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GN&C System Design in a Mass Reduction Exercise

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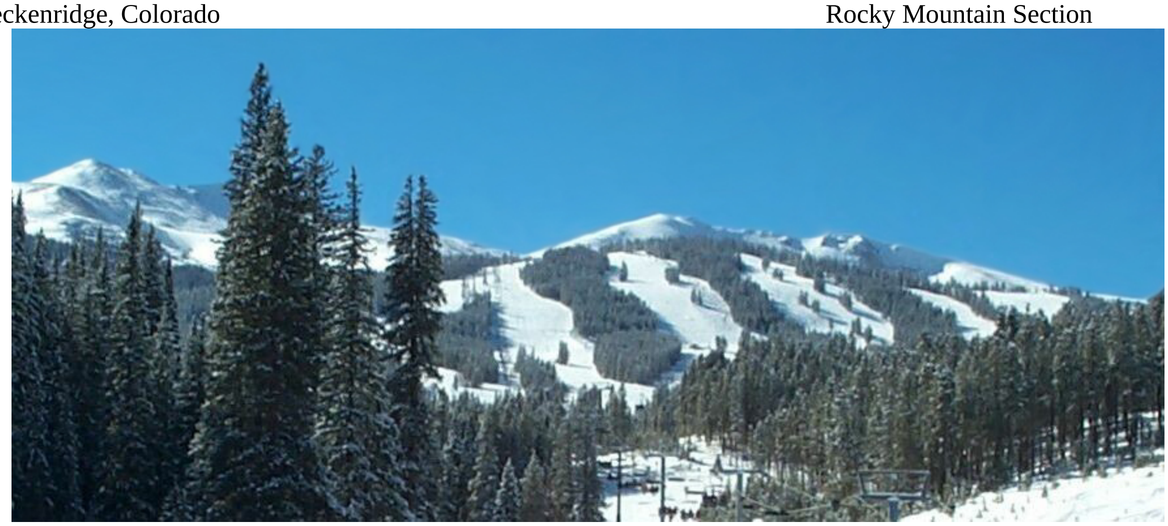

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GUIDANCE, NAVIGATION, AND CONTROL SYSTEM DESIGN IN A MASS REDUCTION EXERCISE

Timothy Crain, Michael Begley, Mark Jackson, and Joey Broome

Early Orion GN&C system designs optimized for robustness, simplicity, and utilization of commercially available components. During the System Definition Review (SDR), all subsystems on Orion were asked to re-optimize with component mass and steady state power as primary design metrics. The objective was to create a mass reserve in the Orion point of departure vehicle design prior to beginning the PDR analysis cycle. The Orion GN&C subsystem team transitioned from a philosophy of absolute 2 fault tolerance for crew safety and 1 fault tolerance for mission success to an approach of 1 fault tolerance for crew safety and risk based redundancy to meet probability allocations of loss of mission and loss of crew. This paper will discuss the analyses, rationale, and end results of this activity regarding Orion navigation sensor hardware, control effectors, and trajectory design.

INTRODUCTION

This paper summarizes the evolution of the Orion Guidance, Navigation, and Control (GN&C) subsystem from the system design review (SDR) baseline design to the preliminary design review (PDR) point of departure (POD) design. The specific roles of Orion for translation authority and GN&C execution encompass ascent abort, Low Earth Orbit (LEO) Rendezvous, Proximity Operations, and Docking (RPOD), Low Lunar Orbit (LLO) maintenance, contingency LLO RPOD, trans-Earth insertion, trans-Earth trajectory correction maneuvers (TCMs), and Entry, Descent and Landing (EDL). An excellent overview of these roles was provided by Tamblyn et. al[i]. At the time of Orion SDR the GN&C subsystem was designed to satisfy fault tolerance and mission capability requirements that far outstripped the Apollo system functionality (see Table 1). The SDR design was a fusion of the NASA and LM pre-contract design concepts utilizing commercially available components, unit redundancy, and comprehensive trajectory designs to fulfill these requirements.

The Orion SDR baseline design was found to exceed mass allocations by ~20% (to be refined in final version). Mass for the total vehicle was constrained in three ways by effective payload weight (EPW) of the launch vehicle, mass at the ARES I separation target, and mass at docking with the lunar lander in LEO. Mass for the Crew module was further constrained by landing loads and launch abort system (LAS) performance. An intensive mass reduction effort was executed at the subsystem level to reduce vehicle mass to meet these constraints. . The objective was to not only be within the Constellation mass allocation but also to create a mass reserve in the Orion PDR point of departure (POD) vehicle design prior to beginning the PDR analysis cycle. Tactics used in this effort included requirements modification, and design improvements.

The Orion GN&C subsystem design team shares responsibility for the navigation sensor hardware and the reaction control system (RCS) hardware with the avionics and propulsion subsystem design teams respectively. The GN&C team also performs flight dynamics analyses to aid the design process for other vehicle subsystems such as thermal control and power. Analyses that directly impacted vehicle mass included navigation sensor reliability, entry trajectory analysis, water and land landing analysis, and orbital trajectory analysis. For the purposes of mass and power reduction, the GN&C team had 4 options available: (1) reduce the count or mass of navigation sensor components, (2) reduce the count or mass of RCS effector strings, (3) reduce the propellant needed to achieve Constellation mandated mission objectives, and (4) reduce the thermal protection system mass needed for guided EDL.

The remainder of this paper describes the processes used by the GN&C team to determine potential mass savings from candidate requirements changes, and to improve GN&C subsystem and vehicle design to minimize mass. The first section describes changes to the GN&C component design to decrease subsystem mass and reduce weight associated with power and cooling requirements, Next, trajectory analyses are described which yielded information about the sensitivity of propellant mass to mission capability. Finally, analyses in the entry descent and landing (EDL) phase are described. These analyses were aimed at minimizing mass required for the thermal protection system (TPS), the CM structure, and entry RCS propellant, as well as determining the mass price for the capability to land on land.

Table 1: Driving GN&C Orion Vehicle Requirements at SDR

Reqt. ID	Paraphrased Requirement
SDR-1	Orion shall be two fault tolerant to catastrophic hazards
SDR-2	Orion shall be single fault tolerant for loss of mission
SDR-3	Orion shall land at sites in the continental US within 5 km of the target.
SDR-4	Orion shall support crew return from 7 day sortie missions on 100% of the lunar surface.
SDR-5	Orion shall return the crew to the Earth from lunar orbit in less than 142 hours.
SDR-6	Orion shall perform automated RPOD

GN&C SUBSYSTEM COMPONENT REDESIGN

A key GN&C SDR design tenet was to keep Fault Detection, Isolation, and Recovery (FDIR) as simple as possible by using sufficient redundancy to protect both for hard failures (components which fail completely in an easily identifiable manner) and soft failures (subtle failures where a dilemma case exists). However, when the Orion SDR design was found to exceed the mass allocation for ARES I and ARES V all subsystems were asked to re-optimize with component mass and steady state power (indirectly leading to thermal control and power system mass) as primary design metrics

Driving Navigation Requirements

The driving requirements on Orion navigation from the Constellation Architecture Requirements Document (CARD) and Orion Systems Requirements Document (SRD) are provided in paraphrased form in Table 2. SDR-7 (crew return during loss of comm.) and SDR-12 (expedited TEI preparation navigation) both led to an optical navigation capability during lunar and transit operations. Most of the other requirements specify the need for absolute inertial navigation and target relative navigation. In particular, SDR-6 (automated RPOD) is one of several requirements that drove the sensor hardware count needed for relative navigation.

Table 2: Orion Vehicle Requirements Driving Navigation

Reqt. ID	Paraphrased Requirement
SDR-7	Orion shall return the crew to Earth during loss of communications with Earth mission support.
SDR-8	Orion shall independently determine integrated stack trajectory, attitude, and attitude rates.
SDR-9	Orion shall function as the maneuvering vehicle during contingency RPOD operations with the LLV in LLO.

SDR-10	Orion shall function as the maneuvering vehicle during RPODU operations with the ISS.
SDR-11	Orion shall perform proximity operations and docking with the LLV with inertial rates up to 0.4 deg/sec per axis.
SDR-12	Orion shall calculate navigation solutions in support of TEI in less than 6 hours.
SDR-13	Orion shall perform relative navigation for RPOD when functioning as the maneuvering vehicle.
SDR-14	Orion shall determine the relative position and velocity during PODU operations when the Orion is the maneuvering vehicle.
SDR-15	Orion shall determine the relative attitude and attitude rate when the Orion is the maneuvering vehicle within 15.2 m (50 feet) of the target vehicle.
SDR-16	Orion shall function as the maneuvering vehicle during undocking and departure proximity operations from the target vehicle at any attitude, in case of an emergency.
SDR-17	Orion shall perform automated RPODU.
SDR-18	Orion shall perform a guided entry that results in landing within 5 km (2.7 nmi) of the intended target at a designated CONUS landing site.
SDR-19	Orion shall provide a dissimilar IMU for the Emergency Entry Mode.
SDR-20	Orion shall provide a hardware altimeter for crew situational awareness needed for manual activation of altitude sensitive events.

