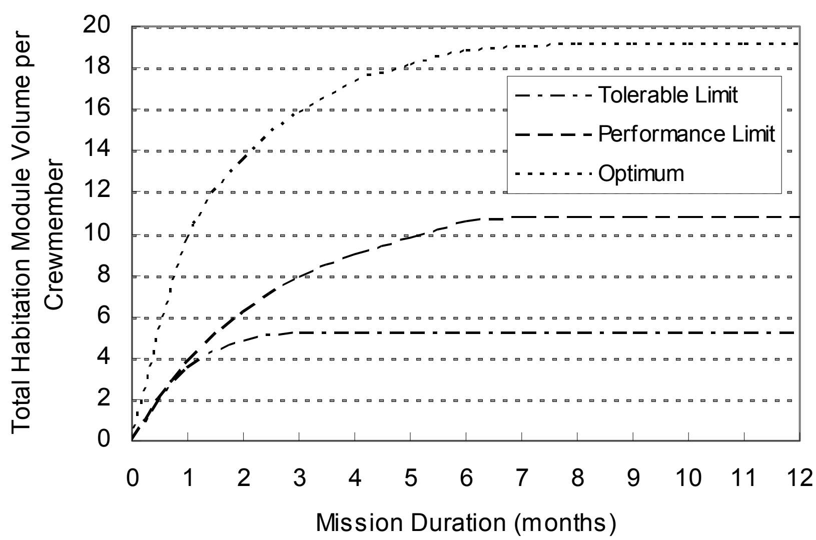


(a) Side view of the seat

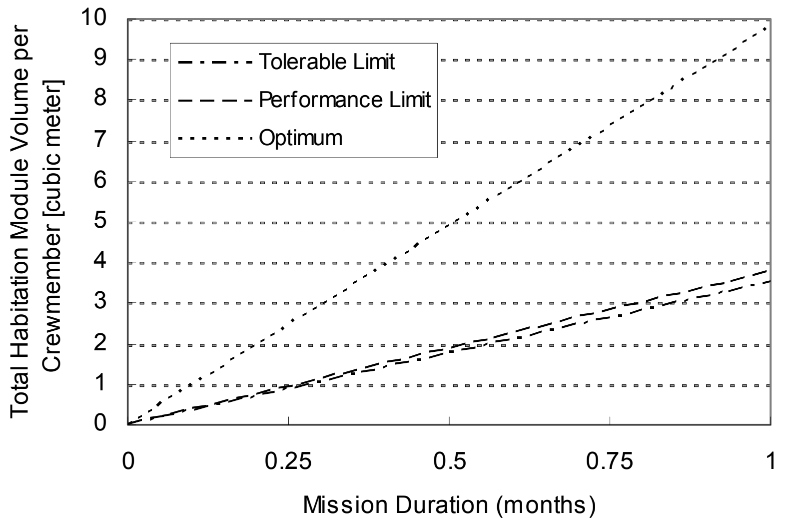


(b) Top view of the cockpit

Figure 27. The cockpit Layout of the Lander



(a) Zero to twelve - month missions



(b) Zero to one - month missions

Figure 28. Relation between mission duration and recommended volume of habitation module

