Altair Descent and Ascent Reference Trajectory Design and Initial Dispersion Analyses

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The Altair Lunar Lander is the linchpin in the Constellation Program (CxP) for human return to the Moon. Altair is delivered to low Earth orbit (LEO) by the Ares V heavy lift launch vehicle, and after subsequent docking with Orion in LEO, the Altair/Orion stack is delivered through translunar injection (TLI). The Altair/Orion stack separating from the Earth departure stage (EDS) shortly after TLI and continues the flight to the Moon as a single stack.

Altair performs the lunar orbit insertion (LOI) maneuver, targeting a 100-km circular orbit. This orbit will be a polar orbit for missions landing near the lunar South Pole. After spending nearly 24 hours in low lunar orbit (LLO), the lander undocks from Orion and performs a series of small maneuvers to set up for descending to the lunar surface. This descent begins with a small deorbit insertion (DOI) maneuver, putting the lander on an orbit that has a perilune of 15.24 km (50,000 ft), the altitude where the actual powered descent initiation (PDI) commences.

At liftoff from Earth, Altair has a mass of 45 metric tons (mt). However after LOI (without Orion attached), the lander mass is slightly less than 33 mt at PDI. The lander currently has a single descent module main engine, with TBD lb$_f$ thrust (TBD N), providing a thrust-to-weight ratio of approximately TBD Earth g's at PDI.

LDAC-3 (Lander design and analysis cycle #3) is the most recently closed design sizing and mass properties iteration. Upgrades for loss of crew (LDAC-2) and loss of mission (LDAC-3) have been incorporated into the lander baseline design (and its Master Equipment List). Also, recently, Altair has been working requirements analyses (LRAC-1). All nominal data here are from the LDAC-3 analysis cycle. All dispersions results here are from LRAC-1 analyses.

Descent Phase

There are three subphases comprising the descent phase of the Altair mission: the braking burn (BB), the approach, and terminal (note a short pitch-up maneuver will be executed near the beginning of approach). The descent subphases are depicted shown in Figure 1.

Implicit guidance algorithms will be used to design the reference trajectories in all these descent subphases. In this approach, we can define, in advance of the mission, a reference trajectory as a vector polynomial function of time that evolves backward from the target state. But the reference trajectory cannot be expected to intersect the initial state of the vehicle due

to navigation and control dispersions. Implicit guidance will generate acceleration commands that consist of that computed using the reference trajectory plus two feedback terms. The first "feedback" acceleration correction is proportional to the difference between the actual and reference vehicle's positions. The second term is proportional to the difference between the actual and reference vehicle's velocities. Implicit guidance algorithm will "drive" the vehicle to achieve the target state in the presences of controller errors, navigation state estimation error and possible re-designation of the landing site.

The braking burn starts at the descent orbit perilune altitude of 15.24 km with the descent module (DM) main engine aligned with the lander's velocity vector. It is done at TBD% thrust to remove the orbital energy at the highest efficiency possible, but not at full thrust to allow for thrust margin between the BB set throttle and the maximum available engine power.

When the braking burn is completed, the lander will perform the pitch-up maneuver. The "nearly vertical" attitude of the lander will provide the crew with better visibility to detect terrain hazards surrounding the landing site. Re-designation of the landing target can then be performed after the pitch-up is completed.

In the approach subphase, the throttle will vary between 60% and 40% full engine power (i.e. thrust). The direction of thrust also varies to track the reference trajectory selected for the approach phase. There will be a hazard detection sensor carried onboard Altair. This critical function will be performed during the approach phase. Using data provided by the hazard detection sensor system, the crew will have to make a decision on the possible need to re-designate a "safer" landing site. The approach subphase ends at 30 meters vertically above the final (selected) touchdown site. A very important factor in this subphase is viewing (both crew look angles and terrain hazard detection sensor viewing).

The terminal subphase is intended to be a quiescent, controlled, vertical descent for 30 seconds at a constant 1 m/s velocity, until its time to shut down the DM engine. This shutdown is intended to occur at 1 meter above the surface. This means the touchdown velocity could be as high as 2.1 m/s.
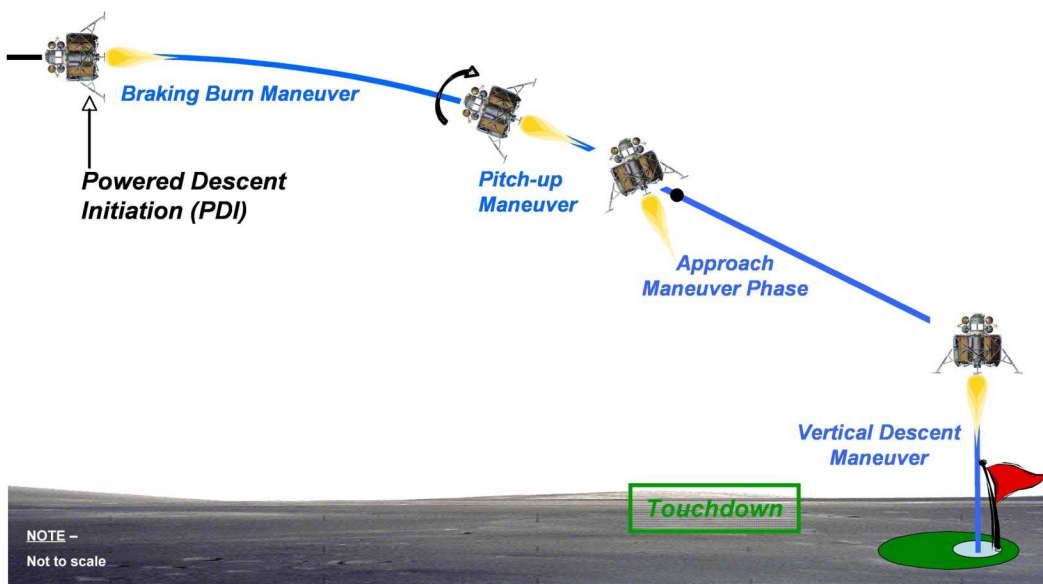

FIGURE 1 -- LUNAR DESCENT SUBPHASES

The mission delta velocities (ΔVs) are tabulated in Table 1. The total main propulsion system (MPS) descent ΔV for the piloted missions is TBD m/s (TBD m/s for cargo), while the current reaction control subsystem (RCS) budget ΔV is TBD m/s.

The main engine first lights up for the LOI burn (up to three LOI burns for non-polar sites). It will then perform the LLO plane change maneuver (when doing polar missions, if necessary). The final burn is the continuous descent and landing burn until shutdown at an altitude of 1 m. There are many small trajectory tuning maneuvers, as well as other RCS propellant usage for attitude control, all done with the RCS thrusters.

The LOI burn ΔV budget for polar missions is 891 m/s, with the global access LOI 3-burn total currently bought in at 950 m/s. The cargo LOI burn is 889 m/s due to smaller g-losses during LOI (better T/W ratio).

The descent ΔV budget currently also includes values for a small redesignation capability and a placeholder dispersions ΔV budget estimates using experience from Apollo 11.

TABLE 1 -- ALTAIR DESCENT MODULE ΔVS

LDAC-3 Descent & Ascent Module Reference Trajectory / ΔV Summaries

Descent ΔV Summary	Sortie, 950 m/s	Outpost	Cargo
Maneuver:	ΔV, m/s	ΔV, m/s	ΔV, m/s
TCMs (up to 4 of them/RCS)	TBD	TBD	TBD
LOI-1 (MPS)	326.0	891.0	889.0
LOI-2 (MPS)	59.0		
LOI-3 (MPS)	565.0		
LLO plane change (MPS)	0.0	28.5	0.0
DOI (RCS)	19.4	19.4	19.4
PDI (MPS)	2016	2016	2149
Redesignation (MPS)	TBD	TBD	TBD
Total Powered Descent	TBD	TBD	TBD
Dispersions (MPS)	53.0	53.0	53.0
Descent RCS	TBD	TBD	TBD
Total MPS Desc. ΔV+Dispersions:	TBD	TBD	TBD
Total MPS ΔV/Prop:	TBD	TBD	TBD
Total RCS ΔV/Prop+misc.:	TBD	TBD	TBD

The first iteration of the descent dispersions analysis was performed during LRAC-1 to begin affirming the ΔV value documented in the CxP descent ΔV requirements. The tool utilized in this work is the ANTARES simulation code (Advanced NASA Technology ARchitecture for Exploration Studies). The approach used was to first disperse one variable at a time. The variables that are currently captured in the "to disperse" list include DM main engine (ME) thrust, ME specific impulse (I_{sp}), propellant mass, c.m. locations (x, y, z) and inertia properties (I_{xx}, I_{yy}, I_{zz}, I_{xy}, I_{xz}, I_{yz}). Additional variables will be added to the dispersions list as the model is further developed.

The results from each individual contribution were root-sum-squared together to provide an interim estimate of total dispersion. This value came out to be TBD m/s. Note that with a subset of the variables that could possibly be dispersed, this value is less than the Apollo 11/Apollo 12 value in the ΔV table above. Future work includes dispersing all parameters together in one Monte Carlo exercise.

Ascent Phase

The ascent phase, also comprised of three subphases, is performed by only the Altair ascent module (AM). These three are the vertical rise, the single axis rotation (SAR), and the PEG (powered explicit guidance) subphases. Each of these has a particular purpose in flight and are depicted in Figure 2.

The vertical rise subphase is to immediately get the lander to a sufficient altitude to allow clearing of local terrain in the subsequent subphases. Although the AM is flying directly against (lunar) gravity to a height of approximately 100 m, which is quite an inefficient way to fly, it is only for a short time -- not more than 10 seconds.

The single axis rotation (SAR) maneuver subphase is to get the AM over to the desired flight path angle as quickly as possible. This is desired so that the subsequent ascent ΔV can be minimized. The earlier in the ascent the SAR is completed, the sooner the optimal flight path can be attained. This subphase is less than 15 seconds (currently TBD s), allowing PEG to be engaged at approximately TBD seconds after lunar liftoff.

The powered explicit guidance (PEG) subphase is then flown until the AM hits its main engine cutoff (MECO) targets (the AM state at MECO). The current AM flight times to MECO are TBD sec for both the Outpost and Sortie missions, so PEG is flying the last TBD seconds of the ascent trajectory. PEG is attempting to minimize the AM propellant required (e.g. the ΔV as well), so a significant majority of the ascent is optimally controlled.

PEG Phase

**Insertion:
R, V targets
$\gamma = 0$**
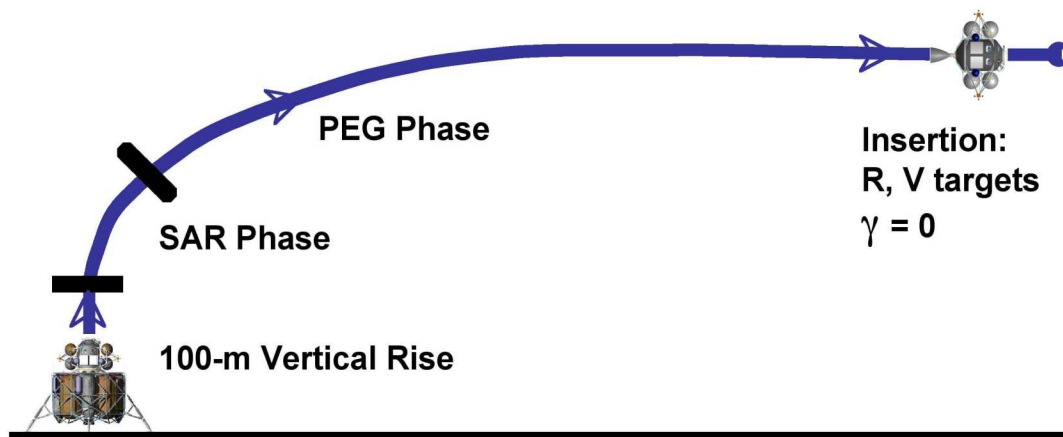
SAR Phase

100-m Vertical Rise

FIGURE 2 -- ASCENT TRAJECTORY SCHEMATIC

Ascent dispersions were also run during LRAC-1. Again these analyses were performed by dispersing one variable at a time. The variables studied in this analysis included AM main engine (ME) thrust, AM ME specific impulse, and AM initial (propellant) mass.

The results from each individual variable was root-sum-squared together to provide an interim total dispersion. This value came out to be TBD m/s. Note that this is only a subset of the variables that could possibly be dispersed. This effort was done using the Simulation and Optimization of Rocket Trajectories (SORT) trajectory tool. The next analysis cycle will have this analyses included in the ANTARES simulation development with results reported. As with the descent, future work will involve dispersing all parameters simultaneously.