The Orion SRD (designated 606A) design navigation sensor suite (see Table 5) satisfied all of the driving requirements listed above. Four integrated GPS/INS units (listed collectively under Inertial Measurement Unit) were used to support dynamic flight in all phases, autonomous crew return in Earth vicinity, and precision navigation updates during EDL. Four units provided 100% 2-fault FDIR via a simple voting scheme similar to what is used in the Space Shuttle today. In this approach, the first failure is identified when a single INS/GPS disagrees with the remaining three beyond a threshold of acceleration or angular rate. The same scheme is used for the remaining three units when a single component exceeds a difference threshold with respect to the other two. Hence, two failures can be quickly detected and isolated by a simple threshold check and vote with the same software. Three rather than four star trackers were provided given that they are not utilized during dynamic flight and time exists to detect a failed star tracker by comparison to the other star tracker as well as INS gyroscope data. The star tracker also provides target-bearing measurements for RPOD and image “grabs” for processing in cis-lunar optical navigation or surface feature tracking in LLO by processing in the redundant Vision Processing Units (VPUs). Note that the primary inertial navigation mode for Orion is based on radiometric ground tracking during nominal mission operations. A backup rate sensor and pressure transducer were included to satisfy the requirements of a dissimilar EDL navigation capability.

The relative navigation system consisted of the three star trackers providing target bearing, 2-way S-band ranging on the communications system omni-directional antennas, two centerline optical cameras and two Vision Navigation Sensor (VNS) sensors. The 2 VNS and one of the centerline cameras are mounted on the inside of the docking port hatch after launch, prior to transitioning from rendezvous to proximity operations at terminal phase initiation (TPI) (see Figure 1). The second centerline camera would serve as a cold backup that could be installed by the crew if necessary. The overall utility of relative navigation sensors from the end of rendezvous through docking is illustrated in Figure 2. The VNS is actually a specification based upon a realistic assessment of industry capabilities for a sensor providing 3-DOF range and bearing from pre-TPI to docking and 6-DOF relative position and attitude for final approach and docking. These capabilities have been demonstrated individually in the Space Shuttle Trajectory Control Sensor (TCS) and a number of recent RPOD technology demonstrators such as DART, XSS-11, and Orbital Express.

Based upon GN&C team experience with both piloted and robotic RPOD missions, a functional VNS and optical camera were deemed critical for mission success per Table 3. With crew available for monitoring of sensor and trajectory data, a single VNS would suffice for 1FT mission success. However, automated operations require 2 VNS for detection of erroneous data via sensor direct comparison. The proximity operations and docking trajectories are designed with breakout capability along the entire trajectory without the use of a relative navigation sensors, therefore closure with the target vehicle could be aborted with a breakout to a hold point for assessment upon detection of a VNS mismatch beyond tolerances.

Figure 1: Orion Transition from Rendezvous to Proximity Operations

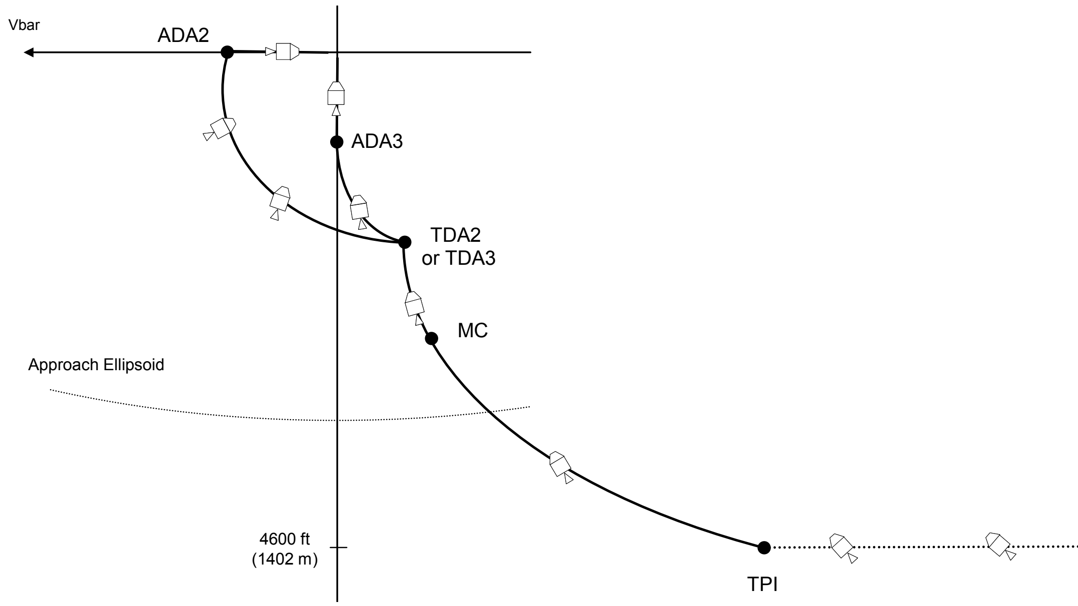

Figure 2: Relative Navigation Sensor Usage

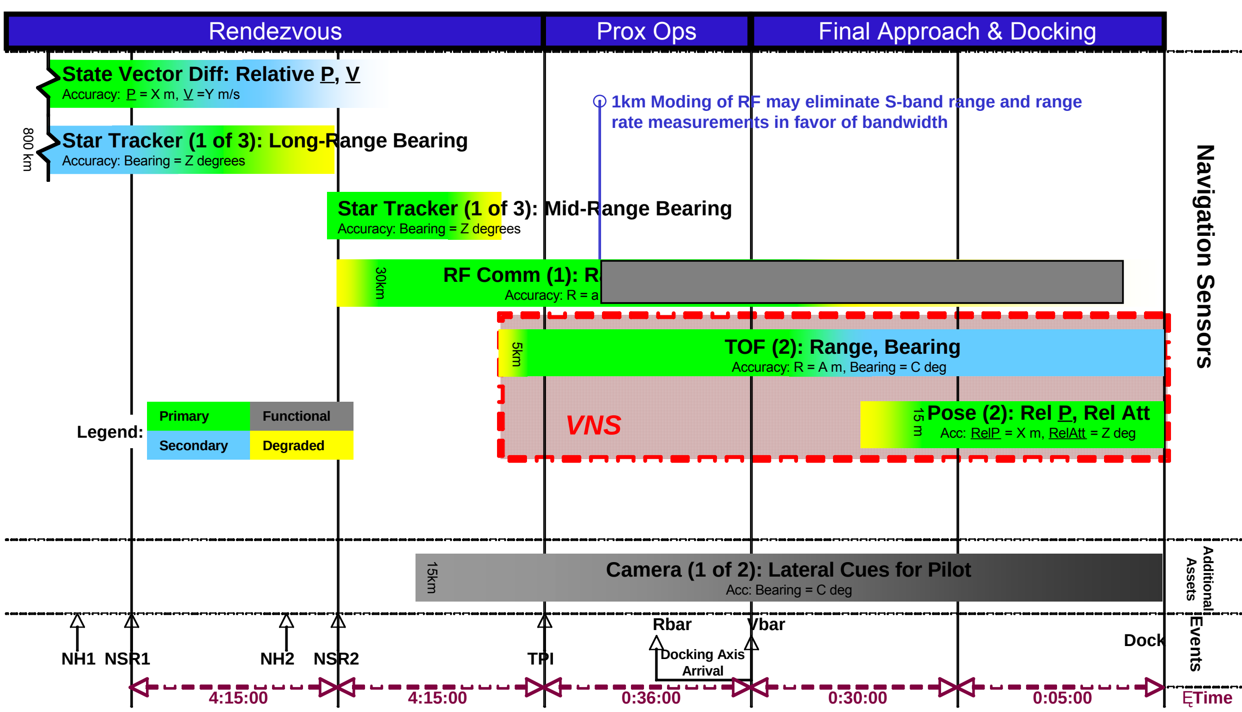

Table 3: Decomposition of RPOD Requirements to Sensor Capability

Phase	Bearing	Range	Range-rate	Pose	Comments
Rendezvous	**X**	**X**	**(*)**		**Bearing & range needed to support TPI maneuver. Bearing-only has been demonstrated, but add'l propellant needed to correct associated targeting errors**
Prox Ops	**X**	**X******	**(*)**		**Range and bearing allow the nav system to maintain a collision avoidance trajectory and acquisition of the docking axis**
Final Approach & Docking	**X**	**X**	**X**	**May be**	**Range dependent moding and PCT initiation. Allows monitoring of closing rate, with target centered, during final approach and docking**

During the initial phase of the mass reduction activity, the Orion project directed the development of a "zero-baseline" vehicle (ZBV) design with the minimal capability to execute a lunar mission with a modified set of driving requirements as paraphrased in Table 4. The navigation sensor compliment was modified to be 2 IMUs, 2 separate GPS receivers, 2 star trackers, 1 VNS, 1 optical camera, and 1 VPU as illustrated in the ZBV column of Table 5. The backup EDL navigation hardware was dropped from the vehicle and a strict hard-failure redundancy approach allowed for the reduction to 2 IMUs without protection for a dilemma resolution during dynamic flight.