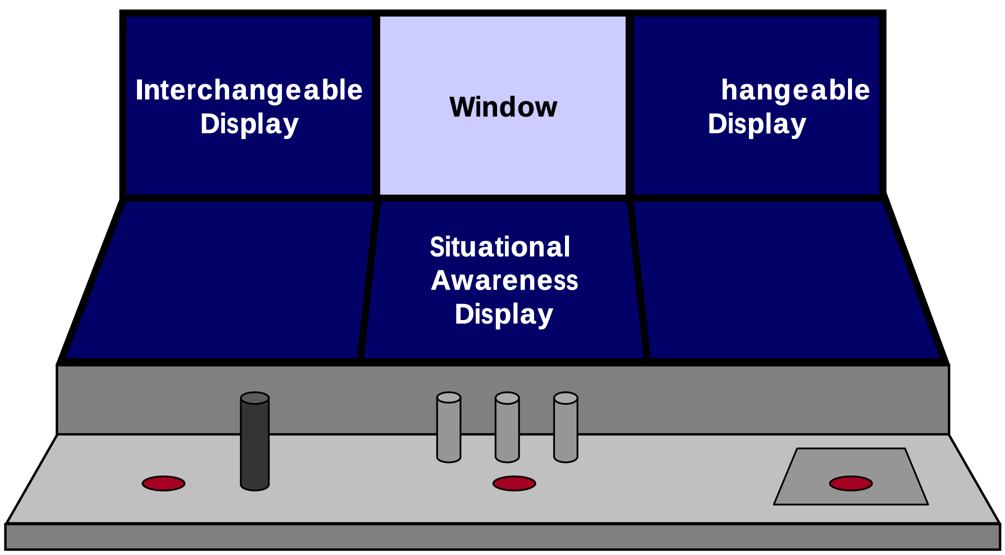


Figure 29. Display layouts

In addition to the three main displays on the lower lane, the interchangeable displays on top can show any of the other displays or external camera views on demand.

Table 18. Cabin environment within lunar lander

CABIN ENVIRONMENT	
Total pressure (kPa)	99.9 -102.7
p_{O2} (kPa)	19.5 - 21.3
p_{CO2} (kPa)	0.4 (max)
Temperature (°C)	18.3 -26.7
Dew Point (°C)	5.0 -16.0
Ventilation rate (m/s)	0.08 - 0.2

Table 19. Crew metabolic consumption and waste output rates

CREW NEEDS	(kg/person/day)
Food solids	0.62
Water	3.53
Oxygen	0.84
SUM	4.99
CREW OUTPUT	
solid waste	0.11
liquid waste	3.87
CO_2	1.00
SUM	4.98

Table 20. Roles during lunar landing

<table>
<tr><th>COMMANDER</th><th>SYSTEM ENGINEER</th><th>DOCTOR/SCIENTIST/ENGINEER</th></tr>
<tr><td>▪ Take over the automation once landing site in FOV
▪ Redesignate the landing site
▪ Make decision to abort</td><td>▪ Monitor the subsystem function
▪ Identify and report subsystem malfunction to the commander
▪ Make suggestions for landing site redesignation
▪ Communicate with ground</td><td rowspan="2">▪ Monitor the subsystem function
▪ Maintain SA</td></tr>
<tr><td colspan="2">▪ Understand the position of the lander over the Moon
▪ Monitor the automated flight to maintain situational awareness (SA defined in testing section)</td></tr>
</table>

8.4 Operations Team Appendices

8.4.1 Full Nominal Procedure

Time from PDI (s)	Event	Commander	Pilot	Computer	MCC
–					
–					
–					
–					
–					
–					Go/NoGo DOI
–	DOI				
–		Don helmet and gloves	Don helmet and gloves	Display landing area from camera views	
–					
–					
–	Landing Radar On			Turn on LR	
–					Uplink to lander
–					Go/NoGo PDI
–		Tell computer to go ahead with PDI		Abort guidance system initiated	
0	PDI	Confirm PDI Initiation		Initiate PDI	Begin 1st burn
5				Display that currently in PDI Mode	Acknowledge
10		Confirm LPD enabled		Landing point designator enabled	
15					
20				Check convergence of LR to inertial	
25				Make weighted corrections to navigation	
30				Present status of LR data at 14000 ft.	
35			Verify LR data good		Confirm LR data good
40					
45		Check Landing Site	Monitor systems		
50				Confirm critical systems nominal	Acknowledge telemetry
55		LPD using camera views			
60					
65					
70					
75					
80					
85					
90					
95					
100					
105					
110					
115				Confirm critical systems nominal	Acknowledge