Rendezvous, Proximity Operations and Docking Phase

As the name implies, the Rendezvous, Proximity Operations, and Docking (RPOD) phase is also comprised of three subphases. The rendezvous phase involves infrequent, discrete maneuvers with coasting phases. The proximity operations phase consists of smaller, more frequent maneuvers, and docking initiates at docking port contact. A simplified graphic of the reference trajectory is shown below in Figure 3.
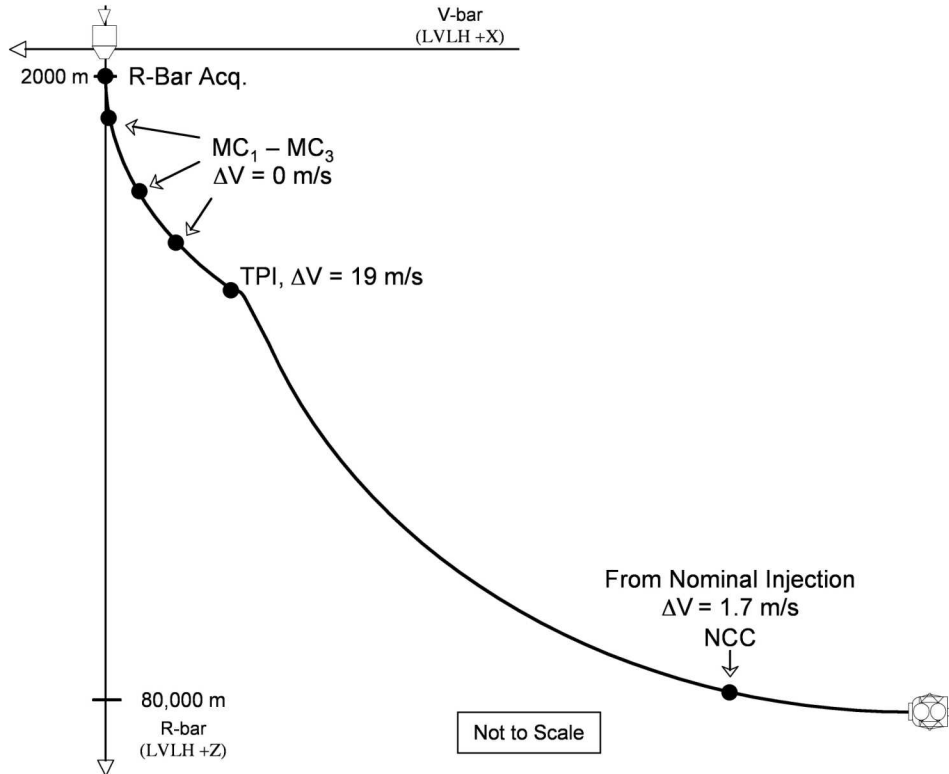

FIGURE 3 -- RENDEZVOUS TRAJECTORY SCHEMATIC

To deliver the Altair back to Orion, the RPOD trajectory utilizes an R-bar approach. The rendezvous phase begins at insertion and continues until Altair acquires the R-bar. During this time, there are five maneuvers. The Corrective Combination (NCC) burn occurs 10 minutes after insertion to clean up any ascent dispersions and target the proper Terminal Phase Initiation (TPI) point. The TPI burn places Altair on a natural coasting trajectory to acquire the R-bar at a point 2 km from Orion with a purely radial (Local Vertical, Local Horizontal (LVLH) -- Z) relative velocity. Between TPI and the R-bar, three Midcourse (MC) maneuvers shape the trajectory and correct any dispersions.

The 2 km acquisition range was chosen as part of a trade study performed during LDAC-2. It must be noted that since the trajectory from TPI to the R-bar is a "coasting" trajectory, the R-bar acquisition distance uniquely determines the location of the TPI point in space. This, in turn, defines the required apolune achieved during ascent.

The Proximity Operations phase begins at R-bar acquisition and concludes at docking port soft contact. To close the distance between the vehicles, Altair performs a series of "glide slope" maneuvers, along with two braking gates which serve to reduce the total profile time. At the conclusion of the glideslope maneuvers, Altair is delivered to a docking port-to-docking port

5

(DP-to-DP) range of 10 m with the proper range-rate for docking, thanks to natural braking from orbital mechanics. This velocity is held constant for the final 10 m until docking port contact.

A summary of the major RPOD events, the time frame, translational ΔVs, and the apolune and perilune are provided in Table 2.

TABLE 2 -- RENDEZVOUS SEQUENCE FOR ALTAIR TO ORION, 1-REV

Trajectory Event	Phase Elapsed Time, h:mm:ss	ΔV, Translation Maneuver	Resultant Apolune x Perilune
AM Lunar Liftoff			
Insertion / MECO			
Corrective Combination			
Terminal Phase Initiation			
Midcourse Corrections 1,2,3			
R-bar Acquisition			
1st & 2nd Braking Gates			
Glide slope maneuvers, total			
Docking			
Totals:	2:23:29	24.4 m/s	

An initial analysis of the RPOD maneuver dispersions was performed as a part of the RAC-1 analysis. The tool utilized was a Linear Covariance (LinCov) analysis code. Five different dispersion sources were considered one at a time. These error sources were environment process noise (from gravity perturbations or movement within Altair), the Orion initial navigation state knowledge, RCS performance (variations in thrust and I_{sp}), RCS misalignment, and an estimate of insertion errors from the Ascent phase.

The five individual error sources were then simultaneously run to estimate the total profile ΔV dispersion. The resultant value was TBD m/s. As part of the next analyses cycles, the dispersion sources will be updated with improved vehicle knowledge, and ANTARES simulation development will provide independent verification.

Summary/Conclusions/Future work

The current budget has captured all the major ΔV needs of the mission and is a good foundation from which to continue improving fidelity of the trajectory analyses and simulation development. The Altair lunar lander ΔV budget estimate and the dispersions for that estimate are continuing to progress over time as the Altair Project Office (APO) Integrated Performance Team works to mature the Altair vehicle models and trajectories.

Simulation work will also continue for quite some time. As flight hardware is selected, the components can be modeled in the simulation for eventual full system simulation, as well as possible hardware-in-the-loop simulation at some point in the future.

References:

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2. Jaggers, Roland F., "Shuttle Powered Explicit Guidance (PEG) Algorithm", Rockwell Space Operations Company, November 1992, JSC-26122.
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The Altair Lunar Lander is one element of NASA's Constellation Program for human return to the Moon. The Altair lander is responsible for several critical maneuvers on the way to and returning from the lunar surface. Since propellant to perform these maneuvers constitutes 50% (cargo) to greater than 60% (piloted) of the total lander mass, it is important to characterize the magnitude of these maneuvers as accurately as possible early in the design process. The Altair Lander descent module main engine performs the lunar orbit insertion burn(s), the low lunar orbit plane change burn when necessary, and the powered descent burn to lunar touchdown. The descent module also performs all trajectory correction maneuvers en route using a storable reaction control system as well as all attitude control functions. The Altair ascent module main engine performs the single, continuous ascent burn from the moon after the seven-day lunar surface mission to the low lunar orbit phasing ellipse. The ascent module then performs all rendezvous, proximity operations and docking maneuvers using the ascent module reaction control system. This paper describes the Altair performance characteristics and results determined thus far from the first four design and analysis cycles and presents the results of analysis and simulation work defining the Altair vehicle's required maneuvers as well as statistical analyses of anticipated dispersions in performance parameters.

Nomenclature

c_0, c_1, c_2	=	Unknown coefficients
g	=	Acceleration due to gravity, m/s^2
I_{sp}	=	Specific impulse, sec
K_P, K_D	=	Gain factors, scalar
r_0, v_0	=	Initial position and velocity
r_t, v_t, a_t	=	Target position, velocity and acceleration
t	=	Time, sec
tgo	=	Time to go, sec
$T/W, T/W_0$	=	Thrust-to-Weight ratio, Initial Thrust-to-Weight, non-dimensional
ΔV	=	Delta velocity, m/s

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δ_{cmd} =   Commanded pitch and yaw gimbal angles, rad

Θ_{err} =   Pitch and roll attitude errors, rad

ω_{err} =   Pitch and roll angular rate errors, rad/sec

I. Introduction

THE Altair Lunar Lander is the linchpin in the Constellation Program (CxP) for human return to the Moon. Altair is delivered to low Earth orbit (LEO) by the Ares V heavy lift launch vehicle, and after subsequent docking with Orion in LEO, the Altair/Orion stack is delivered through trans-lunar injection (TLI). The Altair/Orion stack separates from the Ares V Earth departure stage (EDS) shortly after TLI and continues the flight to the Moon as a single stack.
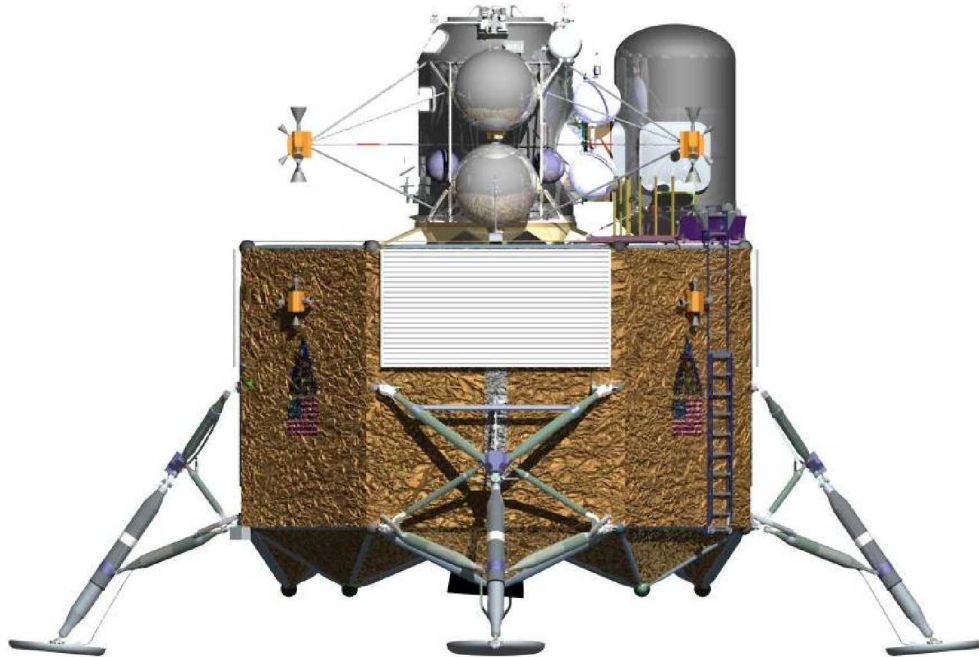

Figure 1. Altair Lunar Lander, Side View. *The Altair Lunar Lander is comprised of an Ascent Module (top, center), an Airlock (top, right), and a Descent Module (bottom, gold Mylar).*

Altair performs the lunar orbit insertion (LOI) maneuver(s), targeting a 100-kilometer (km) circular orbit. This orbit will be a polar orbit for Outpost missions landing near the lunar South Pole and other inclinations for Global Access missions to other points on the lunar surface. After spending approximately 20 to 116 hours in low lunar orbit (LLO), the lander undocks from Orion and performs a series of small maneuvers to set up for descending to the lunar surface. This descent begins with a small descent orbit insertion (DOI) maneuver, putting the lander on an orbit that has a perilune of 15.24 km (50,000 ft), the altitude where the actual powered descent initiation (PDI) commences.

The Altair design team began work in 2007 and has conducted a series of analysis cycles focusing on design and requirements maturation. With each cycle the design matures and evolves to improve reliability and compliance to program requirements. Nominal data presented here is from design and analysis cycle 4 (LDAC-4), completed June 2010. Trajectory dispersions data is from work conducted during the requirements analysis cycles 1 and 2 (LRAC-1 and LRAC-2), completed in June 2009 and December 2009 respectively.

Upgrades for loss of crew (LDAC-2) and loss of mission (LDAC-3) have previously been incorporated into the lander baseline design (and its Master Equipment List). All nominal data here are from the most recent LDAC-4 analysis cycle.

II. Trans-lunar Injection through Descent Orbit Insertion

While active in LEO, the Altair lander is not responsible for any maneuvers until after the TLI burn is completed by the Ares V EDS. The trans-lunar cruise (TLC) phase of the mission spans the 4 days between TLI and initiation

of LOI. During this time Altair is responsible for attitude control of the Orion-Altair stack, and any trajectory correction maneuvers that may be required during transit. These maneuvers are performed by Altair's Descent Module reaction control system (RCS). The first firing of Altair's main engine occurs at LOI. The magnitude and quantity of LOI maneuvers required depends on the mission and landing site selection. Following LOI, the Altair-Orion stack loiters in LLO for a minimum of 24 hours. This time is allocated to allow for sufficient state vector updates to enable a precision landing, and to allow for crew operations which include preparing the Orion vehicle for untended operations, preparing the Altair vehicle for descent, and a crew sleep period. During LLO loiter Altair is responsible for attitude control as well as altitude maintenance as needed. At the conclusion of the LLO loiter period, the uncrewed Orion vehicle separates and backs away from the now crewed Altair. Altair then executes a plane change maneuver if required, and the DOI maneuver.

A. Trans-lunar Cruise (TLC)

During the TLC the Altair will maintain a stack attitude where the Altair DM main engine and landing gear are pointed away from the Sun, and Orion's high gain antenna is pointed toward the Earth with deadbands as high as 20 degrees. Preliminary estimates indicate that this attitude can be maintained with a small amount of propellant, possibly on the order of 6 kg/day.[1]

There are four trajectory correction maneuvers (TCM) planned in the Altair nominal TLC, however it is anticipated that all four will not be executed every mission. These maneuvers are intended to correct for TLI inaccuracies, both natural and induced perturbations to the vehicle, and any navigation errors that accumulate during the transit to the moon. More information about navigation during TLC can be found in Ref. 2. The allocation of delta velocity (ΔV) for each TCM and an anticipated schedule are given in Table 1. Actual TCMs, when required, will be performed at a magnitude and schedule that is optimized during each mission. The root-sum-square (RSS) total budget for the four TCM ΔVs is 25.4 m/s.

Table 1.   Altair TLC Trajectory Correction Maneuver ΔV Summary.