Table 4: Zero-Baseline Driving GN&C Orion Vehicle Requirements

Reqt. ID	Paraphrased Requirement
ZBV-1	Orion shall be one fault tolerant to catastrophic hazards
ZBV-2	Orion shall be zero fault tolerant for loss of mission
ZBV-3	TRADE: Orion shall land at sites in the continental US within 5 km of the target or at US coastal sites.
ZBV-4	TRADE: Orion shall support crew return from 7 day sortie missions on TBD% of the lunar surface.
ZBV-5	TRADE: Orion shall return the crew to the Earth from lunar orbit in less than 142 hours.
ZBV-6	TRADE: Orion shall perform automated RPOD

The ZBV was never intended to be a final design for Orion, rather it would create a mass overhead margin from which hardware could be added to the design in two rounds. During round 1, safety-critical hardware would be added back to make the vehicle acceptable per NASA and LM standards. During round 2, hardware would be added to increase mission success and crew support by utilization of any remaining mass margin while protecting for a managers' reserve of 1,000 kg (REFINE NUMBER) The transition to lower fault tolerance requirements during ZBV was augmented with a more strenuous development of probability of loss of crew (PLOC) and probability of loss of mission (PLOM) requirements for each subsystem derived from Constellation probability allocations to each vehicle in the architecture.

The GN&C team recommended the buyback of a third IMU to protect against first-failure dilemma cases during round 1. While probabilistic risk assessments showed the space-qualified navigation grade IMUs under

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* Range rate is not required in these phases but provides contingency and safety margins.
** Proximity operations range measurement can provide range-rate via differencing or Kalman filtering.

consideration would meet PLOC allocations for GN&C, there was concern that the dynamic flight phases of ascent and entry would stress the hardware in ways not previously experienced in robotic spaceflight. In fact, there were known subtle gyroscope compensation failure modes in terrestrial application IMUs during dynamic flight that could lead to a dilemma beyond the specifications of the component hardware. Therefore, a 3 IMU solution and direct compare voting scheme was preferred to more sophisticated skewed-axis detection of a first IMU failure. 4 IMUs were not pursued when the reliability assessment indicated that common cause failures were dominant and only a 4th dissimilar unit would effectively improve reliability.

The GN&C team recommended buyback of an additional VNS, cold-backup optical camera, and making the VPU internally single fault tolerant during round 2 based upon the fact that all Orion missions require docking for mission success. The original mass and power specifications for the VNS were based upon conservative assumptions from existing hardware from the Space Shuttle (TCS) and DART/Orbital Express (AVGS) in order to allow for flexibility in the LM selection of a VNS supplier. However, a technology challenge was accepted where the mass and power allocation for the VNS hardware was reduced to be equivalent to an order of magnitude to sensors known to be in development for upcoming relative navigation applications or as technology advancement demonstrators.

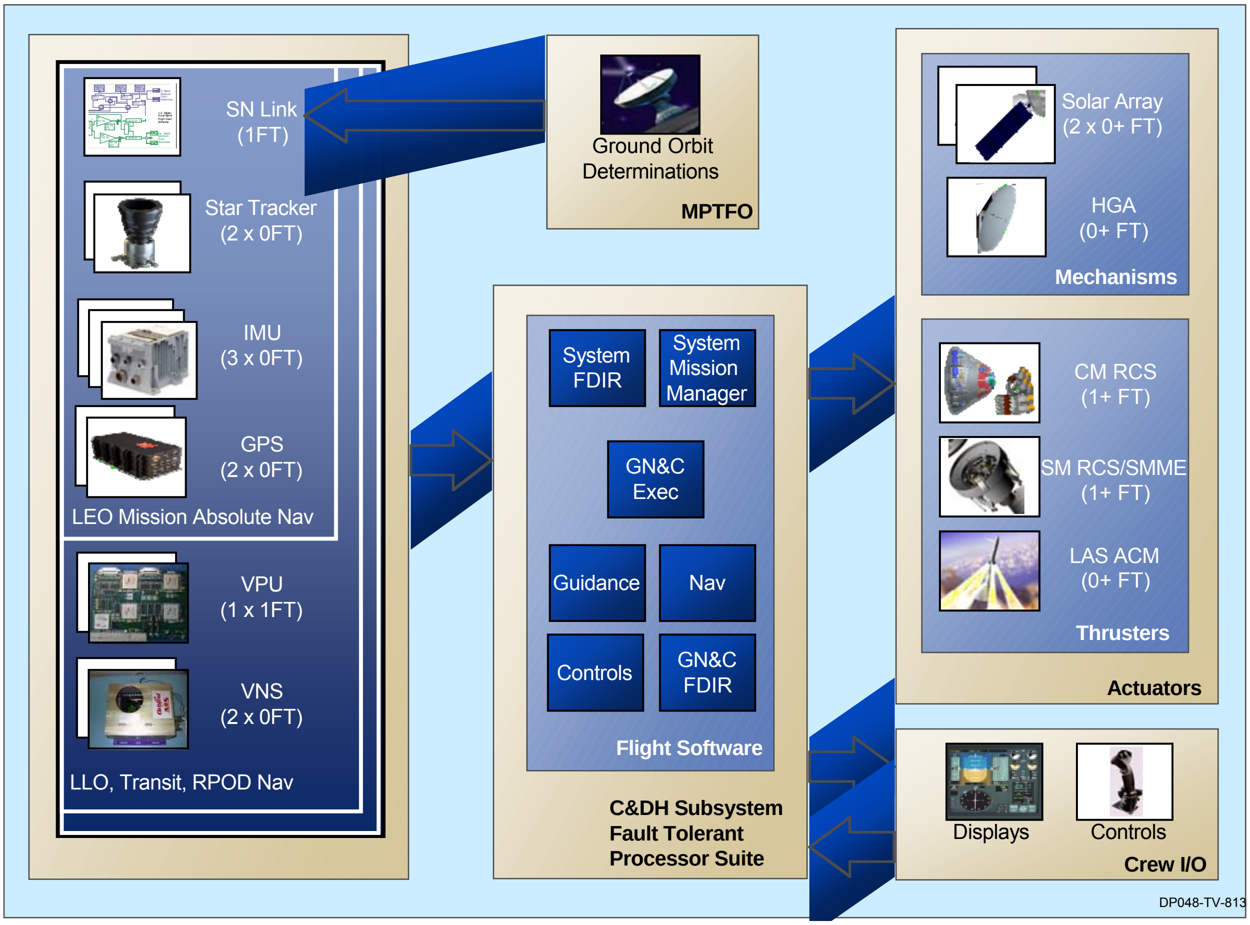
The issue of only 2 star trackers for attitude determination and optical navigation and the loss of the dissimilar navigation hardware for entry were recognized by the Orion project at the end of the mass reduction activity. Rather than delay the beginning of PDR design and analysis cycle, the decision was made to allocate mass from the managers' reserve for an Emergency Scenario Capability (ESC) incorporating manual flight capability, backup system design, and dissimilar navigation sensors for both primary and degraded modes of operation into a single package. The ESC is a design activity for the PDR analysis cycle.

The final navigation sensor compliment of the Orion PDR POD is graphically represented in Figure 3. This design retained 100% of the mission capability of the original design in 606A by applying a probabilistic approach to mission risk tempered the case of IMUs by considerations of flight criticality and in the case of VNS with considerations of current technology and mission success. The overall impact was a reduction of 35 kg (from 108 kg to 73 kg) and a peak power reduction of 321 Watts (from 655 W to 334 W). The total evolution of the GN&C navigation sensor compliment is provided in Table 5.

Table 5: Orion Navigation Hardware Evolution to PDR POD

H/W	CEV606A			Zero Base			PDR POD		
	Qty.	Mass (kg)	Power (ssw)	Qty.	Mass (kg)	Power (ssw)	Qty.	Mass (kg)	Power (ssw)
Inertial Meas. Unit	4	33.6	200	2	9.5	76	3	14.2	114
GPS				2	2.4	24	2	2.4	24
GPS Antennae	8	0.9		4	0.5		4	0.5	
GPS LNA	4	0.9	1	4	0.9	1	4	0.9	1
Star Tracker*	3	10.2	30	2	6.8	20	2	6.8	20
Backup Rate Sensor	1	0.9	3						
EEM Pressure Transducer	1	0.2	0.5						
Centerline Camera	2	5.4	40	1	2.7	20	2	5.4	20
Vision Processing Unit (VPU)	2	26.3	190	1	13.2	95	1	13.2	95
Vision Navigation Sensor (VNS)	2	22.7	190	1	11.3	30	2	22.7	60
VNS Bracket	1	2.3		1	2.3		1	2.3	
LLV Reflectors	5	4.5		5	4.5		5	4.5	
Comm. Ranging	2			2			2		
AbsNav Subtotal		47	235		20	121		25	159
RelNav Subtotal		61	420		34	145		48	175
Total		108	655		54	266		73	334
		Change from 606A			**-54**	**-389**		**-35**	**-321**