120	Begin 2nd Burn				Begin 2nd Burn
125		Confirm 2nd Burn Begun			Acknowledge
130					
135				Display that currently in 2nd Burn	
140					
145					
150					
155					
160					
165					
170					
175					
180			Confirm critical systems nominal		
185					
190					
195					
200					
205				Check LR data	
210				Check convergence of LR to inertial	
215				Make weighted corrections to navigation	
220				Present status of LR data at 14000 ft.	
225			Verify LR data good		Confirm LR data good
230					
235			Confirm critical systems nominal		
240					
245					
250					
255					
260					
265					
270					
275					
280					
285					
290	Begin Hover			Initiate Hover Phase	Begin Hover
291			Monitor rate of descent	Display that currently in Final Landing Phase	
292					
293					
294			Callout Rate of Descent		
295		Control rate of descent if needed			
296					
297			Monitor attitude		
298					
299			Callout attitude		
300		Control attitude			

		if needed			
301					
302		Confirm Hover Phase to MCC	Callout attitude		
303					
304		Confirm Hover Phase to MCC		Initiate Attitude Hold	
305		Verify landing site in camera view is landing site out window			
306					
307					
308					
309					
310			Confirm critical systems nominal		Check landing site acquisition
311					
312					
313					
314					
315					
316					
317					
318					
319					
320	LPD	LPD using window view			
321					
322					
323					
324					
325					
326					
327					
328					
329					
330			Callout Rate of Descent		
335					
340			Confirm critical systems nominal		
345					
350					
355					
360			Callout Rate of Descent		
365			Callout Rate of Descent		
370	Touchd own	Lunar Contact Checklist	Lunar Contact Checklist		
		Stay/NoStay			Stay/NoStay

8.4.2 Failure Modes & Effects Analysis Results

Item	Failure mode	Potential Effects of Failure	Phase of Failure	Critical Failure ?	Operational Procedures for Recovery	Operational Procedures for Repeat of Failure after Recovery	Design Recomm-endations
Descent engine	Descent engine flame-out	not enough thrust to land, LOM, LOC	1st burn	Yes	Attempt restart to land; compensate with other engines/RCS; ascent stage abort to orbit	Attempt restart to descent stage abort; ascent stage abort to orbit	
	Descent engine flame-out	not enough thrust to land, LOM, LOC	2nd burn	Yes	Attempt restart to land; compensate with other engines/RCS; ascent stage abort to orbit	Attempt restart to descent stage abort; ascent stage abort to orbit	
	Descent engine flame-out	hard landing, LOM, LOC	Hover	Yes	Attempt restart to land; compensate with other engines/RCS; ascent stage abort to orbit	Attempt restart; crash-land	
	Descent engine throttled own failure	LOM	1st burn	Yes	Compensate with shutdown of other engines/RCS; ascent stage abort to orbit		
	Descent engine throttled own failure	LOM	2nd burn	Yes	Compensate with shutdown of other engines/RCS; ascent stage abort to orbit		
	Descent engine throttled own failure	LOM, hard landing	Hover	Yes	Compensate with shutdown of other engines/RCS; ascent stage abort to orbit		
	Descent engine fire	LOM, LOC	Any	Yes	Ascent stage abort to orbit		
Ascent engine	Probabl e ascent engine function ality loss	LOM	1st burn	Yes	Descent stage abort to orbit; abort to landing		Note: probable functional-ity loss may include mechan-ical or electrical redundant failures
	Probabl e ascent	LOM	2nd burn	Yes	Descent stage abort to orbit;		

	engine functionality loss				abort to landing		
	Probable ascent engine functionality loss	LOM	Hover	Yes	Abort to landing		Must use ascent stage of backup lander on surface
	Ascent engine fire	LOM, LOC	1st burn	Yes	Descent stage abort to orbit, bailout		Make plans to allow for CEV to automatically recover spacewalkers
	Ascent engine fire	LOM, LOC	2nd burn	Yes	Descent stage abort to orbit, bailout		Wear pressure suits on descent
	Ascent engine fire	LOM, LOC	Hover	Yes	Abort to landing		Minimize this risk
RCS	RCS motor loss	reduced landing safety and accuracy, LOM	Any	Yes	Compensate for loss with descent engine and functioning thrusters		
	RCS motor on	reduced landing safety and accuracy, LOM	1st or 2nd burn	Yes	Compensate for loss; ascent stage abort to orbit		
	RCS motor on	reduced landing safety and accuracy, LOM	Hover	Yes	Compensate for loss; ascent stage abort to orbit		Check maximum rate of pitch/roll/ yaw to minimize sudden crash risk
GN&C	GN&C software failure	LOM	Any	No	Increase voice communication to ground, land manually		
	IMU failure	LOM	1st burn	No	Descent stage abort to orbit		
	IMU failure	Reduced landing accuracy, LOM	2nd burn	No	Pitchup early to begin Hover phase and land manually		
	IMU failure	Reduced landing accuracy, LOM	Hover	No		Land manually	
Radar	Landing radar failure	Reduced landing accuracy, LOM	1st burn	No	Rely on IMU and voice to ground		
	Landing radar	Reduced landing	2nd burn	No	Rely on IMU and voice to ground		