TCM #	3σ ΔV, m/s	Time from TLI, hours	Time from LOI, hours
TCM-1	21.5	2 to 3	93 to 94
TCM-2	2.0	~24	~72
TCM-3	3.7	~62	~34
TCM-4	2.6	~88	~8
RSS Total:	25.4 m/s		

B. Lunar Orbit Insertion (LOI)

The LOI burn was determined by CxP Level 2 analyses.[3,4] The LOI ΔVs are 894 m/s for the piloted Outpost mission, 953 m/s for the Global Access Sortie mission, and 892 m/s for the Cargo mission.[3,4] LOI dispersions were determined by the same analyses which generated the TCM ΔVs in the previous section resulting in an additional 3 m/s to cover for various error sources.[2]

At the Lunar Capability Concept Review (LCCR) in June 2008, CxP decided to allocate 950 m/s for the Global Access mission for LOI.[4] This value is a compromise between percentage of the lunar surface that is accessible, the temporal availability of the more difficult sites, and the desire to minimize the LLO loiter duration and the LOI ΔV requirement. The Outpost and Cargo ΔVs are just a resultant of the same analyses, but sized for just the South Pole location. All of these LOI ΔVs are intended to accommodate as many departure (TLI) and arrival (LOI) epochs as possible over the course of the 18.6-year cycle of the Moon's inclination variation.

C. Low Lunar Orbit (LLO)

During the LLO loiter period, Altair is responsible for both the attitude and altitude maintenance for the Altair/Orion stack. It is important to keep the stack oriented in a solar inertial attitude so that Orion's solar arrays are pointed toward the Sun for power generation. For Orion thermal constraints, it is desired to keep the lunar orbit altitude within a 10-km deadband about the nominal 100-km altitude. For polar missions, the loiter time will be approximately 24 hours. For global access missions, the low lunar loiter time could be up to four additional days beyond the normal 1-day LLO loiter.

To determine an ACS budget, a scan of different solar beta angles, lunar orbit inclinations and ascending nodes was completed. From these studies, an average propellant usage for attitude and altitude maintenance was determined. For the attitude maintenance budget, 30 kg/day was budgeted. For the altitude maintenance budget, although none is needed for a 1-day loiter, 35 kg/day was budgeted for the 5-day loiter.

For the Outpost mission, it will likely be necessary to perform a LLO plane change maneuver to place the orbit plane over the landing site if the Outpost site is not located very close to the pole. The Altair lander can perform up to a 1 degree maneuver and the corresponding ΔV budget for this maneuver is 28.5 m/s. This is done with the DM MPS system to minimize the propellant needed for the maneuver.

D. Descent Orbit Insertion

The descent orbit insertion burn is done using the DM RCS. The DOI maneuver drops the lander from the LLO circular orbit to an elliptical orbit with perilune much nearer the surface. This maneuver is done approximately half of an orbit from the landing point. This is done with the DM RCS to preserve the accuracy of the knowledge of the lander state to facilitate landing within the constraint of achieving 1 km accuracy at touchdown for Sortie missions (and prior to Outpost establishment) and 100-meter accuracy for Outpost missions after it has been established.

III. Descent Phase

The descent phase is one continuous burn, beginning at Powered Descent Initiation (PDI). There are three subphases comprising the descent phase of the Altair mission: the braking burn (BB), the approach, and terminal (note a short pitch-up maneuver will be executed near the beginning of approach), all shown in Figure 1.

At liftoff from Earth, for the Sortie and Outpost mission, the Altair lander has a mass of 45 metric tons (mt). In LLO after separation from Orion, the lander masses are 31 mt and 32 mt at PDI, respectively. The DM main propulsion system (MPS) currently has a single descent module main engine, a nominal specific impulse (I_{sp}) of 450.1 sec (with a minimum guaranteed I_{sp} of 448.6 sec), with 18,627 lb_f thrust (82,900 N), providing a thrust-to-weight (T/W) ratio of 0.24 Earth g's at PDI. The cargo mission masses are 54.5 mt at liftoff and 43.5 mt at PDI (Cargo T/W = 0.18 Earth g's).
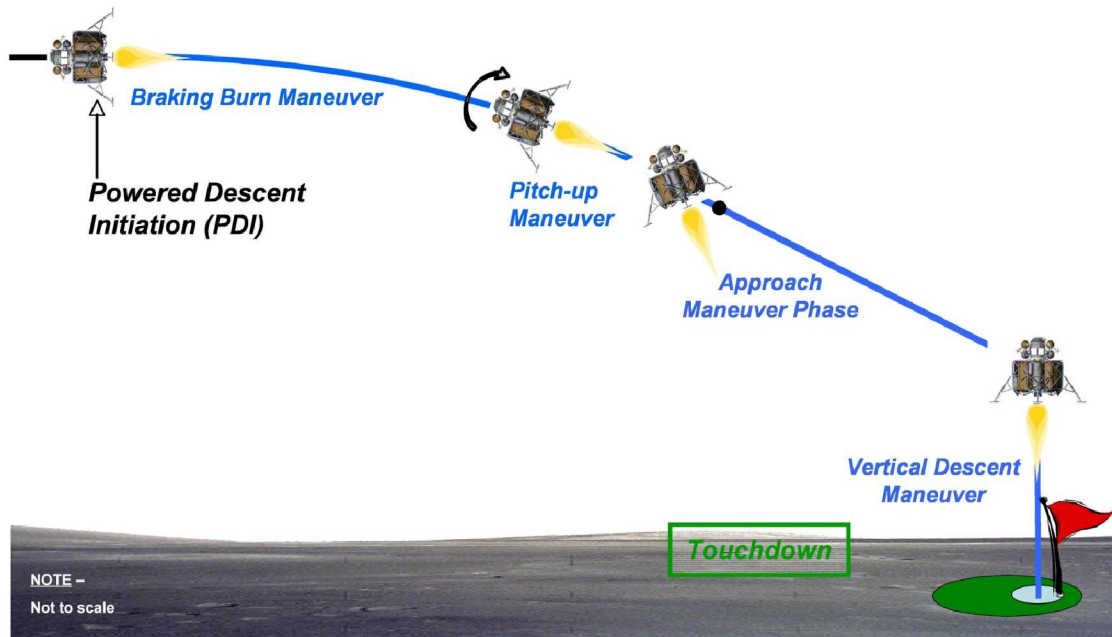

Figure 2. Lunar Descent Subphases. *The descent phase is comprised of a braking burn subphase, a pitch-up maneuver, an approach subphase and a terminal descent subphase.*

A. Braking Burn

The braking burn starts at the descent orbit perilune altitude of 15.24 km with the descent module (DM) main engine aligned with the lander's velocity vector. It is done at 92% thrust to remove the orbital energy at the highest efficiency possible, but not at full thrust to allow for thrust margin between the BB set throttle and the maximum available engine power. Thrust margin is necessary in order be able to remove dispersions during the braking burn.

B. Pitch-up Maneuver

When the braking burn is completed, the lander will perform the pitch-up maneuver. The "nearly vertical" attitude of the lander will provide the crew with better visibility to detect terrain hazards surrounding the landing site. Re-designation of the landing target can then be performed after the pitch-up is completed.

C. Approach

During the approach phase, the vehicle descends at a lower throttle (roughly between 60% and 40% full engine thrust) while the landing area is examined for hazards. The magnitude and direction of thrust varies to track the reference trajectory selected for the approach phase. A hazard detection sensor carried onboard is used to assist in identification of the best location in which to land. For the piloted mission, the crew will have to make a decision on the possible need to re-designate to a "safer" landing site. This function can also be carried out autonomously for a cargo mission or in case of an incapacitated crewmember. The trajectory design must account for providing a landing approach that enables adequate viewing of the landing area by the both the crew and the hazard detection system.[5-10]

If a new landing location is selected, the target is updated in the GN&C software and the guidance calculates an updated trajectory that delivers the vehicle to the new location. This changes the throttle and attitude profile for the remaining portion of the phase. An additional amount of propellant is expected to be used and this has been allocated in the ΔV budget. Redesignation can occur multiple times during the Approach Phase.

Whether redesignation occurs or not, the approach subphase ends at 30 meters vertically above the final (selected) touchdown site.

D. Terminal

The terminal subphase is intended to be a quiescent, controlled, vertical descent for 30 seconds at a constant 1 m/s velocity over the last 30 meters of altitude. The DM engine shutdown occurs just prior to touchdown and the shutdown sequence is initiated at 1 meter above the surface. Assuming a free fall from 1-m height at -1 m/s, the expected worst case touchdown velocity is 2.1 m/s.

E. Descent Dispersions

The first iteration of the descent dispersions analysis was performed during LRAC-1 to begin affirming the ΔV dispersions budget documented in the CxP descent ΔV requirements. The tool utilized in this work is the ANTARES simulation code (Advanced NASA Technology ARchitecture for Exploration Studies). This 6-DOF simulation was developed under Altair for use in both GN&C development and performance assessment. Guidance and control algorithms, along with a perfect navigation assumption, were used. The study approach used was to first disperse one variable at a time. The variables that are currently captured in the "to disperse" list include DM main engine (DME) thrust, DME specific impulse (I_{sp}), propellant mass, c.m. locations (x, y, z) and inertia properties (I_{xx}, I_{yy}, I_{zz}, I_{xy}, I_{xz}, I_{yz}). Additional variables will be added to the dispersions list as the model is further developed.

The results from each individual contribution were RSS'd together to provide an interim estimate of total dispersion. This value came out to be 28.2 m/s. Note that with a subset of the variables that could possibly be dispersed, this value is less than the Apollo 11/Apollo 12 ΔV value in Table 2 below.[11] Future work includes dispersing all parameters together in one Monte Carlo exercise.[12]

F. Guidance Description

The guidance algorithm is based on the Apollo Lunar Module Guidance.[5] During the braking and approach phases, the acceleration profile is restricted to be a quadratic equation of time.

$$a = c_0 + c_1 t + c_2 t^2 \tag{1}$$

Analytic equations for velocity and position can be integrated from this equation and the coefficients c_1 and c_2 can solved using the initial position and velocity, v_0 and r_0.

$$c_0 = a_t - 6\frac{(v_t + v_0)}{tgo} + 12\frac{(r_t - r_0)}{tgo^2} \tag{2}$$

$$c_1 = -6\frac{a_t}{tgo} + 6\frac{(5v_t + 3v_0)}{tgo^2} - 48\frac{(r_t - r_0)}{tgo^3}$$

(3)

$$c_2 = 6\frac{a_t}{tgo^2} - 12\frac{(2v_t + v_0)}{tgo^3} + 36\frac{(r_t - r_0)}{tgo^4}$$

(4)

The length of the burn, tgo, and the position, velocity, and acceleration targets (r_t, v_t, a_t) must be provided. These equations are solved in each axis. A new set of targets is used for each phase.

G. Controller Description

The control algorithm uses thrust vector control to steer the vehicle and was developed by the Autonomous Landing and Hazard Avoidance Technology (ALHAT).[6] The thrust vector control (TVC) is used to control the vehicle's pitch and yaw. The TVC has no control authority in the roll axis. RCS jets are used to control roll. A simple proportional-derivative (PD) controller is driven by pitch and yaw attitude and angular velocity errors.

$$\delta_{cmd} = K_P\Theta_{err} + K_D\omega_{err}$$

(5)

Where the commanded pitch and yaw gimbal angles, δ_{cmd}, is determined by the pitch and roll attitude errors, Θ_{err}, and pitch and yaw angular rate errors, ω_{err}. The two gains, K_P and K_D are scalars.

H. Descent Module Delta Velocities (ΔVs)

The mission ΔVs for the descent module are tabulated in Table 2. The total MPS descent ΔV for the piloted missions requires 2074 m/s (2202 m/s for cargo), while the current RCS budget ΔV requires up to 76 m/s.

The main engine first lights up for the LOI burn (up to three LOI burns for non-polar sites). It will then perform the LLO plane change maneuver (when doing polar missions, if necessary). The final burn is the continuous descent

Table 2.   Altair Descent Module MPS and RCS ΔV Summary.

	LDAC-4 Sortie	LDAC-4 Outpost	
	25.4	25.4	22.9
	6.4 kg/day	6.4 kg/day	~4 kg/day
	0.6	0.6	0.7
	326.0	891.0	889.0
	59.0		
	565.0		
	+2.6	+2.6	+2.6
LOI Subtotal (MPS)	952.6	893.6	891.6
	7.0	7.0	7.0
	10.1		
	30 kg/day	30 kg/day	30 kg/day
LLO plane change (MPS)		28.5	
	19.4	19.4	19.4
	2.2	2.2	1.7
	1798.9	1798.9	1937.1
	53.0	53.0	53.0
	+5.0	+5.0	+0.0
Powered Descent Subtotal (MPS)	2073.8	2073.8	2202.3
	+0.0	+0.0	
Total MPS Descent ΔV:	2073.8	2073.8	2202.3
Total MPS ΔV:	3026.4	2995.9	3093.8
	+11.0	+11.0	+11.0
Total RCS ΔV:	75.6	65.5	62.6

and landing burn until shutdown at an altitude of 1 m. The descent ΔV budget currently also includes values for a small redesignation capability and a placeholder dispersions ΔV budget estimates using experience from Apollo 11.[11] The DME is lit as many as four times and as few as two for this stable of missions.

There are many small trajectory tuning maneuvers shown in the table, as well as other RCS propellant usage for attitude control, all done with the RCS thrusters.

IV. Ascent Phase

The ascent phase, also comprised of three subphases, is performed by only the Altair ascent module (AM). These three subphases are the vertical rise, the single axis rotation (SAR), and the powered explicit guidance (PEG) subphases. Each of these has a particular purpose in flight and are depicted in Figure 3.