Figure 3: Orion PDR POD Sensor and GN&C Configuration



Driving SM Flight Control Requirements

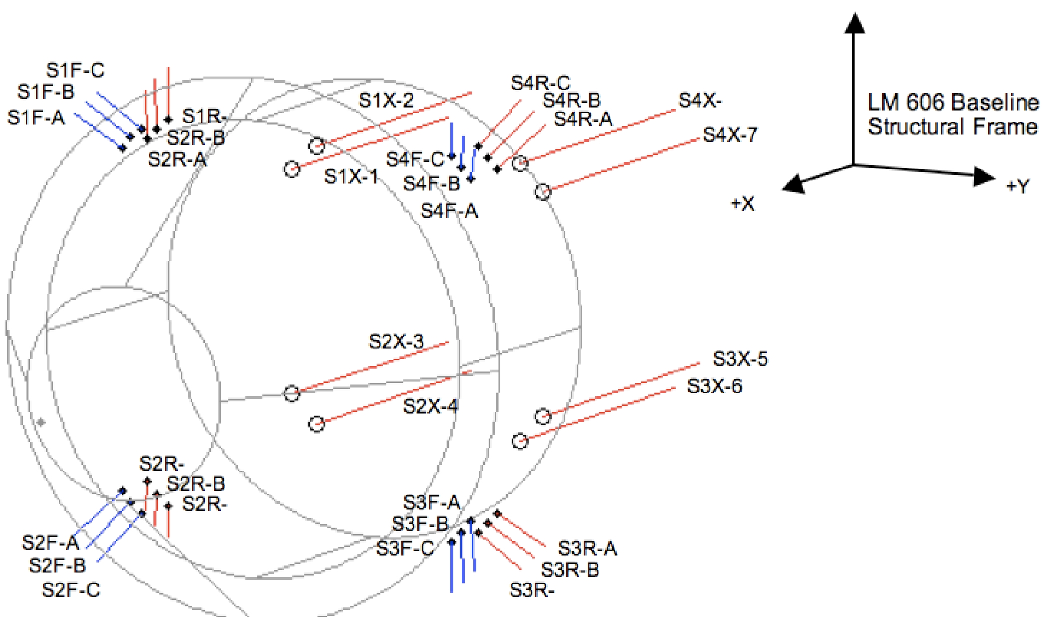
The driving requirements on Orion navigation from the CARD and Orion SRD are provided in paraphrased form in Table 6. The 606A RCS design was a three-string system deliberately sized to be able to meet all requirements via the functional operation of a single string[ii]. An isometric view of the RCS system is provided in Figure 4. The attitude control thrusters are laid out in a "block swap" geometry of 4 pods, each with six 110 N (25 lbf) thrusters (three strings – A, B, and C – with a forward-firing and rear-firing thruster per string). Pod 1 is located on the upper-starboard-side of the vehicle and the pod numbers increment up in a counter-clockwise direction, as seen from the front view. For each pod, the B string thruster is located 45 degrees from the X-Y plane; this angle is referred to as the "clock angle". For each pod, the A string thruster is roughly 3 degrees toward the X-Y plane, whereas the C string thruster is roughly 3 degrees away from the X-Y plane. Each forward and rear firing thruster has 45 degrees of separation from a unit vector in the X direction; this angle is referred to as the "cant angle". Each thruster is also skewed 12 degrees toward the X-Z plane, as referenced from the local radial vector. This angle is referred to as the "skew angle" and allows for roll authority. Recall that the local radial vectors for the A and C strings are roughly 3 degrees away from the B string local radial vector.

The GN&C team originally worked with the propulsion team to design the RCS system on the SM to provide full mission capability on a single string. Stress requirements such as docking with a disabled LLV in LLO, undocking from a disabled ISS, and some classes of ascent aborts could use multiple strings of RCS if necessary. However, analysis during the mass reduction exercise indicated that all of these stress cases could be accommodated via two strings with acceptable reduction in margin. Therefore, the redesign of the RCS system for PDR POD per the reduced fault tolerance guidelines was to eliminate the "C" string of the RCS system.

Table 6: Orion Vehicle Requirements Driving Flight Control

Reqt. ID	Paraphrased Requirement
SDR-21	Orion shall provide contingency attitude control of the mated Orion/LLV configuration.
SDR-22	Orion shall provide attitude control of the Orion/LLV ascent stage in LLO.
SDR-23	Orion shall function as the maneuvering vehicle during RPODU operations with the ISS.
SDR-24	Orion shall perform proximity operations and docking with the LLV with inertial rates up to 0.4 deg/sec per axis.
SDR-25	Orion shall function as the maneuvering vehicle during undocking and departure in any attitude in emergency.
SDR-26	Orion shall function as the maneuvering vehicle during RPODU operations with the LLV/Ares V EDS in LEO.
SDR-27	Emergency separation from ISS at 2 deg/sec.

Figure 4: SM Thruster Layout – CEV Isometric View



LUNAR MISSION TRAJECTORY ANALYSIS

At the time of the Orion SDR, approximately one-third of the total mass of the Orion vehicle on the launch pad was propellant to support attitude control through the RCS and translation for 1,492 m/s (4,895 ft/s) of delta-V across a lunar sortie or outpost mission. This value was selected through careful negotiation between the Orion mission analysis and GN&C teams and systems engineers handling the distribution of flight performance in the Constellation program. It was found that this value supported 7 day sorties to the vicinity of the lunar poles with appropriate dispersions for GN&C performance and expedited or auxiliary propulsion return trajectories but did not provide return missions from 100% of the lunar surface over 100% of the metonic cycle. The temporal coverage of the lunar surface for sortie missions was evaluated via mid-fidelity scan of mission epoch and landing site location. The Earth-return concept of operations is for the Orion vehicle to perform any plane change necessary to move to an overflight of the lunar lander for ascent. Therefore the relative declination of the hyperbolic departure asymptote from the Moon for a given return trajectory may dictate a subsequent plane change from the ascent orbit plane. By assuming 12 hour mission start intervals, a fixed transfer from LEO to Lunar surface timeline, surface locations in 5 degree increments of latitude and longitude, a fixed surface stay to ascent timeline, and an approximate optimization of the TEI sequence the temporal and spatial coverages in Figure 5 and Figure 6 were obtained. Figure 5 illustrates that approximately 64% of the surface of the moon was available 100% of the metonic cycle at the nominal tank loading for the PDR POD without any change to mission template above. The Orion PDR POD propellant tanks are actually oversized to have a capacity of supporting 1,716 m/s (5,306 ft/s) which is contrasted against the tank load in Figure 6. The end result of the mission design and propellant loading/sizing assessment effort was that Orion and Constellation agreed that the demonstrated mission capability with a nominal tank propellant loading with the options for "missionization" (shortening surface stays, extending loiter, choosing mission location and date) and tank overfilling if booster margin is realized provided an effective system for global lunar access. The details of this analysis are beyond the scope of this paper but will be presented in the context of a stand-alone paper at a future opportunity.

Figure 5: Temporal Coverage of the Lunar Surface for Standard Mission Template

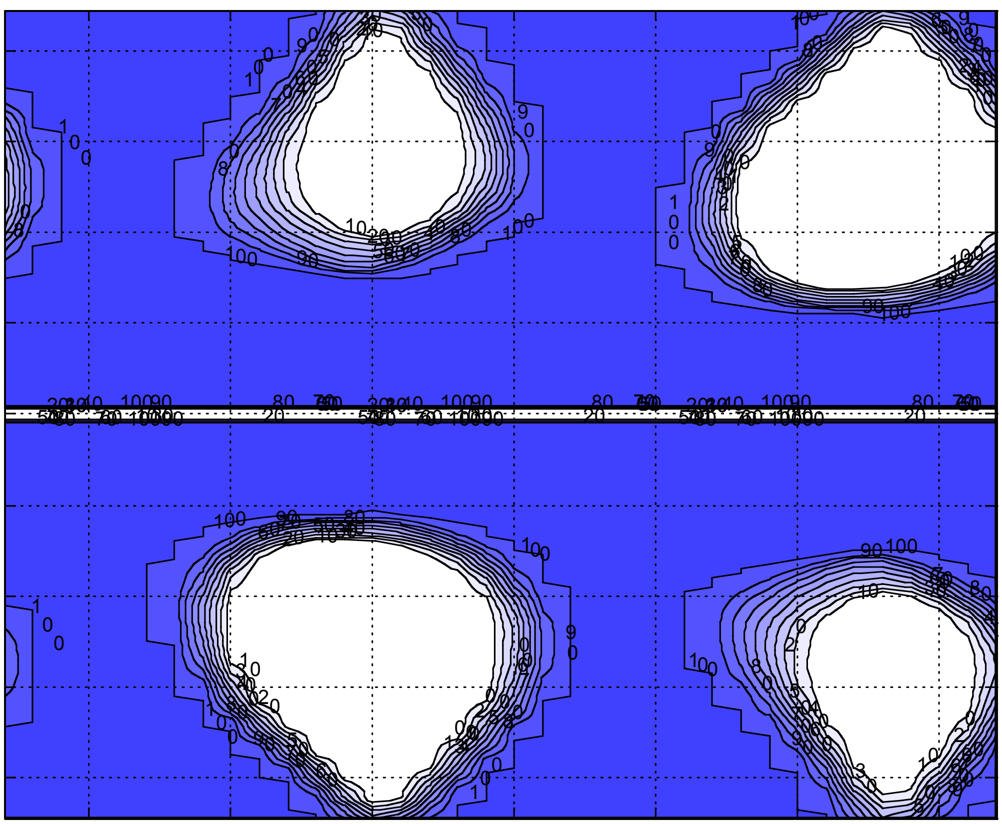
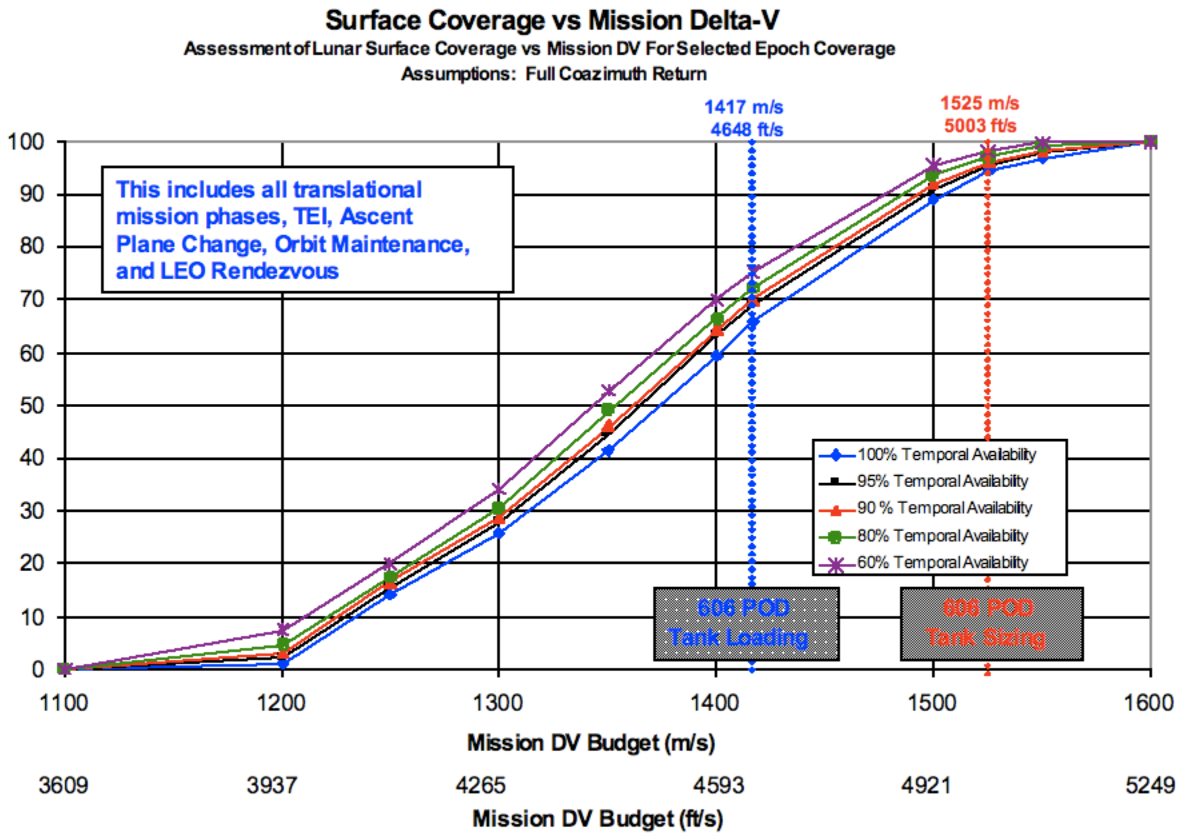