	failure	accuracy, LOM				
	Landing radar failure	Reduced landing accuracy, hard landing?	Hover	No	Land manually	
Camera	External Vision System failure	LOM	1st burn	No	Pitchup early	
	External Vision System failure	LOM	2nd burn	No	Pitchup early	
	External Vision System failure	LOM	Hover	No	Land manually	
Electrical System	Wiring failure	LOM, LOC	Any	Yes	Yell at designers over radio	Include redundant wiring and add automatoic switchover for electrical systems
ECLSS	CO2 scrubber failure	crew incapacitation, LOM	1st burn	No	Descent stage abort to orbit; ascent stage abort to orbit	Wear pressure suits
	CO2 scrubber failure	crew incapacitation, LOM	2nd burn or Hover	No	Abort to landing	
	Thermal system failure	LOM	Any	No	Abort to landing, decrease mission duration	Check suvivability of lander in cold storage
	Humidity control system failure/humidity increase	LOM	Any	No	Abort to landing	Need to have crew in pressure suits; check survivability of electronics
	O2 production failure/pressure loss	crew incapacitation, LOM, LOC	1st burn	Yes	Descent stage abort to orbit; ascent stage abort to orbit	Need to have pressure suits, see CO2 scrubber failure
	O2 production failure/pressure loss	crew incapacitation, LOM, LOC	2nd burn	Yes	Abort to landing	
	O2 production failure/pressure	crew incapacitation, LOM, LOC	Hover	Yes	Abort to landing	

	loss					
Communication System	Communications transmission loss	nuisance	1st burn	No	Attempt to restore communications through CEV; land independently	
	Communications transmission loss	nuisance	2nd burn	No	Land independently	
	Communications transmission loss	nuisance	Hover	No	Land independently	
Command and Control System	Command software failure	LOM, LOC	1st burn	Yes	Attempt software patch; descent stage abort to orbit; ascent stage abort to orbit	Have backup software available, use completely different set of software in abort situation
	Command software failure	LOM, LOC	2nd burn	Yes	Attempt software patch; descent stage abort to orbit; ascent stage abort to orbit	
	Command software failure	LOM, LOC	Hover	Yes	Attempt software patch; ascent stage abort to orbit; abort to landing	
Cabin	Cabin fire	LOM, LOC	Any	Yes	Extinguish fire; abort to landing, bailout, run	Need extremely capable fire-suppresion system, also pressure suits
User Interface System	LCD screen failure: color bias, flicker, blank screen, etc.	Nuisance	Any	No	switch a different screen to the appropriate display needed	
	input devices failure	Nuisance	Any	No	switch to different set of input devices and screen	

8.4.3 Flight Rules

General Rules for Landing

- The LSAM must retain redundant capability in critical systems throughout the landing sequence; otherwise the mission must be aborted.
- A mission segment cannot begin if communication with Mission Control is lost, but shall continue in the even of such a loss during execution.

Authority Rules

- Mission Control has final authority for Go/No Go for initiating any burn.
- Mission Control may suggest abort decisions, but the Mission Commander must make the final decision to abort.
- The Mission Commander has final authority over touchdown site redesignation.
- The Mission Commander may take over manual control at any time without prior approval from Mission Control.
- If the Mission Commander becomes incapacitated for any reason, the LSAM Pilot shall assume his place.