A. Vertical Rise

The vertical rise subphase is to immediately get the lander to a sufficient altitude to allow clearing of local terrain in the subsequent subphases. Although the AM is flying directly against (lunar) gravity to a height of approximately 100 meters, which is quite an inefficient way to fly, it is only for a short time -- not more than 10 seconds.
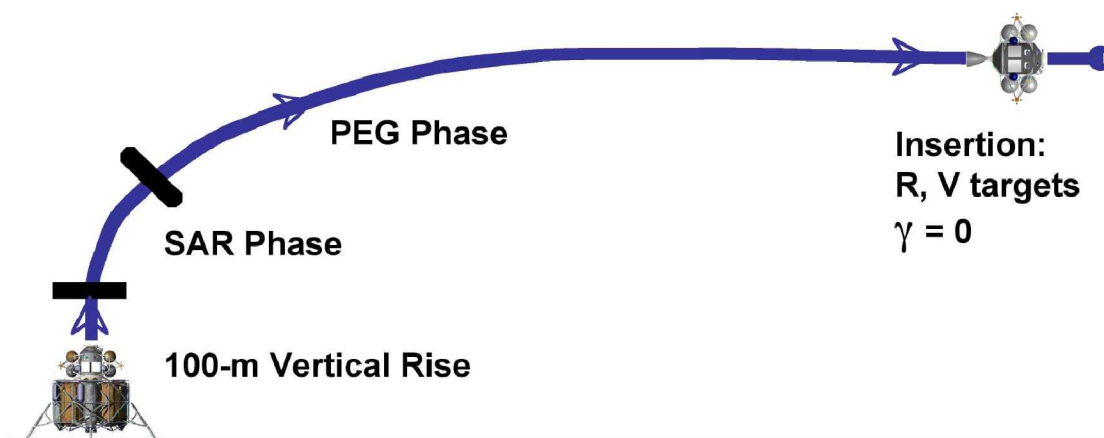

Figure 3. Lunar Ascent Trajectory Schematic. *The ascent phase is comprised of a vertical rise subphase, a single-axis rotation maneuver, and a powered explicit guidance (PEG) subphase.*

B. Single Axis Rotation

The SAR maneuver is an attitude-rate and attitude-acceleration limited maneuver designed to re-orient the AM to the desired attitude for the start of the PEG phase as quickly as possible. The earlier in the ascent the SAR is completed, the sooner the optimal flight path can be attained. This subphase is less than 15 seconds (currently 11 seconds), allowing PEG to be engaged at approximately 20 seconds after lunar liftoff. The maximum and desired pitch rate during SAR is 5 degrees per second.

C. Powered Explicit Guidance

The PEG subphase is then flown until the AM hits its main engine cutoff (MECO) targets (the AM state at MECO). The current AM flight times to MECO are 401 sec for the Sortie mission, so PEG is flying approximately the last 380-385 seconds of the ascent trajectory. The Outpost mission has a burn time of 430 sec (~410-415 sec for PEG). PEG is attempting to minimize the AM propellant required (e.g. the ΔV as well), so about 95% of the ascent is optimally controlled.[13,14,15]

D. Ascent Module Delta Velocities (ΔVs)

The mission ΔVs for the AM are tabulated in Table 3 for both the Sortie and Outpost piloted cases. The lunar ascent is a single continuous burn for the AM MPS. The AM MPS uses a 5000 lb$_f$ (24,500 N) storable propellant engine with an Isp of 320 sec, the major difference between the two cases is the different initial mass at lunar liftoff (or abort during descent) for each variation of the vehicle, thus requiring a slightly higher ΔV done by the MPS for

the Outpost mission (Sortie $T/W_0 = 0.35$; Outpost $T/W_0 = 0.33$). The subsequent portions of the missions are nearly identical for the two cases.

The ΔV budgets for both the MPS dispersions and a lunar launch window are added to that performed by the MPS. The launch window analyses was performed for a range of launch site latitudes and target orbit inclinations. To envelope three launch opportunities (one attempt two hours prior to nominal liftoff and one attempt two hours after nominal liftoff), 8 m/s was determined to be necessary for the worst-case lunar surface location.

Table 3. Altair Ascent Module MPS and RCS ΔV Summary.

	LDAC-4 Sortie	LDAC-4 Outpost
	1829.5	1840.6
	8.0	8.0
	+5.1	+5.1
	10.0	10.0
	+74.0	+74.0
	1.8	1.8
	19.0	19.0
	3.7	3.7
	+22.4	+22.4
Total Powered Ascent ΔV:	**1973.4**	**1984.5**

V. Rendezvous, Proximity Operations and Docking Phase

As the name implies, the Rendezvous, Proximity Operations, and Docking (RPOD) phase is also comprised of three subphases. The rendezvous phase involves infrequent, discrete maneuvers with coasting phases. The proximity operations phase consists of smaller, more frequent maneuvers, and docking initiates at docking port contact. A simplified graphic of the reference trajectory is shown below in Figure 4.

A. Rendezvous

To deliver the Altair back to Orion, the RPOD trajectory utilizes an R-bar (i.e. the orbit radius vector) approach. The rendezvous phase begins at insertion and continues until Altair acquires the R-bar. During this time, there are five maneuvers. The Corrective Combination (NCC) burn occurs 10 minutes after insertion to clean up any ascent dispersions and target the proper Terminal Phase Initiation (TPI) point. This time delay should allow Altair to perform relative navigation using its star tracker and S-band communication system for post-insertion estimates of bearing and range/range-rate, respectively.

The TPI burn places Altair on a natural coasting trajectory to acquire the R-bar at a point 2 km below Orion with a purely radial (Local Vertical, Local Horizontal (LVLH)-Z) relative velocity. Between TPI and the R-bar, three Midcourse (MC) maneuvers shape the trajectory and correct any dispersions.

The MC burns were placed to allow the Altair Laser Imaging, Detection, and Ranging (LIDAR) system to acquire Orion between MC_2 and MC_3. There is roughly 330 seconds between the outer edge of the LIDAR range and the MC_3 burn, which allows plenty of time for the LIDAR to acquire the target and process navigation knowledge prior to performing MC_3. Using the improved state knowledge from the LIDAR, Altair is able to significantly reduce its dispersions (by a factor of 4 for downrange and a factor of 22 for altitude) before R-bar acquisition.

The 2 km acquisition range was selected as part of a trade study performed during LDAC-2. It must be noted that since the trajectory from TPI to the R-bar is a "coasting" trajectory, the R-bar acquisition distance uniquely determines the location of the TPI point in space. This, in turn, defines the required apolune achieved during ascent.
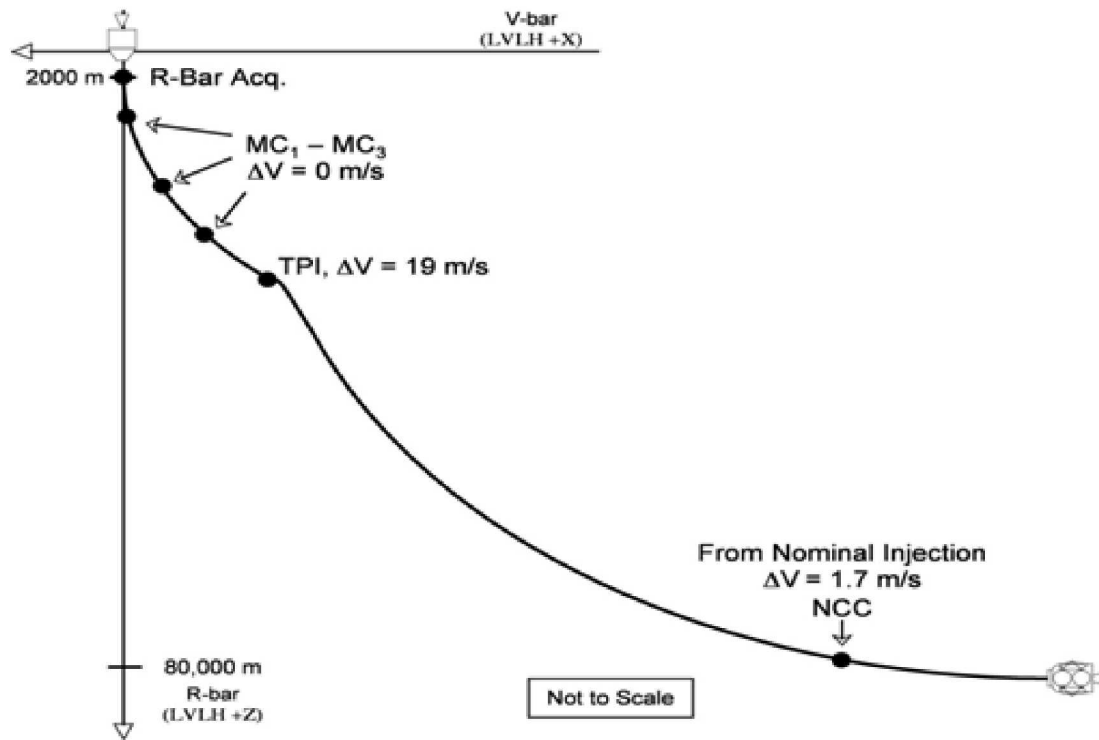

Figure 4. Rendezvous Trajectory Schematic. *The RPOD phase is comprised of a rendezvous subphase, a proximity operations maneuvers, and a docking subphase.*

B. Proximity Operations

The Proximity Operations phase begins at R-bar acquisition and concludes at docking port soft contact. To close the distance between the vehicles, Altair performs a series of "glide slope" maneuvers, along with two braking gates, the first a 0.2 m/s burn at 100 m, and the second a 0.1 m/s burn at 50 m, which serve to reduce the total profile time by nearly 15 minutes. The distance of the vehicles at the times of the braking gates, along with the time between the second braking gate and docking, should provide enough time for any perturbations on Orion from plume impingement to dampen out. However, this assumption should be validated in the future through simulation. At the conclusion of the glideslope maneuvers, Altair is delivered to a docking port-to-docking port (DP-to-DP) range of 10 m with the proper range-rate for docking, thanks to natural braking from orbital mechanics. This velocity is held constant at -0.03 m/s for the final 10 m until docking port contact.

C. Docking

Altair and Orion will utilize the Low-Impact Docking System (LIDS) for docking. LIDS is a fully androgynous docking system, and is the standard for all docking in the Constellation Program. Any docking system places constraints on the contact conditions at time of docking, and LIDS is no exception. As a 3-DOF tool, Linear Covariance (LinCov) analysis is inappropriate to perform docking analysis, and the simulation was terminated at the beginning of the constant approach. A detailed analysis of Altair/Orion docking in LLO has been performed, however, and the results indicate that the current Altair configuration is able to successfully complete the docking.[16]

A summary of the major RPOD events, the time frame, translational ΔVs, and the apolune and perilune are provided in Table 4. The total deterministic ΔV for RPOD is 24.4 m/s.

Table 4. Altair Rendezvous Sequence for Altair to Orion (1-rev).

Trajectory Event	Phase Elapsed Time, h:mm:ss	ΔV, Translation Maneuver	Resultant Apolune x Perilune
AM Lunar Liftoff	0:00:00	----	----
Insertion / MECO	0:06:46	----	79 x 15 km
Corrective Combination	0:16:46	1.7 m/s	86 x 14 km
Terminal Phase Initiation	1:02:56	19 m/s	99 x 85 km
Midcourse Corrections 1, 2, 3	1:14:19 - 1:38:04	0 m/s	99 x 85 km
R-bar Acquisition	1:42:04	0.4 m/s	99 x 85 km
1st & 2nd Braking Gates	2:07:44, 2:11:16	0.3, 0.1 m/s	100x98, 100x99 km
Glide slope maneuvers, total	----	2.9 m/s	----
Docking	2:23:29	----	100 x 100 km
Totals:	**2:23:29**	**24.4 m/s**	

D. RPOD Dispersions

An initial analysis of the RPOD maneuver dispersions was performed as a part of the RAC-1 analysis. The tool utilized was a LinCov analysis code. One of the outputs of the LinCov analysis is the statistical mean and standard deviation for each maneuver. To determine the sensitivity of the RPOD profile to various dispersion sources, five different dispersion sources were considered one at a time. These error sources were environment process noise (from gravity perturbations or movement within Altair), the Orion initial navigation state knowledge, RCS performance (variations in thrust and I_{sp}), RCS misalignment, and an estimate of injection errors from a Monte Carlo simulation of the Ascent phase.

Table 5. ΔV Dispersion Analysis for Altair LLO RPOD.

	1σ ΔV Dispersion					
	FLAK	Orion State Error	Thrust / Isp	Injection	Misalignment	All Errors

24.315

46.723 m/s

The five individual error sources were then simultaneously run to estimate the total profile ΔV dispersion. Since each burn occurs in the orbital plane, the nominal plus 3-σ values were summed (rather than RSS'd) to obtain a conservative estimate of total dispersion ΔV. The resultant value was an additional 22.4 m/s above the nominal ΔV, raising the total RPOD ΔV budget to 47 m/s. The results of this analysis is provided in Table 5.

As part of the next analyses cycles, the dispersion sources will be updated with improved vehicle knowledge, and ANTARES simulation development will provide independent verification.

VI. Conclusion

The current Altair maneuver budget captures all the major ΔV needs for this set of missions (Global Access Sortie, Outpost, Cargo) and is a good foundation from which to continue improving fidelity of the trajectory analyses and simulation development. The Altair lunar lander ΔV budget estimate and the dispersions for that estimate will continue to mature in fidelity and accuracy over time as the Altair Project Office (APO) Integrated Performance Team develops the Altair vehicle models and trajectory simulations further.