Figure 6: Surface Coverage vs. Delta-V for Tank Loading and Sizing



ENTRY DESCENT AND LANDING ANALYSIS

Driving EDL Requirements

Table 7 contains the driving requirements pertaining to the EDL flight phase.

Table 7: Orion Vehicle Requirements Driving EDL

Reqt. ID	Paraphrased Requirement
CA????	Orion shall skip?
	Land Landing
	5 km landing accuracy, 10km water
	Water Landing and sea states

The entry trajectory and the touchdown landing conditions have a large impact on Crew Module (CM) mass. These in turn are driven by fundamental vehicle requirements, such as the requirement to land on land or water, the required glide range, the crossrange capability, and the landing accuracy requirements.

Figure X is a simplified picture of a typical Lunar return entry profile. At Lunar return velocity, the spacecraft may extend its glide range by executing a “skip” maneuver, which includes a second exo-atmospheric phase as depicted. When the entry interface is close to the landing site, the CM may execute a direct entry, which starts at the segment labeled “6” in the figure. The direct entry is the only entry type used on low Earth orbit missions, including ISS docking missions.

Once the descent phase is complete, drogue chutes are deployed (point 7a in the figure) followed by main chutes. Touchdown structural loads are determined by descent rate, wind conditions, and touchdown conditions (land or water and sea state). CM components whose mass are effected by entry trajectory parameters include:

- The thermal protection system (TPS) consisting of the heat shield and backshell protection,
- The weight of the landing and recovery system (LRS) consisting of chutes and airbags (for land touchdown),
- Structural mass required to withstand touchdown loads
- RCS propellant for bank control and stabilization during the entry trajectory
- Ballast mass required to offset the CM center of mass to achieve a desired lift to drag ratio (L/D)

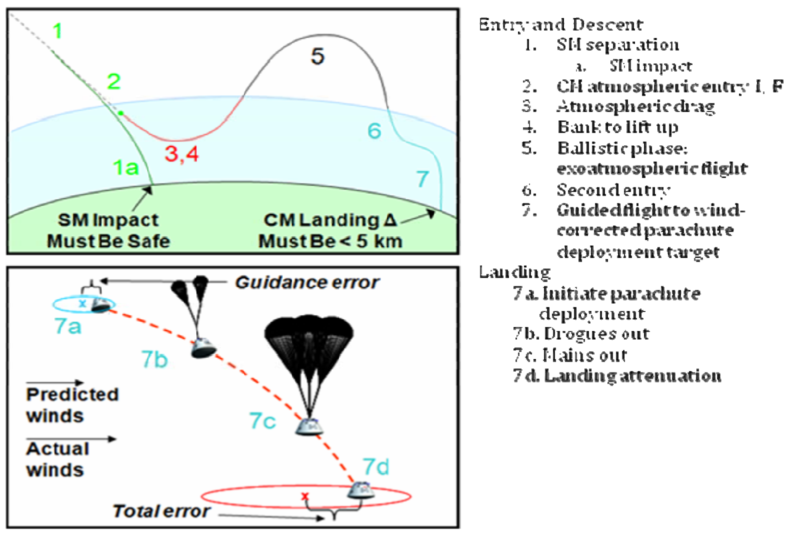


Figure X. Skip Entry Illustration

EDL Analysis Plan

In order to meet the CM mass allocation, two types of data were required. First data were needed to quantify the sensitivity of mass to requirements and requirement parameters. Once these relationships were known, the mass “cost” for each capability could be understood and traded, so that intelligent decisions could be made about whether or not to remove or reduce candidate requirements. The second type of data was the sensitivity to design parameters. These sensitivity data enabled mass-optimization of the design for a given set of requirements.

The primary means of determining sensitivities was through parametric analysis using time-domain Monte Carlo simulation. These analyses were conducted using the NASA ANTARES simulation [], a high fidelity, 6 DOF simulation with a full guided entry capability. Each Monte Carlo set consisted of 3000 simulation runs. Some of the more important sensitivities investigated in this study are listed in Table 8.

Table 8: EDL Sensitivity Examples

Examples of Sensitivities Investigated	
Capability footprint to L/D	Mass to entry azimuth
Mass (TPS, ballast and RCS propellant) to L/D	Propellant mass and accuracy to drogue deploy altitude
Mass (DV propellant) to L/D for ISS entry	Landing accuracy to main chute descent rate
Mass to entry range	Structural mass to sea state at water touchdown
Mass to initial flight path angle	Mass and landing accuracy to GPS availability

The first three entries deal with the lift to drag ratio (L/D). Lift is accomplished in a capsule by placing the center of gravity off the capsule longitudinal axis. This produces a trim condition (balancing of aerodynamic torques) at a non-zero angle of attack. Lift is thereby produced in the direction [OPPOSITE] the CG offset. The larger the CG offset, the greater the lift and the resulting L/D. Higher L/D means greater vehicle capability to reach landing sites – especially sites with large crossrange. However increasing L/D requires ballast to offset the vehicle COM from centerline. L/D also affects flight time for a given range – thereby affecting heat load and control propellant. For entries after ISS missions, the CM is required to perform a height raising maneuver to ensure separation from the SM. Reducing L/D may increase the magnitude of this maneuver, in turn increasing propellant usage.

Entry range from the entry interface condition (EI) to the landing site affects the time of flight and the lift profile commanded by guidance. Very long ranges require a skip entry as described above. Data was needed to relate these factors to TPS mass. For a given range, the initial flight path angle (FPA) also affects the lift profile and the nature of the skip entry, so a sensitivity of TPS mass to flight path angle was expected.

Entry azimuth (the vehicle heading at EI) affects the initial atmospheric relative velocity, thereby affecting heat load and TPS mass. This is especially true for Lunar return missions. Easterly headings at EI are in the same direction as Earth rotation, so initial atmosphere-relative velocities are smaller than for northerly headings. Azimuths between east and north are within the ranges required to achieve planned landing sites as discussed below in land and water landing considerations.