Mission Segment Rules

DOI

- Landing gear must be fully extended and locked prior to DOI
- DOI will be aborted for any of the following:
  - Attitude deviations greater than TBD degrees (Number to come from GN&C)
  - Rates greater than TBD degrees (Number to come from human factors)
  - Overburn of TBD m/s (Number to come from GN&C)

Descent Orbit Coast

- Any residual rates must be nulled
- LSAM orbit will be confirmed with Mission Control
- LSAM checkout must be completed 10 minutes prior to PDI
  - Additional orbits are acceptable to comply with this rule

PDI

- PDI will be initated automatically to assure accurate thrust vector alignment and spacecraft attitude
- Powered Descent will be aborted for any of the following:
  - Attitude deviations greater than TBD degrees (Number to come from GN&C)
  - Rates greater than TBD degrees (Number to come from human factors)
  - Uncorrected deviations outside the trajectory boundary

Hover/Touchdown

- Voice communications between the Mission Commander and LSAM Pilot have top priority
- Communications with Mission Control should be kept to a minimum and only utilized in an off-nominal situation
- An ascent stage abort to orbit will be performed for any of the following:
  - Rates greater than TBD degrees/second (Number to come from human factors)
  - Vertical velocity greater than TBD m/s (Number to come from structures group)
  - Horizontal velocity greater than TBD m/s (Number to come from structures group)

Post-Touchdown

- Completion of the safing procedure has top priority
- Ascent stage abort to orbit for any of the following:
    - Failure to shut down and safe descent engines
    - Failure to safe any other part of the LSAM

Failure Action Rules

Malfunction	Flight Phase	Action to be taken
Loss of 1 Descent Engine	DOI	Continue, attempt to restart.
	Powered Descent Burn 1	Continue, attempt restart.
	Powered Descent Burn 2	Continue, attempt to restart, abort to landing
	Hover	Continue, abort to landing
Malfunction	**Flight Phase**	**Action to be taken**
Loss of 2 Descent Engines	DOI	Continue, attempt to restart. Must be remedied prior to PDI
	Powered Descent Burn 1	Attempt to restart; abort to orbit of unsuccessful
	Powered Descent Burn 2	Continue, attempt to restart, abort to landing
	Hover	Continue, abort to landing
Malfunction	**Flight Phase**	**Action to be taken**
Descent Stage Fire	Any	Ascent stage abort to orbit
Malfunction	**Flight Phase**	**Action to be taken**
IMU Failure	DOI	Continue, use DSN. Must be remedied prior to PDI
	Powered Descent Burn 1	Continue, use DSN if possible, otherwise abort to orbit
	Powered Descent Burn 2	Continue, use landing radar if possible, otherwise abort to orbit
	Hover	Continue, abort to landing (under manual control if necessary)
Malfunction	**Flight Phase**	**Action to be taken**
Probable Loss of Ascent Engine	DOI	Continue; troubleshoot during descent orbit coast. Abort to orbit if not remedied
	Powered Descent Burn 1	Descent stage abort to orbit
	Powered Descent Burn 2	Descent stage abort to orbit until fuel is too low for descent stage abort, then abort to landing
	Hover	Abort to landing
Malfunction	**Flight Phase**	**Action to be taken**
LSAM life support	DOI	Continue, remedy during descent orbit coast. Must be remedied for PDI.
	Powered Descent Burn 1	Continue, troubleshoot if possible
	Powered Descent Burn 2	Continue, abort to landing
	Hover	Continue, abort to landing
Malfunction	**Flight Phase**	**Action to be taken**
Command Software	DOI	Continue to descent orbit and troubleshoot if possible, otherwise abort to orbit
	Powered Descent Burn 1	Attempt to troubleshoot,

		otherwise abort to orbit
	Powered Descent Burn 2	Abort to landing if possible (manual if needed) otherwise abort to orbit
	Hover	Abort to landing (manual if needed)
Malfunction	**Flight Phase**	**Action to be taken**
Power System Failure	DOI	Switch to backup, continue to descent orbit. Troubleshoot as needed before PDI
	Powered Descent Burn 1	Switch to backup, continue
	Powered Descent Burn 2	Switch to backup, continue
	Hover	Switch to backup, abort to landing