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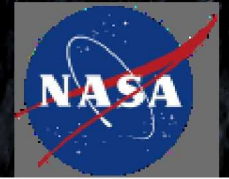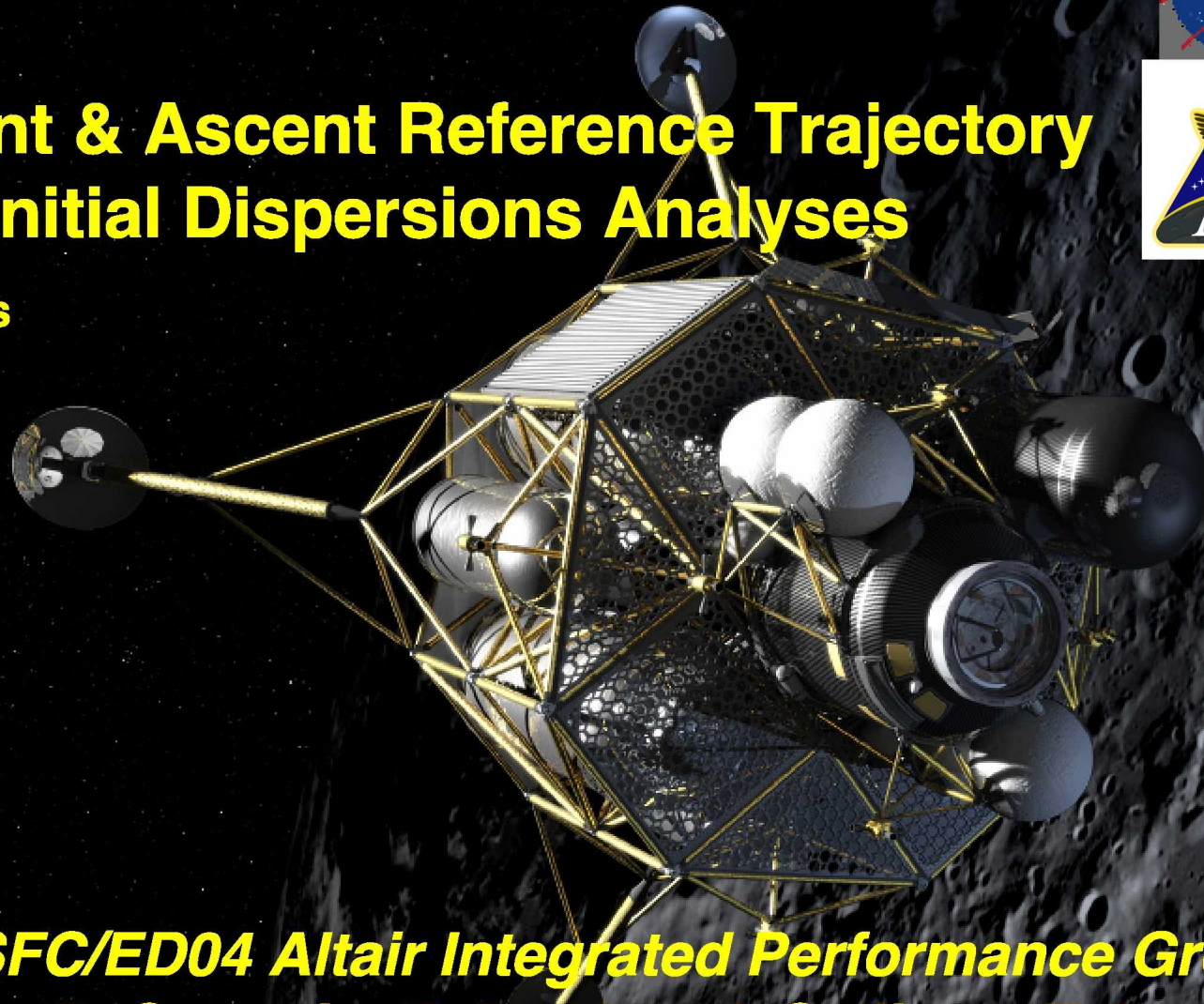
Mission Phases/Outline:

♦ **Trans-Lunar Cruise/Coast**
  • TCM ΔVs & dispersions

♦ **Lunar Orbit Insertion**
  • LOI ΔV(s) & clean-up

♦ **Low Lunar Orbit Loiter**
  • Attitude and altitude maintenance

♦ **Plane Change ΔV**

♦ **Descent Orbit Insertion ΔV**

♦ **Powered Descent**
  • ΔV & dispersions

♦ **Ascent**
  • MPS ΔVs & dispersion

♦ **Rendezvous, Prox Ops & Docking**
  • RCS ΔVs & dispersions

Airlo
Ascent Mod
Descent Mod
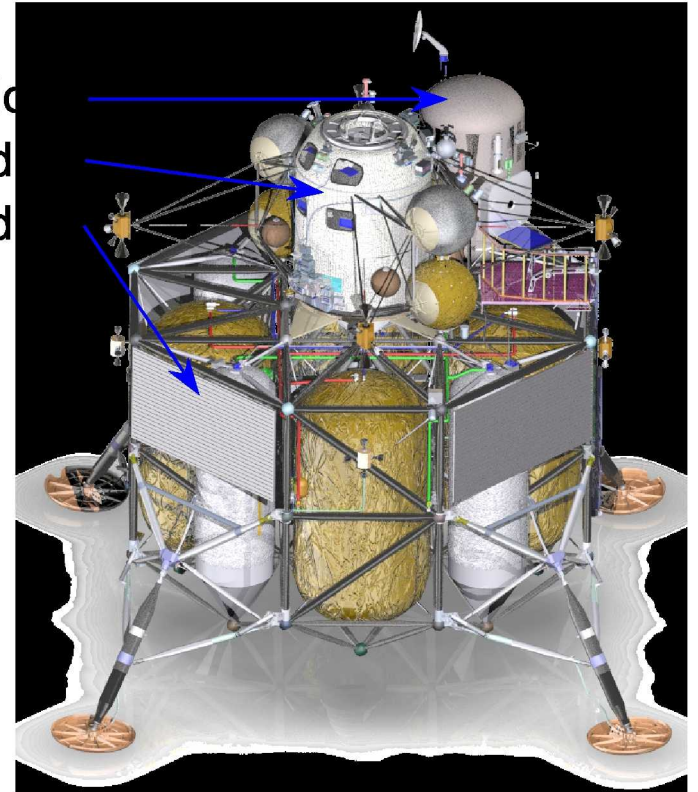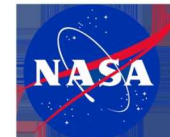
Altair Lunar Lander (LL)
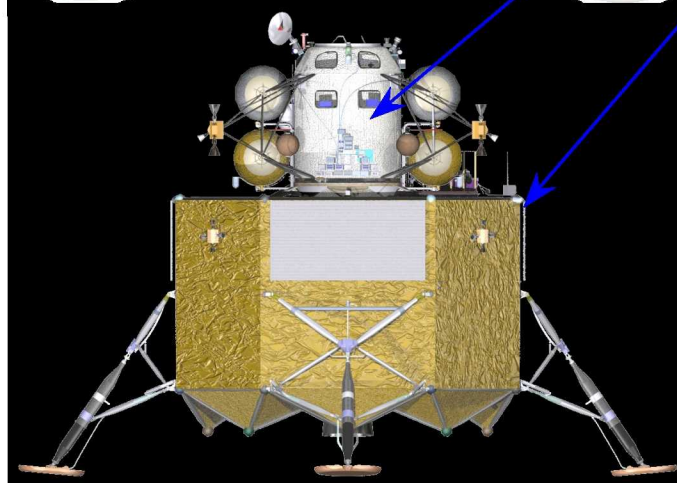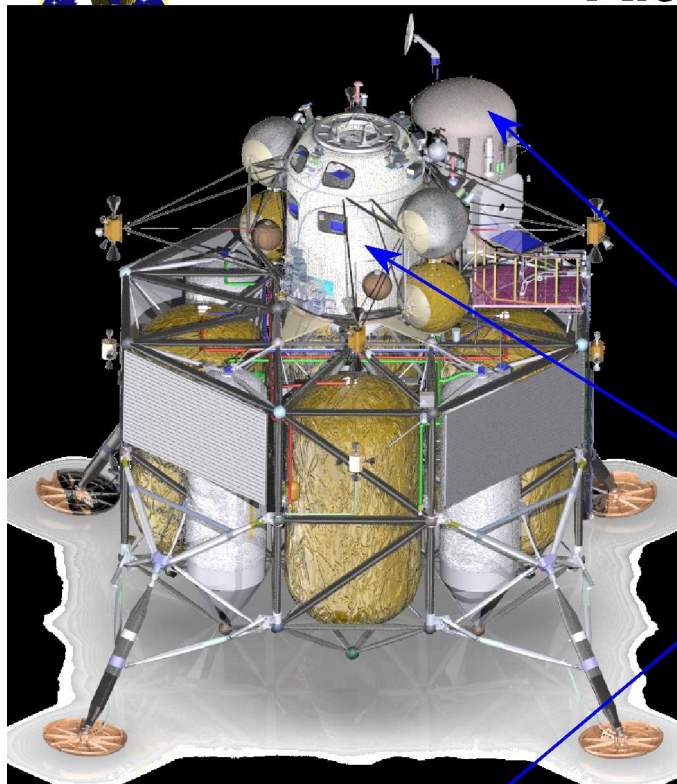
- ♦ **Altair -- overall L**
  - 9.1 m x 9.6 m

- ♦ **Airlock (AL)**
  - 1.75 m x 2.75 m

- ♦ **Ascent Module (AM**
  - 2.5 m x 5 m x 5.8

- ♦ **Descent Module (D**
  - 8.2 m x 7.8 m 6.2

Front View

Top View
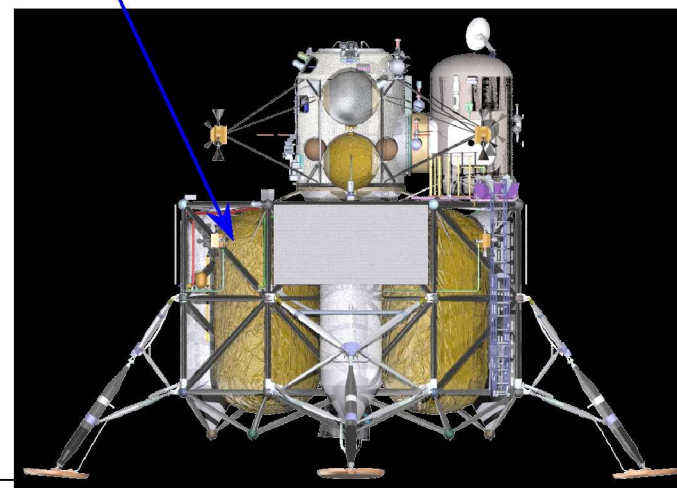
Side View

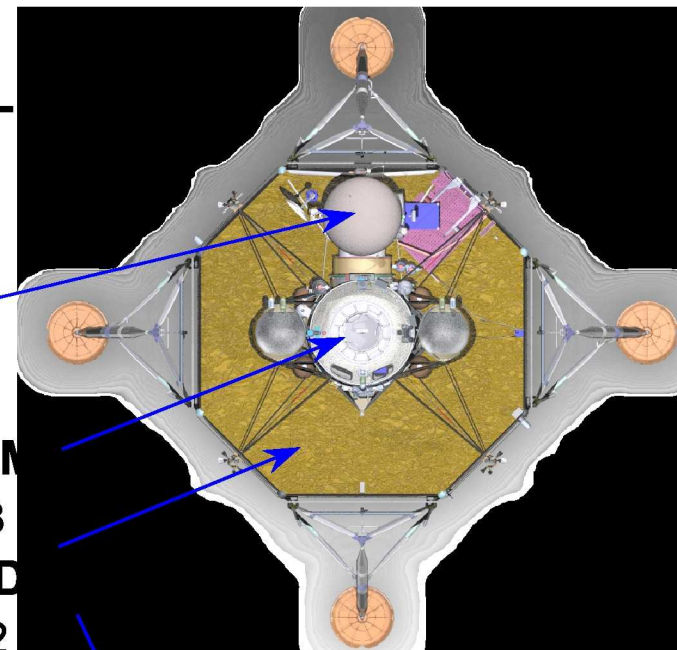# Trans-Lunar Injection (TLI) & Trans-Lunar Cruise (TLC)

♦ **EDS completes the TLI burn (up to ~3165 m/s + gravity-losses) and separates from the LL/Orion (CEV)**

♦ **TLC is between 3.7 to 4 days from TLI to Lunar Orbit Insertion (LOI)**

♦ **The LL/CEV stack will fly w/ the CEV-tail-to-Sun most of the time**

- This attitude will have the Sun at the head of the lander, with the LL partially shaded by the CEV.
- Due to the unknown direction the DM RCS thrusters will have to fire to perform the Trajectory Correction Maneuvers (TCMs/MCCs), the LL/CEV stack will need to alter its coasting attitude for the TCMs/MCCs, then return to the preferred coasting attitude.