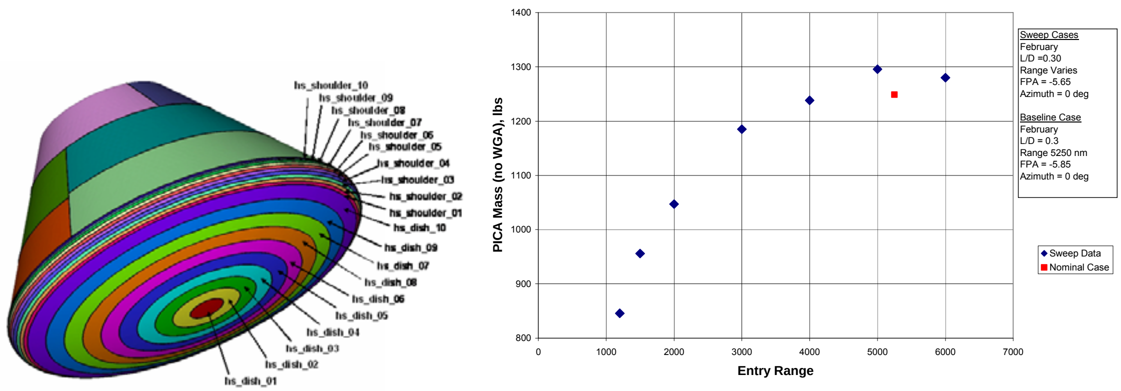
Vehicle mass is also affected by touchdown conditions since the structure must accommodate impact loads. The baseline design jettisons the heat shield and uses airbags for landing attenuation during land landings. During water landings, the heat shield is retained and airbags are not used. The airbag system weighs approximately XXX kg, so eliminating the land landing requirement represents a significant mass savings. Touchdown loads for both water and land are a function of terminal descent rate on the main chutes. The fault tolerance requirements dictate that the CM must survive landing loads with one of the three chutes failed. Increasing chute size decreases descent rate but increases chute mass, so data were needed to optimize the main chute system. Other sensitivities are also important in the landing phase, including drogue deploy altitude, which affects landing accuracy. Higher drogue deploy altitude usually reduces RCS propellant mass since final trajectory control ends at drogue deploy. Finally, the sea state at touchdown for water landings is an important structural loads driver. Wind velocity, wave height and wave slope all play a role.

The analyses conducted to assess the above sensitivities may be divided into pre-drogue deploy and post drogue deploy analyses. Pre deploy analysis was primarily aimed at understanding the impacts of spacecraft and trajectory parameters on TPS mass and RCS propellant. The post deploy analyses provided data to make decisions on chute sizing, structural mass and chute deployment strategies. The following sections are: a description of the TPS mass determination process used to assess TPS mass as a function of spacecraft and trajectory parameters, a description of the pre-deploy analyses and a description of the post deploy analyses

TPS mass determination process

Many of the sensitivities in Table X depend at least partly on required TPS thickness. Determination of TPS mass was accomplished in a multi-step process involving several NASA and contractor organizations [NAME THEM AND PUT IN ACKNOWLEDGEMENTS]. The process started with the GN&C team executing a set of Monte Carlo runs where a parameter of interest was varied systematically, and all other parameters were dispersed according to model doctrine. The resulting trajectories were then provided to [CAP] for high fidelity heating analysis. Once the heating dispersions were understood, a nominal trajectory and its associated heating profile were provided to the [TPS HEATSHEILD AND BACKSHELL PEOPLE] to determine required TPS thickness for the nominal case. A scale factor was then applied that was based upon the heating dispersions determined earlier. Figure 7 shows a visualization of one of the TPS modeling tools and some results for TPS mass as function of entry range.

Figure 7: TPS Heatshield and Backshell Bins and Example Results for Entry Range Variation



Pre Drogue Deploy Analysis Results

Table 9 is a portion of the Monte Carlo run matrix used to assess pre-drogue deploy sensitivities. The blue highlighted regions indicated the parameter of interest in the study. An initial set of runs were executed to determine the effect of time of year on heat loads. Heat load statistics from the worst month were then used to find the TPS factor of safety used in subsequent sensitivity studies.

Table 9: EDL Monte Carlo Sensitivity Run Matrix

	L/D	EI Range (nm)	EI Inertial FPA (deg)	Landing Site	EI Inertial Velocity (fps)	Entry Azimuth (deg)	Month	GPS Acquisition	Responsible Engineer	Quality Control Enginer	Monte Carlo Simulation	Nominal Trajectory Provided to TPS	MC Trajectories Provided to TPS
TPS Dispersion Factor	0.3	5250 (EI-1)	-5.65	UTTR	36046	0	Jan	Y, Y, Y	Putnam	Rea	Yes	Yes	Yes
	0.3	5250 (EI-1)	-5.65	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	Yes
	0.3	5250 (EI-1)	-5.65	UTTR	36046	0	Mar	Y, Y, Y	Putnam	Rea	Yes	Yes	Yes
	0.3	5250 (EI-1)	-5.65	UTTR	36046	0	April	Y, Y, Y	Putnam	Rea	Yes	Yes	Yes
	0.3	5250 (EI-1)	-5.65	UTTR	36046	0	May	Y, Y, Y	Putnam	Rea	Yes	Yes	Yes
	0.3	5250 (EI-1)	-5.65	UTTR	36046	0	June	Y, Y, Y	Putnam	Rea	Yes	Yes	Yes
	0.3	5250 (EI-1)	-5.65	UTTR	36046	0	July	Y, Y, Y	Putnam	Rea	Yes	Yes	Yes
	0.3	5250 (EI-1)	-5.65	UTTR	36046	0	August	Y, Y, Y	Putnam	Rea	Yes	Yes	Yes
	0.3	5250 (EI-1)	-5.65	UTTR	36046	0	September	Y, Y, Y	Putnam	Rea	Yes	Yes	Yes
	0.3	5250 (EI-1)	-5.65	UTTR	36046	0	October	Y, Y, Y	Putnam	Rea	Yes	Yes	Yes
	0.3	5250 (EI-1)	-5.65	UTTR	36046	0	November	Y, Y, Y	Putnam	Rea	Yes	Yes	Yes
	0.3	5250 (EI-1)	-5.65	UTTR	36046	0	December	Y, Y, Y	Putnam	Rea	Yes	Yes	Yes
	0.3	4010 (EI-2)	-5.65	UTTR	36046	0	September	Y, Y, Y	Putnam	Rea	Yes	Yes	Yes
	0.3	2160 (EI-5)	-5.65	UTTR	36046	47.8	September	Y, Y, Y	Putnam	Rea	Yes	Yes	Yes
L/D to Mass (TPS & Ballast)	0.33	5250 (EI-1a)	-5.65	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	Yes
	0.3	5250 (EI-1a)	-5.65	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	Yes
	0.27	5250 (EI-1a)	-5.65	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	Yes
	0.23	5250 (EI-1a)	-5.65	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	Yes
L/D to Mass (TPS & Ballast) w/ variable FPA	0.33	5250 (EI-1a)	-5.95	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	Yes
	0.3	5250 (EI-1a)	-5.85	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	Yes
	0.27	5250 (EI-1a)	-5.65	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	Yes
	0.23	5250 (EI-1a)	-5.55	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	Yes
Entry Range to Mass	0.3	6000	-5.65	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	No
	0.3	5000	-5.65	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	No
	0.3	4000	-5.65	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	No
	0.3	3000	-5.65	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	No
	0.3	2000	-5.65	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	No
	0.3	1500	-5.65	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	No
	0.3	1200	-6.00	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	No
Entry FPA to Mass	0.3	5250 (EI-1a)	-5.45	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	No
	0.3	5250 (EI-1a)	-5.55	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	No
	0.3	5250 (EI-1a)	-5.65	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	No
	0.3	5250 (EI-1a)	-5.75	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	No
	0.3	5250 (EI-1a)	-5.85	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	No
	0.3	5250 (EI-1a)	-5.95	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	No
	0.3	5250 (EI-1a)	-6.05	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	No
	0.3	5250 (EI-1a)	-6.15	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	No
	0.3	5250 (EI-1a)	-6.25	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	No
	0.3	5250 (EI-1a)	-6.35	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	No
Entry Azimuth/Velocity to Mass	0.3	5250 (EI-1a)	-5.65	UTTR	36046	0	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	No
	0.3	5250	-5.65	UTTR	36046	15	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	No
	0.3	5250	-5.65	UTTR	36046	30	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	No
	0.3	5250	-5.65	UTTR	36046	50	Feb	Y, Y, Y	Putnam	Rea	Yes	Yes	No

Figures X1 through X6 provide some of the consolidated results of the EDL sensitivities studies. Figure X1shows miss distance for four values of L/D for a skip entry trajectory. Note that when L/D is reduced to 0.23, the 6 Km (2.7 nm) landing accuracy requirement (solid line) is exceeded by the mean+3sigma results. For this and other reasons[], 0.27 was considered the minimum acceptable L/D for skip entries.

Figure X2 is an example of the total mass sensitivity (heat shield, backshell, and RCS propellant) to L/D for a vehicle of [18900 lbm in KG]. Each column represents the results of the TPS and RCS analysis from a 3000 case Monte Carlo set. These data quantify the mass sensitivity to L/D (but note that the ballast mass is book-kept separately). Note that mass is reduced with L/D. As will be discussed later, this is at the cost of footprint capability.