Trans-Lunar Cruise/Coast Trajectory (Earth-fixed frame, Copernicus Trajectory Tool)

Low Lunar Orbit (LLO) & S.P. Descent Trajectory

- **LOI burn puts the LL/CEV into a 100 km circular orbit ~above the lunar poles**
- **After a LLO loiter of 1-5 days duration, the LL and CEV will separate**
  - CEV will do the separation burn
- **LL will do up to 1° plane change ~3/4 rev before descent orbit insertion (DOI)**
  - ~20-second burn with DM main engine at 60% (not required for Cargo or Sortie DRMs)
- **DOI to target South Pole LZ uses the four 100-lb$_f$ axially pointing DM RCS thrusters**
  - ~6 minutes for the crewed mission; ~9 minutes for cargo

De-orbit
Burn (DOI)

Low Lunar
Orbit Phase

100 x 100 km

Transfer
Orbit Coast

100 x 15.24 km

Landing Site/Zone (LZ)

Powered Descent (PDI) Burn
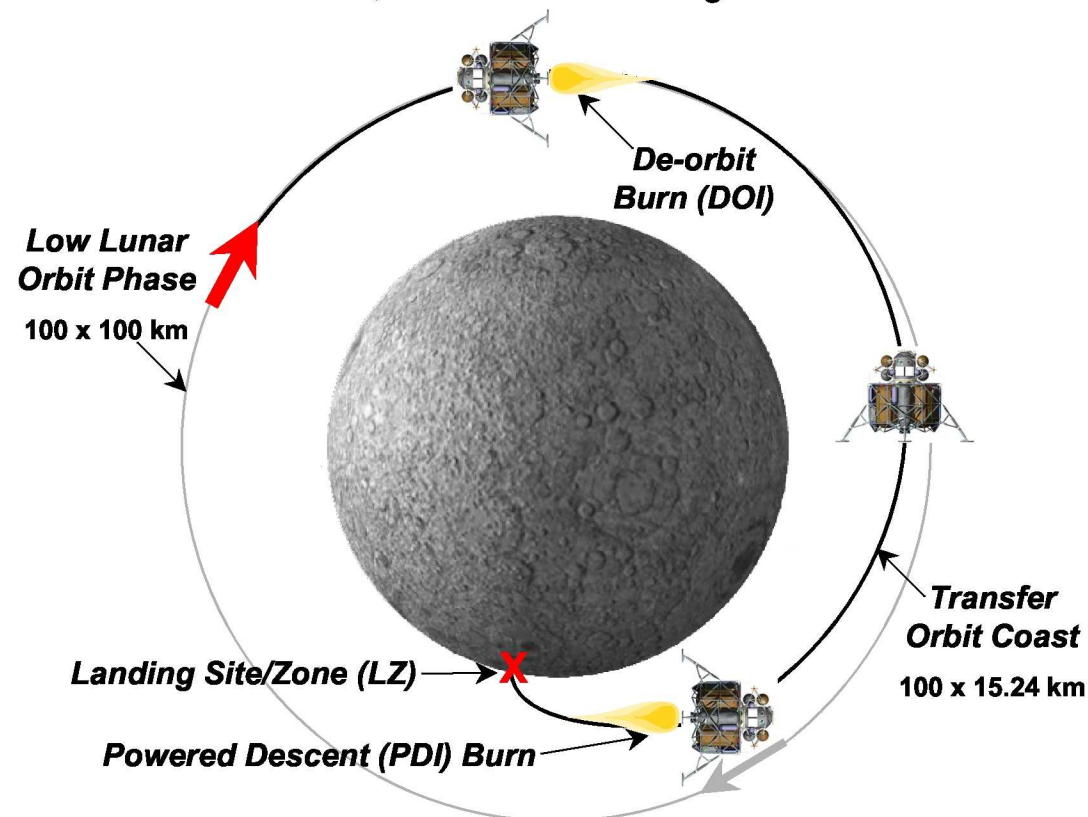
Altair -- Baseline Separation in Polar Orbit

Undocking to Landing Maneuver Sequence -- Free Return Separation
Initial orbit: Polar

1: Back-away (0.0 rev)
4: DOI (2.0 rev)

3: Plane Change (1.25 rev)

Equator: ———
Moon's Rotation: ⟶

5: Landing (2.5 rev)
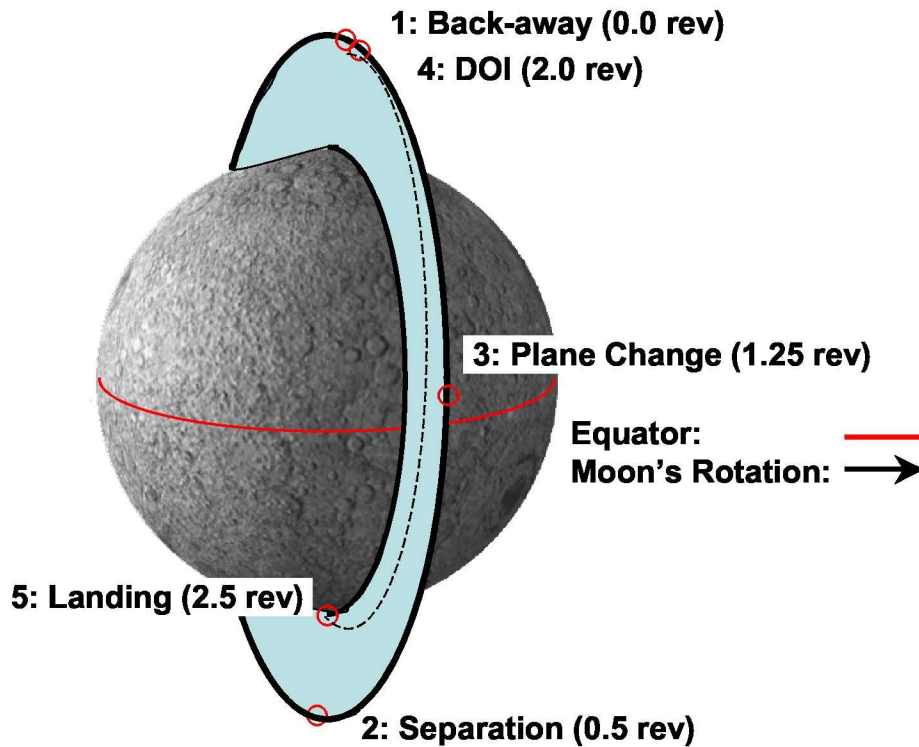
2: Separation (0.5 rev)

Trajectory Event	Phase Elapsed Time	Active Vehicle
	hh:mm:ss	
Undocking	00:00:00	Orion
Backaway 1	00:00:00	Orion
Backaway 2	00:01:00	Orion
Separation	01:00:04	Orion
Plane Change	02:16:04	Altair
DOI	03:55:58	Altair

Aerosciences & Flight Mechanics Division
Robert Merriam / JSC / EG5 / 281-483-3298

6

Lunar Descent Trajectory Phases

Powered Descent Initiation (PDI)

Braking Phase
h = 15,240 m (15.24 km)
R = ~560,000 m (560 km)

Braking Burn Maneuver
Efficiently reduce velocity from orbital speeds (92-95% throttle)

Pitch Up Phase
h = 400 m
R = 1300 m

Pitch-up Maneuver
15 sec pitch-up & throttle down phase

100 km

Deorbit maneuver

Transfer Orbit Phase (coast)

Powered Descent Phase

Approach Phase
h = 270 m
R = 850 m

Approach Phase:
- **Duration = 77 sec;**
- **16° approach path angle**

Approach Maneuver
Variable duration guidance phase (HD/HI/HA)

Final Descent Phase
h_0 = 30 m

Terminal Phase/Vertical Descent Maneuver to Touchdown
30 sec vertical descent to surface at constant acceleration

NOTE –
Not to scale

Touchdown

7

Lunar Descent Guidance

- ◆ **Algorithm based upon the Apollo Lunar Module Guidance:**
  - Acceleration profile is modeled/restricted to a quadratic (in each axis):

$$a = c_0 + c_1 t + c_2 t^2$$

- ◆ **Solve for coefficients at each guidance call**
  - Solve for c_1 and c_2 using r_0 and v_0 (t_{go}, r_{t_f}, v_{t_f}, a_t are given)

$$c_0 = a_t - 6 \frac{(v_t + v_0)}{tgo} + 12 \frac{(r_t - r_0)}{tgo^2}$$

$$c_1 = -6 \frac{a_t}{tgo} + 6 \frac{(5v_t + 3v_0)}{tgo^2} - 48 \frac{(r_t - r_0)}{tgo^3}$$

$$c_2 = 6 \frac{a_t}{tgo^2} - 12 \frac{(2v_t + v_0)}{tgo^3} + 36 \frac{(r_t - r_0)}{tgo^4}$$

a = acceleration command
t = time since last guidance call
tgo = time to go until end of phase
v_0 = current velocity
r_0 = current position
a_t = acceleration target (at end of phase)
v_t = velocity target (at end of phase)
r_t = position target (at end of phase)

- ◆ **At the time of each guidance call, t=0, therefore the command is equal to c_0**
  - c_1 and c_2 only needed between guidance cycles or near end of phase
- ◆ **Analytic expression for velocity & position can be obtained from this solution**

Lunar Descent Dispersions

- **Initial state at Powered Descent Initiation**
  - PDI: -71.6° latitude, -179.8° longitude, 15.24 km altitude
  - Landing target (LZ): -89.9° latitude, 0° longitude
- **Perfect Navigation**
- **Guidance Phases**
  - Braking and Approach Phases: Quadratic acceleration profile
  - Terminal Phase: Vertical descent at constant velocity following a proportional error feedback law
- **Vehicle Mass Properties**
  - Constructed of 5 mass bodies (wet AM, dry DM, AL, 2 dynamic masses DM main engine propellant, DM RCS propellant)
- **DM Effectors**
  - Gimbaled Main Engine, Max thrust: 18,627 lb_f, Isp = 448.6 sec
    - Thrust Vector Control with a proportional-derivative controller
  - Reaction Control System:
    - Sixteen 100 lb_f jets grouped into 4 orthogonal clusters, 0° skew
    - Phase-plane controller

- **Calculated cumulative 3σ effect by taking root-sum-square of contributing terms: Thrust, I_{sp}, Prop mass, c.g. location, inertias):**
  - $3\sigma_{cumulative} = \sqrt{[(3\sigma_{Thr})^2 + (3\sigma_{Isp})^2 + 3\sigma_{prop})^2 + (3\sigma_{c.g.})^2 .. + (3\sigma_{Ixx})^2 + ..]}$

- **Cumulative dispersion effect on landing state and ΔV:**
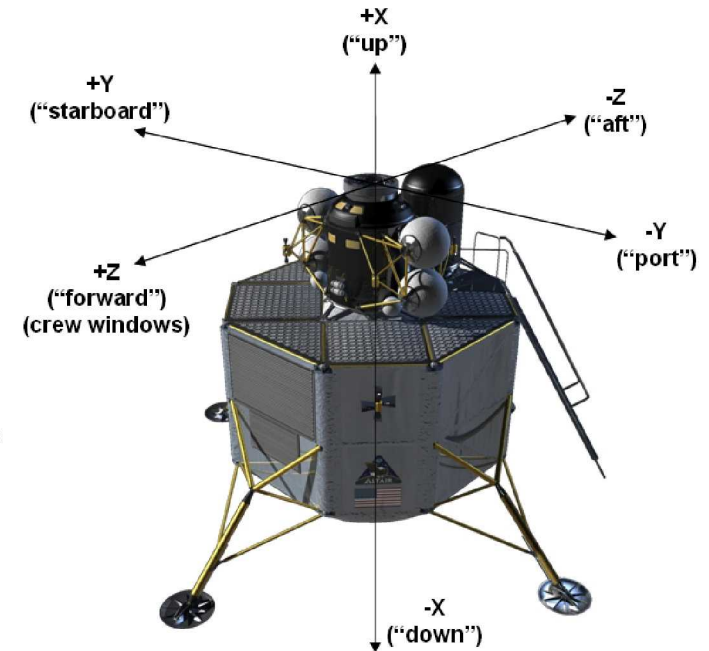
	Cumulative 3σ
Range	14.1 m
Velocity	3.6 m/s
Plumbline Pitch	23.4°
Attitude Rate	26.2°/s
Propellant	443.4 kg
ΔV	**28.3 m/s**

- **With a subset of the variables that could possibly be dispersed, 28.3 m/s is less than the Apollo 11/12 ΔV dispersion of 53 m/s.**

LDAC-4 DM Sortie/Outpost & Cargo ΔVs
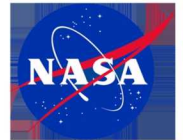
Event #	Event / Description	Piloted LL ΔV (S / O, m/s)	Burn time (S / O, min)	Cargo LL ΔV (m/s)	Burn time (min)
1-4	Midcourse Correction (MCC)	≤25.4	≤15.3	≤23	≤11.5
5	Propellant Settling Burn	0.55	0.3	0.65	0.3
6	Lunar Orbit Insertion (LOI)	953 / 894 * (3-burn / 1-burn LOI)	11.0 / 10.4	892* (1-burn LOI)	8.7
7	LOI Clean-up	≤7	≤3.4	≤7	≤2.9
8,9	Altitude Maint. (Attitude=30kg/day)	≤10.1 / 0	≤4.8 / 0	--	--
10	Plane Change (after CEV Sep.)	0 / ≤28.5	0 / ≤0.3	--	--
11	De-Orbit Maneuver (DOI)	19.4	5.6 / 5.7	19.4	7.9
12	Propellant Settling Burn	2.15	0.6	1.65	0.7
13	Powered Descent (PD+FAL)	2016*	12.8 / 13.1	2149*	17.5
	Crew-in-the-Loop	0	--	--	--
	RCS/ACS for Powered Desc.	11	--	11	--
	Dispersions	≤53	incl. above	≤53	incl. above
	Landing site re-designation	5	incl. above	0	--

Total: 3026/2996 m/s
(RCS = 76 / 66 m/s)

Total: 3094 m/s
(RCS= 63 m/s)

* These change depending on LL mass used (as a result of the vehicle T/W during LOI & PDI+FAL)

PEG Phase

Insertion:
R, V targets
γ = 0

SORT: Single axis Rotation (SAR)

ANTARES: Linear pure pitch rotation
[later will be an optimal Single Axis Rotation (SAR) Phase]