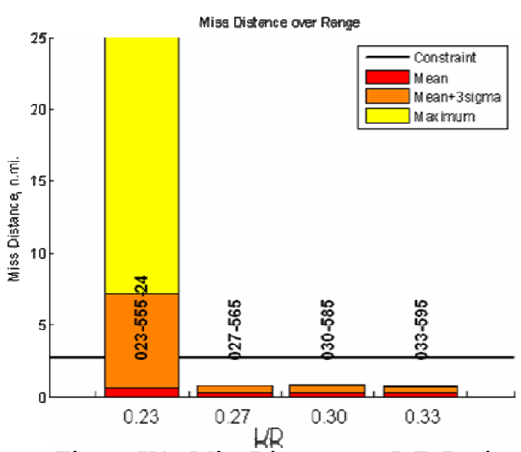


Figure X1. Miss Distance vs. L/D Ratio

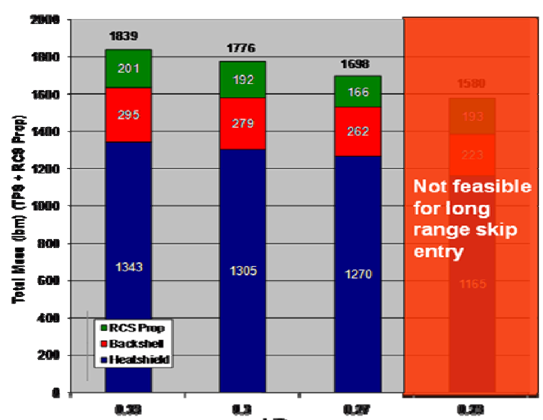


Figure X2. Combined TPS/RCS Mass vs. L/D

Figure X3 shows similar example results for sensitivity to initial flight path angle. These results provide information about how lunar return targeting may be planned to reduce heat loads and resulting TPS mass. Flight path angles in excess of about 6° resulted in [LOADS EXCEEDANCES?] so these angles were eliminated as candidates for mass reduction

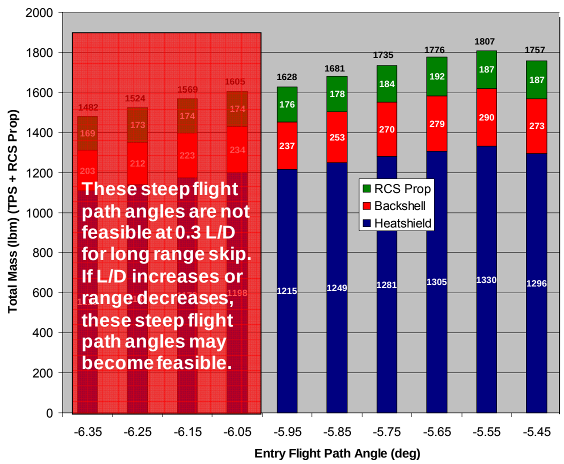


Figure X3. TPS and RCS Mass vs. FPA

Figure X4 shows the sensitivity of mass to entry range. As expected, the mass required for successful entry reduces with decreasing range. These data provided quantitative information to trade range capability and its associated mass, against the operational issues arising from a short range vehicle. These trades are discussed in more detail below.

Finally, Figure X5 depicts the landing footprints for [XXX] values of L/D. This helps to visualize the operational impact changing L/D.

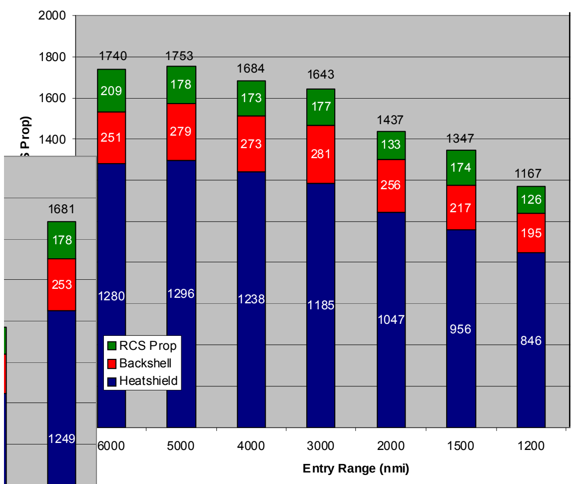


Figure X5. TPS and RCS Mass vs. Entry Range

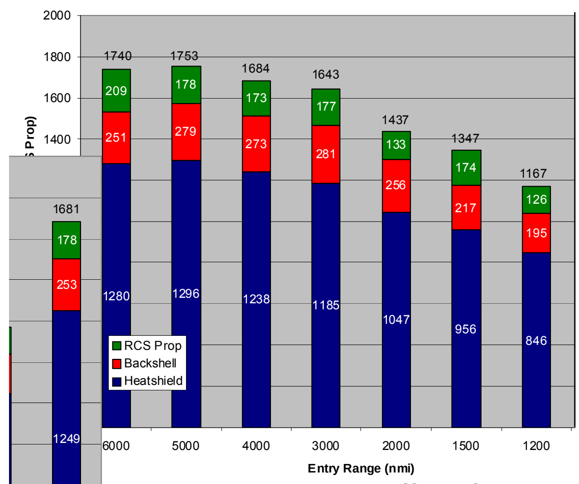


Figure X6. Footprint Effect of L/D

Post Drogue Deploy Analysis

In addition to determining the sensitivity of Mass to vehicle and trajectory parameters, data were generated to determine the effects of changing parameters on landing accuracy. Of particular interest were the effect on landing accuracy of changes in descent rate (related to main chute size and mass), presence or absence of GPS measurements, as well as drogue chute and main chute deployment altitude. Landing accuracy studies were conducted using statistical models of winds at the landing site. A method of biasing the guidance drogue deploy target point was developed which biased the drogue deploy point to cause the CM to hit the center of the landing area based on winds measured [4

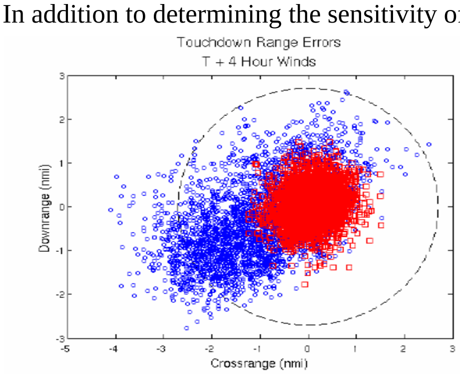


Figure X7. Effect of Drogue Deploy Bias

hours] prior to landing. Figure X7 shows the effect of biasing the drogue deploy point. The blue circles are touchdown points for a set of runs whose drogue target was base on nominal winds, while the red circles show the improvement achieved by biasing with measured winds. The red results still show distribution since the landing zone winds change during the four hours from measurement to touchdown. The next sections describe analyses conducted to evaluate the impact of changes in main chute sizing, drogue deploy altitude, and presence or absence of GPS for altitude determination. These results were chosen from a large group of studies, many of which are omitted for brevity.

Figure X8 shows results for the descent rate study. The figure shows the statistics for the range from the landing area center (range error) for four values of descent rate for both the biased, and non-biased case. The descent rate values were bounded by preliminary ranges of main chute sizes from the Landing and Recovery System (LRS) team. As expected, slower descent rates reduced landing accuracy for both the biased and unbiased tests. However, even the slowest descent rate (18 ft/s) met the 2.7 nm landing accuracy requirement when wind information was used to bias the drogue deploy point.

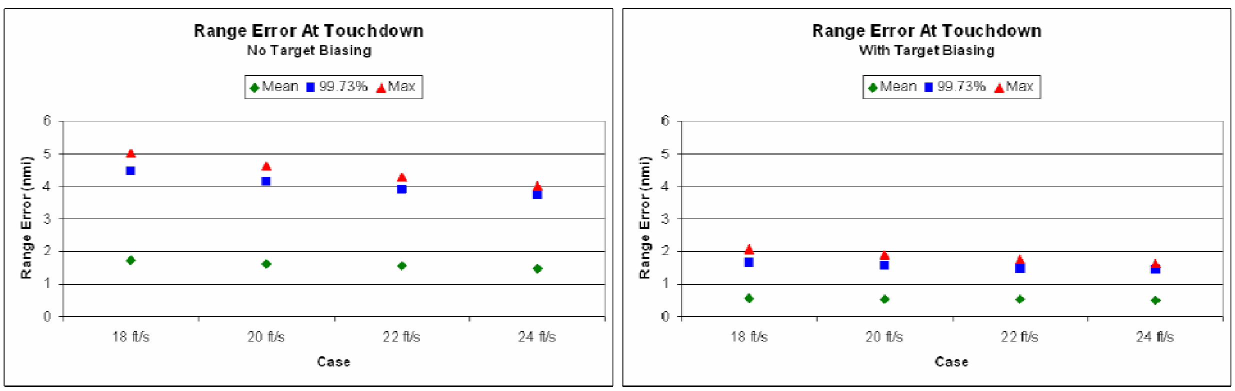


Figure X8. Results of Descent Rate Study

The baseline vehicle main chutes were sized to generate an XXX ft/s descent rate with a CM weighing [23000 lb] at landing. From these data, the decision was made to use the same main chutes with the new CM whose lower mass resulted in a descent rate of 18 ft/s.

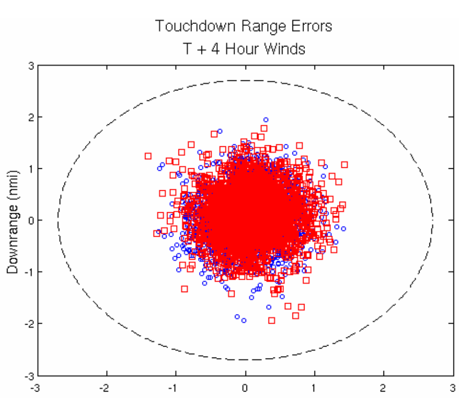


Figure X9. Landing Distributions for 40K ft (red) and 50K ft (blue) Drogue Deploy

Figure X9 overlays the touchdown points for drogue deploy altitudes of 40 and 50 thousand feet. These results show that touchdown accuracy is not very sensitive to drogue deploy altitude changes on the order investigated. Drogue deploy altitude is limited by mach at drogue deploy, so further increases were not considered. Table X shows the difference in propellant consumption for the higher drogue deploy altitude. Note that a small decrease in propellant may be achieved by increasing deploy altitude. [SAY SOMETHING ABOUT MAIN DEPLOY AND LEAD IN TO GPS ALTITUDE DETERMINATION]

The next study determined the effect of GPS availability on

Propellant Usage Prior to Drogue Deploy (lbm)			
	Mean	99.73 %	Max
40 kft	145.26	178.65	181.94
50 kft	136.50	171.67	195.90

drogue deploy altitude error. The baseline vehicle uses an integrated GPS/INS solution to determine the Earth-fixed state, including the altitude measurement used for drogue deployment. Lack of GPS, or loss of GPS input during entry, can result in large altitude errors at the drogue deploy time. The results of these analyses provide data to the vehicle design team to make decisions such as whether to add barometric altitude measurement and how many GPS instruments are required to robustly achieve landing accuracy.