100-m Vertical Rise

Ascent Time (to 1st perilune), sec

Ground Range, sec
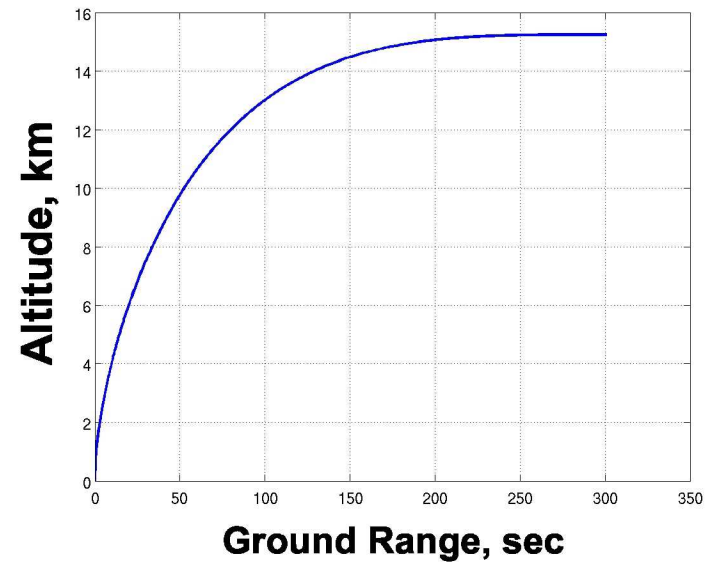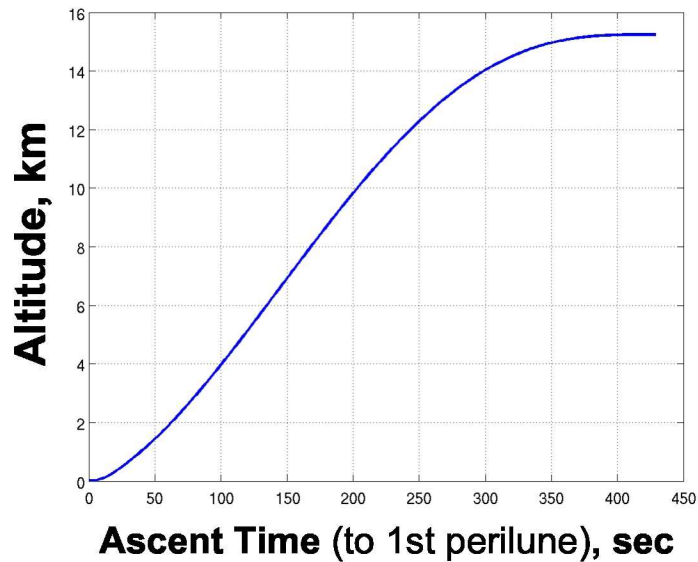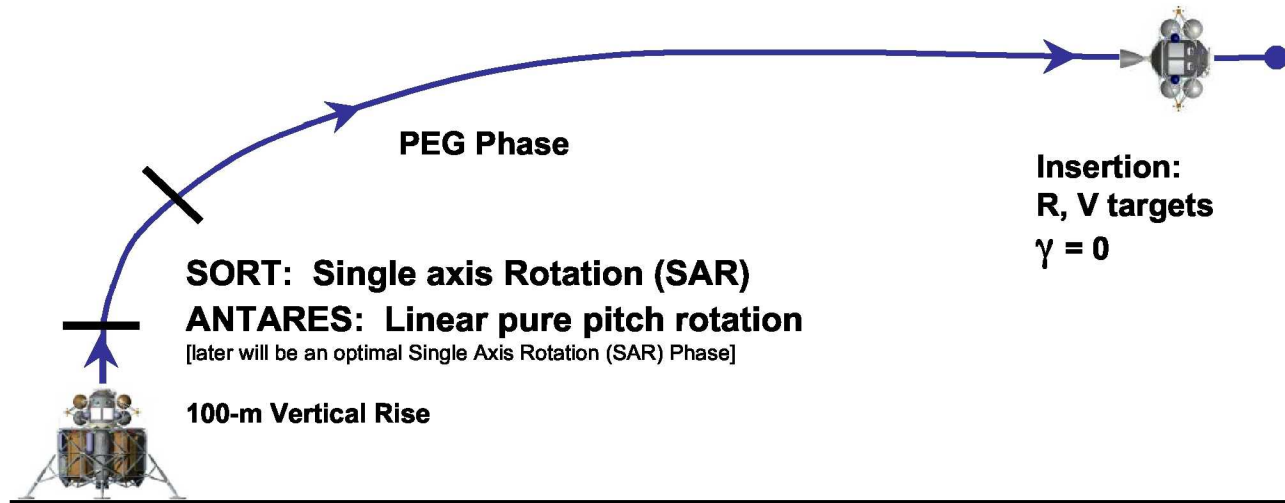
Lunar Ascent Trajectory Profile/Guidance

Optimal Ascent Region (to orbital insertion):
- Governed by Optimal Control Theory -- PEG
- Majority of ascent
- Final orbit: 15.24 km x 86.2 km perilune x apolune alt.

Transition Region (SAR):
- Initial FPA to Optimal FPA
- Pitch Rate ~5°/sec
- ~10-15 seconds in length

Thrust Direction

Vertical Rise Region:
- Constant Flight Path Angle (FPA)
- Rise Altitude = 100 m
  (for terrain clearance)
- ~7-10 seconds in length
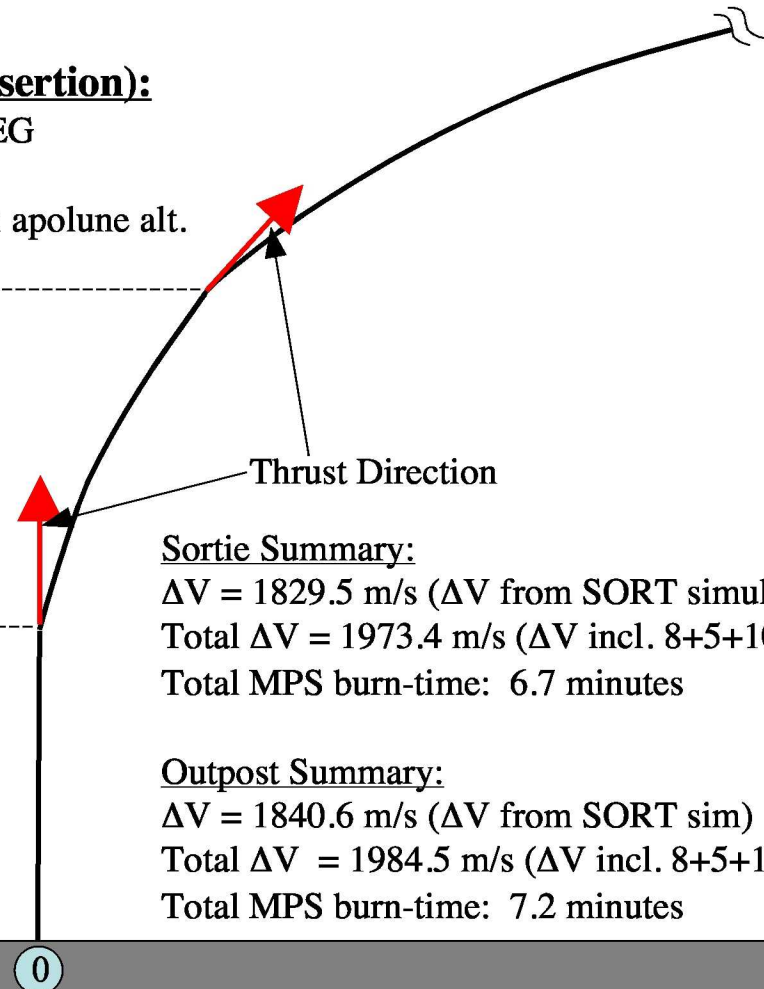
Sortie Summary:
ΔV = 1829.5 m/s (ΔV from SORT simulation)
Total ΔV = 1973.4 m/s (ΔV incl. 8+5+10+47+74 m/s)
Total MPS burn-time: 6.7 minutes

Outpost Summary:
ΔV = 1840.6 m/s (ΔV from SORT sim)
Total ΔV = 1984.5 m/s (ΔV incl. 8+5+10+47+74 m/s)
Total MPS burn-time: 7.2 minutes

0

Lunar Ascent Dispersions

- **Ascent Module (AM) Initial State**
  - 0° long., -89.9° lat.; Body +z-axis (crew windows) facing north
- **Perfect Navigation**
- **Guidance Phases**
  - 100-meter vertical rise
  - Linear pure pitch rotation or SAR
    - 5°/s pitch, Single-Axis time optimal Rotation (SAR) equations are derived
  - Powered Explicit Guidance (PEG) to orbit insertion (15.24 x 86.2 km alt.)
- **AM Mass Properties**
  - Constructed of 4 mass bodies (dry AM, 4 propellant dynamic masses)
- **AM Effectors**
  - Main Engine, Isp = 320 sec
    - Fixed gimbal aligned w/ vehicle c/l; Max thrust: 5500 lbf
  - Reaction Control System, Isp = 300 sec
    - 20 jets grouped into 4 clusters; each cluster: two 200lb$_f$ thrusters, ±X; one 100lb$_f$ thruster, 30° from ±Y towards -X; two 25lb$_f$ thrusters, 45° from ±X towards ±Z

+X ("up")
+Y ("starboard")
+Z ("forward") (crew windows)

Pitch Torque about Body y-axis (N*m)

Simulation Time (sec)
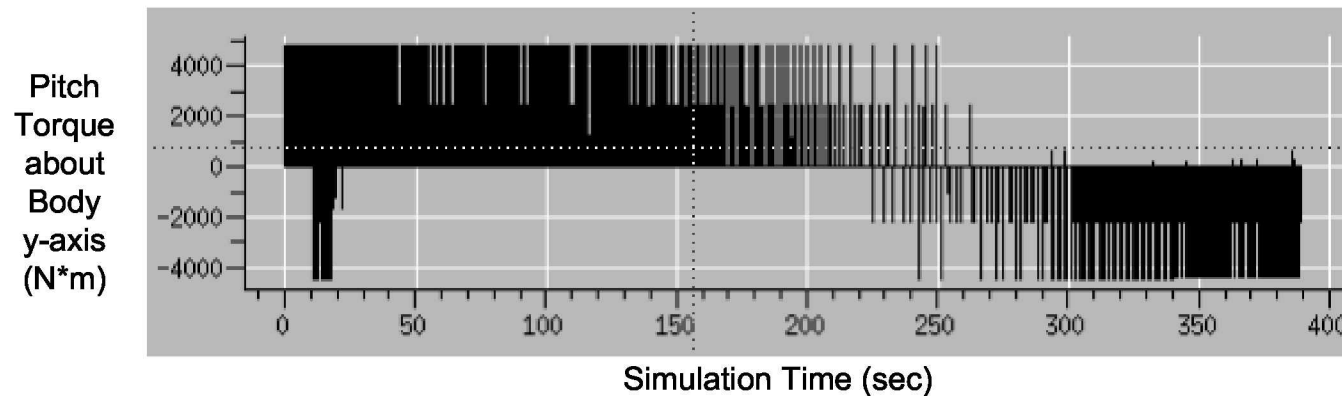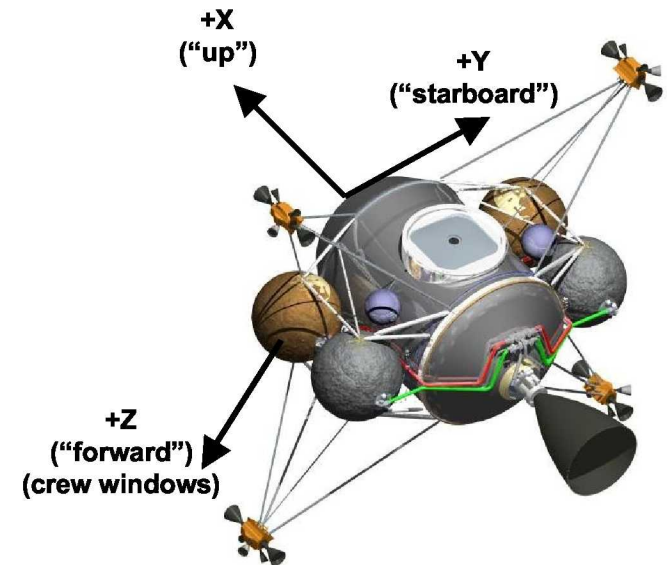
Lunar Ascent Dispersions

- **Initial ANTARES ascent simulation uses ~17 kg of RCS propellant**
  - Ascent engine was canted 2.9 from the center-line to align with c.g. location at mid-burn
    - Large improvement from early cases
- **Apollo ascent RCS fuel consumption was around 50 kg**
  - Some of this prop was used to assist in liftoff thrust-to-weight
- **Best case result** (perfect world, constantly centered/balanced c.g.) **requires 11 kg**

Final State	Nominal Result		All 12 dispersed simultaneously
Burn Duration (s)	389.35	mean	389.16
		3σ	8.23
Orbital Radius Magnitude (m)	1,752,704	mean	1,752,703
		3σ	8.40
Orbital Velocity Magnitude (m/s)	1623.02	Mean	1623.12
		3σ	1.31
Attitude Rate Magnitude (°/s)	0.71	mean	0.57
		3σ	0.75
RCS Propellant (kg)	16.46	mean	19.10
		3σ	7.50
ΔV (m/s)	1779.03	mean	1779.27
Dispersion result for this subset of variables:		**3σ**	**3.74 m/s**

SORT result gives 5.1 m/s

Post-chatter RCS: 16.5 kg!