Figure X10 shows the drogue deploy altitude error statistics for several GPS availability cases. Cases evaluated were:

- GPS always available (GPS 111)
- GPS available until 2nd entry blackout (GPS 110)
- GPS available until 1st entry blackout (GPS 100)
    - Represents case with a good initial state but no GPS support
- GPS not available (GPS 000)
    - Initial state from ground uplink prior to EI

Figure X11 shows the statistical altitude error and touchdown range error for these GPS availability cases. These results reflect the effect of IMU propagation accuracy from the point of GPS loss, as well as the assumed quality of the initial state, be it from the ground or GPS. The range error results were determined without drogue deploy biasing and assuming a -23.85 ft/s descent rate.

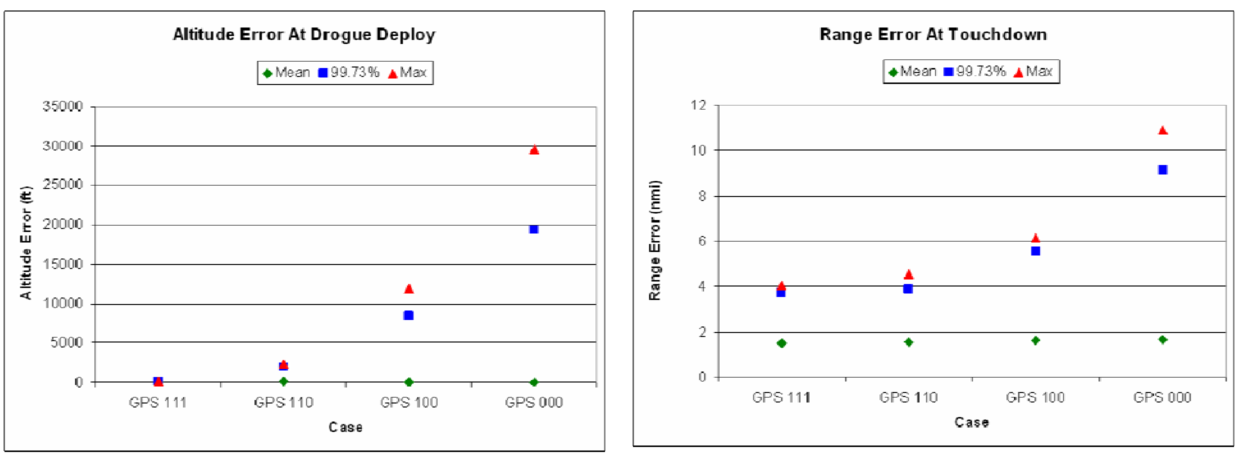


Figure X11. GPS Availability Results

The deploy altitude error results are of particular interest from the point of view of crew safety. In fact, for the 100 case, 56 runs resulted in failure to deploy the main chutes prior to impact, and case 000 (no GPS at all) resulted in over one hundred such failures.

Based on these data, and other on-orbit results discussed above, two GPS receivers were retained as part of the GNC sensor suite (a reduction from the original three), and a backup barometric altimeter (very low mass) was added to the POD vehicle

Operational Performance Trade Studies

Many of the changes discussed above had operational implications beyond the numerical capability impacts. In these cases, the GNC team conducted studies to assess operational impacts of candidate mass-saving changes. Some of the most important impacts had to do with landing and recovery operations. In this context, three general operational options were considered:

1. Retain land landing capability at southern California sites
    a. Requires guided skip entry
    b. Requires airbag system for landing
2. Support "shallow water" landing capability
    a. Land near the coast of southern California where sea states are often low
    b. Weather diverts require deep water recovery capability
    c. Requires guided skip entry (slightly shorter range than land landing)
3. Support only deep water landing capability
    a. Only direct entry required
    b. Deep water sites should be consolidated to simplify recovery operations

Figure X12 shows a comparison of landing footprints and Service Module (SM) disposal regions for the water landing options; 2 and 3 above. The red diamonds are the entry interface (EI) points, the blue ovals depict the SM debris field, and the pink boxes are the CM footprint from the EI states. The light blue regions represent SM disposal constraints, so the blue ovals cannot overlap these areas. Finally, the black dashed lines are bounds on the latitudes of the Lunar antipodes (the antipode is the intersection of the Moon-Earth vector with the Earth's surface opposite the Moon).

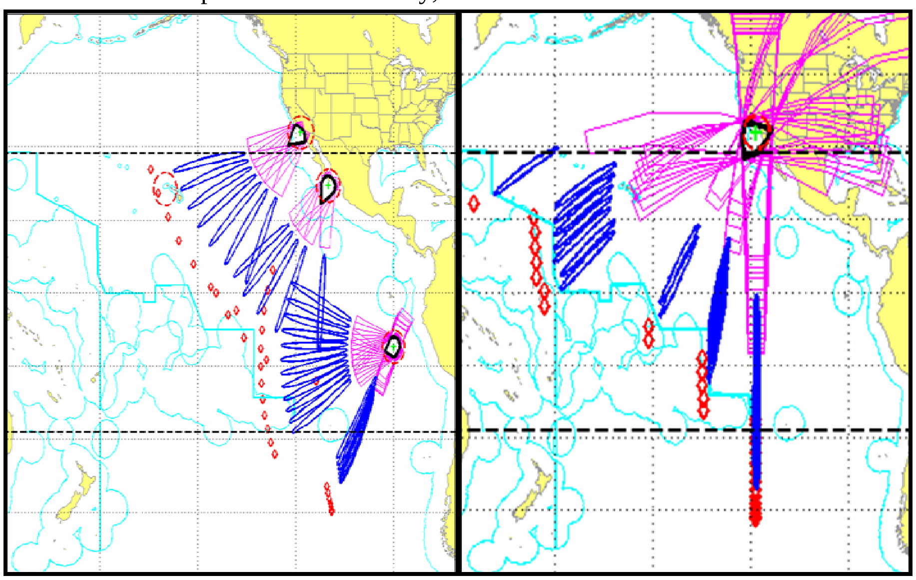


Figure X12. Recovery Site Visualization for Direct and Skip Entries

The SM disposal constraints together with the latitude of the Lunar antipode drive allowable return azimuths for entry. As the left figure shows, the California coastal landing sites cannot be reached with a direct entry from southerly antipode locations. For this reason, a strategy was created to consolidate landing areas to a minimum number of sites given the 2500 nm direct entry capability. This approach would require pre-positioned, recovery assets at one of these sites for nominal recoveries. Shallow water recovery would be available at 2 of the three sites, but deep water recovery would be required for returns during portions of each month in years with southerly antipode latitudes.

The right figure shows the entry strategies for a skip-capable vehicle to a California landing site. Availability analysis has shown that landing is available in this region [XXX] percent of [DAYS IN THE YEAR?]. However, deep water landing recovery would still be required for contingency.

The land landing entry strategy is similar to the coastal water landing strategy, so the plot is omitted. However, as noted above, land landing requires additional LRS and structural mass – on the order of 1500 lbs. Additionally, land landing may require higher L/D (and the associated ballast and TPS mass) to allow for landing site diverts after EI. Finally, land landing does not preclude certain deep water landing contingencies such as expedited lunar return for which azimuth control and time of flight are not controlled in favor of minimizing flight time or return propellant.

The above discussion only provides a general flavor of the considerations that went in to selecting the POD entry design. Further information …[INSERT REFERENCE] The next section summarizes the requirements changes and final POD design for the GN&C subsystem components as well as the subsystems effecting EDL performance.

CONCLUSIONS

The end result of the mass reduction effort of Orion was a PDR POD GN&C system capable of all of the functionality of the SDR 606A design with reduced fault tolerance for ascent and orbit operations. Navigation hardware was reduced by 33% in mass and 50% in peak power. Trajectory capability was deemed to provide the required transportation support for global lunar access via missionization or taking advantage of ARES I/V performance margin. Entry operations were curtailed with a change to targeted water landing with an option for land landing at a CONUS sight planned for via structural scarring. Multiple opportunities for design improvement were identified and will be pursued along with the overall GN&C system and trajectory design in the PDR design analysis cycle.

ACKNOWLEDGMENT

The technical content of this paper was based on the efforts of the NASA/LM Orion Flight Dynamics team and borrowed heavily from the technical reports of x, y, and z authored by …..

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[ii] McCants, E., Molina, L., Sims, C., and Sullivan, K., "*606 Service Module RCS Thruster Configuration Flight Control Assessment*," CEV-GNC-07-032, June 12, 2007.