RCS Propellant (kg)

Ascent w/ 2.9° engine cant

Simulation Time (s)

Relative Motion Between Altair AM and Orion

Vbar (LVLH +X)

R-Bar Acq. 2000 m

MC$_1$-MC$_3$

LIDAR Acq. 5000 m

TPI

$\Delta V = 19$ m/s
x = -25 km
z = 14 km

	Time (h:mm:ss)	Range (m)	Reference Location	Event Description
1	0:00:00	272,449	C.G.	MECO/Insertion
2	0:10:00	197,827	C.G.	NCC Maneuver
3	0:56:10	28,615	C.G.	Terminal Phase Initiation
4	1:07:33	16,318	C.G.	1st Midcourse Maneuver
5	1:18:56	7780	C.G.	2nd Midcourse Maneuver
6	1:24:43	5000	C.G.	LIDAR Acquisition
7	1:30:18	3143	C.G.	3rd Midcourse Maneuver
8	1:35:18	2000	C.G.	R-bar Acquisition

NCC

From Nominal Injection
$\Delta V = 1.75$ m/s

80,000 m

Rbar (LVLH +Z)

Relative Altitude (km)

Downrange (km)

Approx. to Scale

15

LDAC-4 Sortie & Outpost Ascent ΔVs

Event #	Event / Description	Sortie AM ΔVs (m/s)	Outpost AM ΔVs (m/s)
1	Powered Ascent, deterministic	1829.5 †	1840.6 †
	Ascent Launch Window (mainly Sortie)	8	8
	Ascent Dispersions (MPS)	+5.1	+5.1
	MPS ΔV total:	**1842.6**	**1853.7**
	Ascent RCS	10*	10*
2	Ascent Abort	74*	74*
3	(Number) Corrective Combination	1.75*	1.75*
4	TPI	19.0*	19.0*
5-13	Rend., Prox Ops & Docking, deterministic	3.7*	3.7*
	RPOD Dispersions	22.4*	22.4*
	RPOD ΔV subtotal:	+46.8*	+46.8*
	RCS+RPOD ΔV total:	**130.8***	**130.8***
	Total ΔV (MPS+RCS):	**1973.4**	**1984.5**

† These ΔVs are generated by the post-LDAC-3 AM trajectory/sizing curve fit.

* This is all done using RCS/ACS propellant.

Lander Mission Flight Phases
1-page Summary

- ◆ **Low Earth Orbit (LEO)** — **241 km** (130 nmi) **circ., 29° incl.**
  - Rendezvous & Docking — As soon as Flight Day 1; As late as Flight Day 3
  - Loiter — 4 days max (~93 hours)
  - Attitude — Gravity gradient except for RPOD

- ◆ **Trans-Lunar Injection (TLI) & Coast** — **4-day trip** (nominal)
  - Trajectory Correction Maneuvers (TCMs) — 4 TCM/MCCs @ 25.4 m/s (23 m/s for cargo)

- ◆ **Lunar Orbit Insertion (LOI-1, -2, -3)** — **3-/1-burn ΔV = 953/894 m/s** (Cargo 892 m/s)

- ◆ **Low Lunar Orbit (LLO)** — **100 x 100 km circular, ~90° incl.**
  - TCM/MCC during loiter in LLO — 1 per day very likely, $|\Delta V|$ uncertain (possibly 2)
  - Separation from CEV (= active) — Polar: 2.5 revs prior to TD @ LZ / Global: 1.5 hrs
  - Plane Change (Pl.Ch.) for Polar LZ w/ MPS — 1° = 28.5 m/s
  - Descent Orbit Insertion (DOI) w/ RCS — 6 min (from 100x100 km to 100x15) (9 min. cargo)

- ◆ **Descent** — **100 x 15.24 km elliptical**
  - Powered Descent Initiation (PDI) — 2074 m/s (Piloted), 2202 m/s (Cargo)
  - Duration, minutes: PDI to TD — 11.1 (Sortie), 10.5 (Outpost), 16.2 (Cargo)
  - Braking Burn (BB) — ~92-95% throttle
  - Approach Phase — ~77 sec
  - Terminal/Final Approach & Landing — (FAL) ~30 sec

- ◆ **Ascent** — **6.7-7.2 minutes total** (MPS to MECO only)
  - Vertical Rise — 100 m in ~8-10 sec
  - Single Axis Rotation (SAR) — ~5°/s for ~10-12 sec
  - Powered Explicit Guidance (PEG) — Rest of AM MPS burn (~380-410 sec) to 15x86 km

- ◆ **LLO Rendezvous & Docking** — **Nominal = $\leq$3 hrs, 47 m/s** (14+ burns)

- ◆ **AM De-orbit** — 0 m/s currently (nominal 27-60 m/s)

Trajectory/Performance Back-up

ACS:	Attitude Control System
ADD:	Architecture Definition/Design/Description Document
ALHAT:	Autonomous Landing and Hazard Avoidance Technology
AM:	(Altair) Ascent Module
ANTARES:	Advanced NASA Technology ARchitecture for Exploration Studies
BB:	Braking Burn
CEV:	Orion / Crew Excursion Vehicle
DM:	(Altair) Descent Module
DOI:	Descent Orbit Insertion (or De-Orbit Insertion)
DRM:	Design Reference Mission (Sortie, Outpost or Cargo)
EDS:	Earth Departure Stage (on Ares V LV)
FAL:	Final Approach and Landing
HA:	Hazard Avoidance (3rd step)
HD:	Hazard Detection (1st step)
HI:	Human Interaction (2nd step)
IVDDD:	Integrated Vehicle Design Definition Document
LDAC-4:	Lander Design & Analysis Cycle #4
LEO:	Low Earth Orbit (100 - 130 nmi)
LLO:	Low Lunar Orbit (typically 100 km)
LOI:	Lunar Orbit Insertion (up to 3 burns)
LV:	Ares V Launch Vehicle
LZ:	Landing Zone (Landing Site)
MAVERIC:	Marshall Aerospace VEhicle Representation In "C"
MC/MCC:	Midcourse Correction (same as TCM)
MECO:	Main Engine Cut-Off (for lunar ascent)
MMOD:	Micro-Meteoroid & Orbital Debris
MPS;	Main Propulsion System (AM or DM)
NCC:	(Number) Correction Change (during lunar ascent)
PD/PDI:	Powered Descent Initiation
PEG:	Powered Explicit Guidance
RAC:	Requirements Analysis Cycle
RCS:	Reaction Control System
RPOD:	Rendezvous, Proximity Operations & Docking
SAR:	Single Axis Rotation (during AM ascent)
SORT:	Simulation to Optimize R_ Trajectories
SPAR:	Spacecraft Performance Analysis Report
TCM:	Trajectory Correction Maneuver (same as MCC)
TLC:	Trans-Lunar Coast or Trans-Lunar Cruise
TLI:	Trans-Lunar Injection
TPI:	Terminal Phase Initiation (during RPOD)

ALTAIR
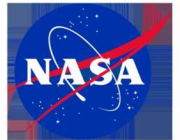
Example Trajectories for Trans-Lunar Coast
Overlay of trajectory & landing site lighting conditions at arrival (Lunar "morning")

NASA

Site	Name / Description	Lat.	Long.	Type
A	South Pole	89.9° S	180° W	Polar
B	Far-side Aitken Basin Floor	54° S	162° W	Other
C	Orientale Basin Floor	19° S	88° W	Near-side
D_{12}	Oceanus Procellarum	3° S	43° W	Near-side
E	Mare Smythii	2.5° N	86.5° E	Near-side
F_{11}	W/NW Mare Tranquillitatis	8° N	21° E	Near-side
G_{15}	Rima Bode	13° N	3.9° W	Near-side
H	Aristarchus Plateau	26° N	49° W	Near-side
I	Central Far-side Highlands	26° N	178° E	Far-side
J	North Pole	89.5° N	91° E	Polar

Last Quarter

B, I

New Moon

A, E, J

Full Moon

C

F

D, H

G

First Quarter
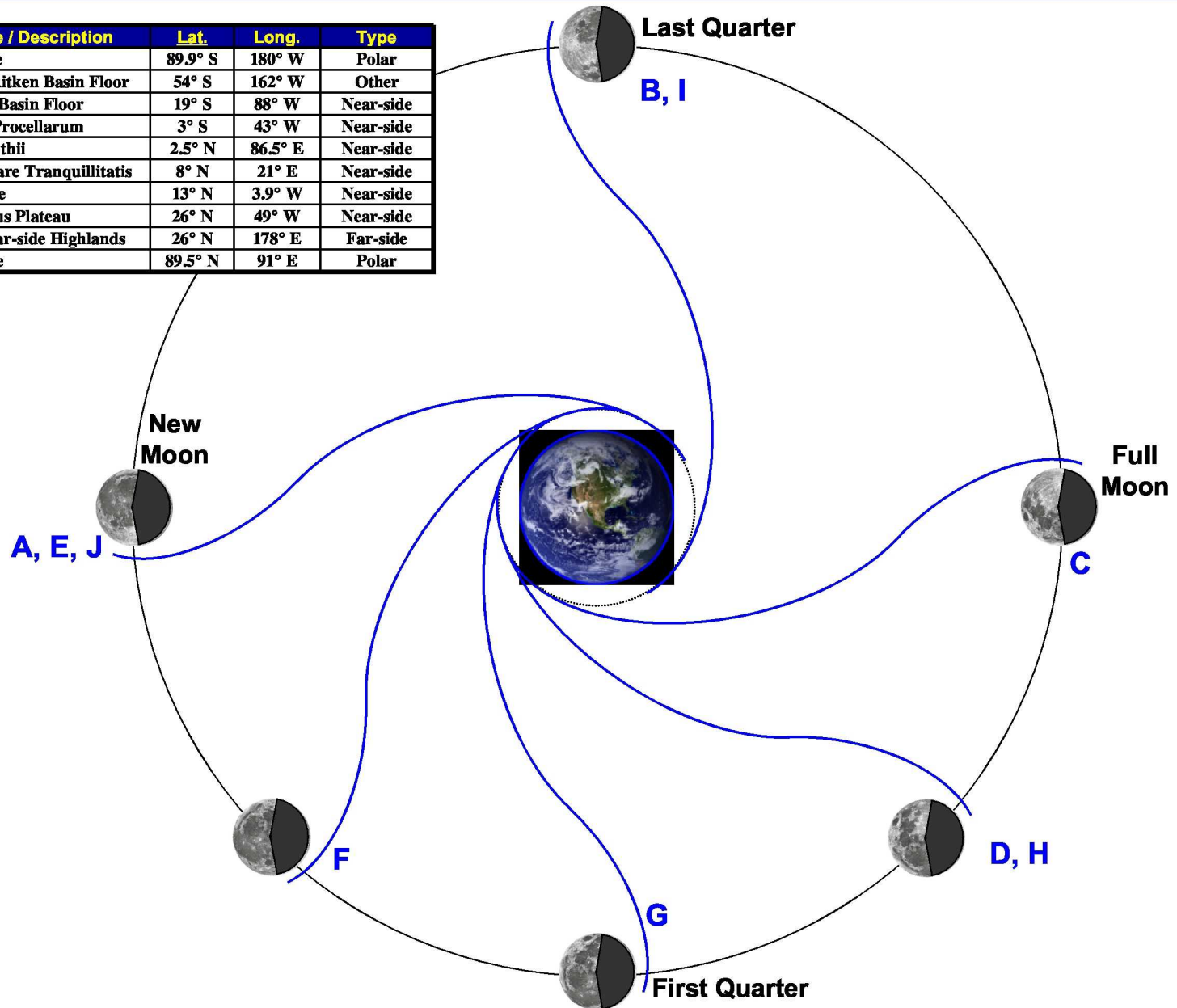
The 10 ESAS Sites & The 6 Apollo Sites

Site	Name / Description	Latitude	Longitude	Classification	# LOI Burns
A	South Pole	89.9° S	180° W	Polar	1
B	Far-side Aitken Basin Floor	54° S	162° W	Other	3
C	Orientale Basin Floor	19° S	88° W	Near-side	1 (probably) or 3 (unlikely)
D_{12}	Oceanus Procellarum	3° S	43° W	Near-side	1
E	Mare Smythii	2.5° N	86.5° E	Near-side	1
F_{11}	W/NW Mare Tranquillitatis	8° N	21° E	Near-side	1
G_{15}	Rima Bode	13° N	3.9° W	Near-side	1 (probably) or 3 (unlikely)
H	Aristarchus Plateau	26° N	49° W	Near-side	1 (probably) or 3 (possible)
I	Central Far-side Highlands	26° N	178° E	Far-side	1 (probably) or 3 (possible)
J	North Pole	89.5° N	91° E	Polar	1

Site	Name / Description	Latitude / Longitude Source: National Space Science Data Center		Classification	# LOI Burns (ΔV, m/s) Source: Apollo By The Numbers, SP-4029
Apollo 11	Mare Tranquillitatis	0.67° N	23.47° E	Near-side	1 (889 m/s)
Apollo 12	Ocean of Storms	3.01° S	23.42° W	Near-side	1 (881 m/s)
Apollo 14	Fra Mauro	3.65° S	17.47° W	Near-side	1 (921 m/s)
Apollo 15	Hadley-Apennine region	26.13° N	3.63° E	Near-side	1 (914 m/s)
Apollo 16	Descartes Highlands	8.97° S	15.50° E	Near-side	1 (854 m/s)
Apollo 17	Taurus-Littrow Highlands	20.19° N	30.77° E	Near-side	1 (911 m/s)

High Priority Lunar Exploration Sites
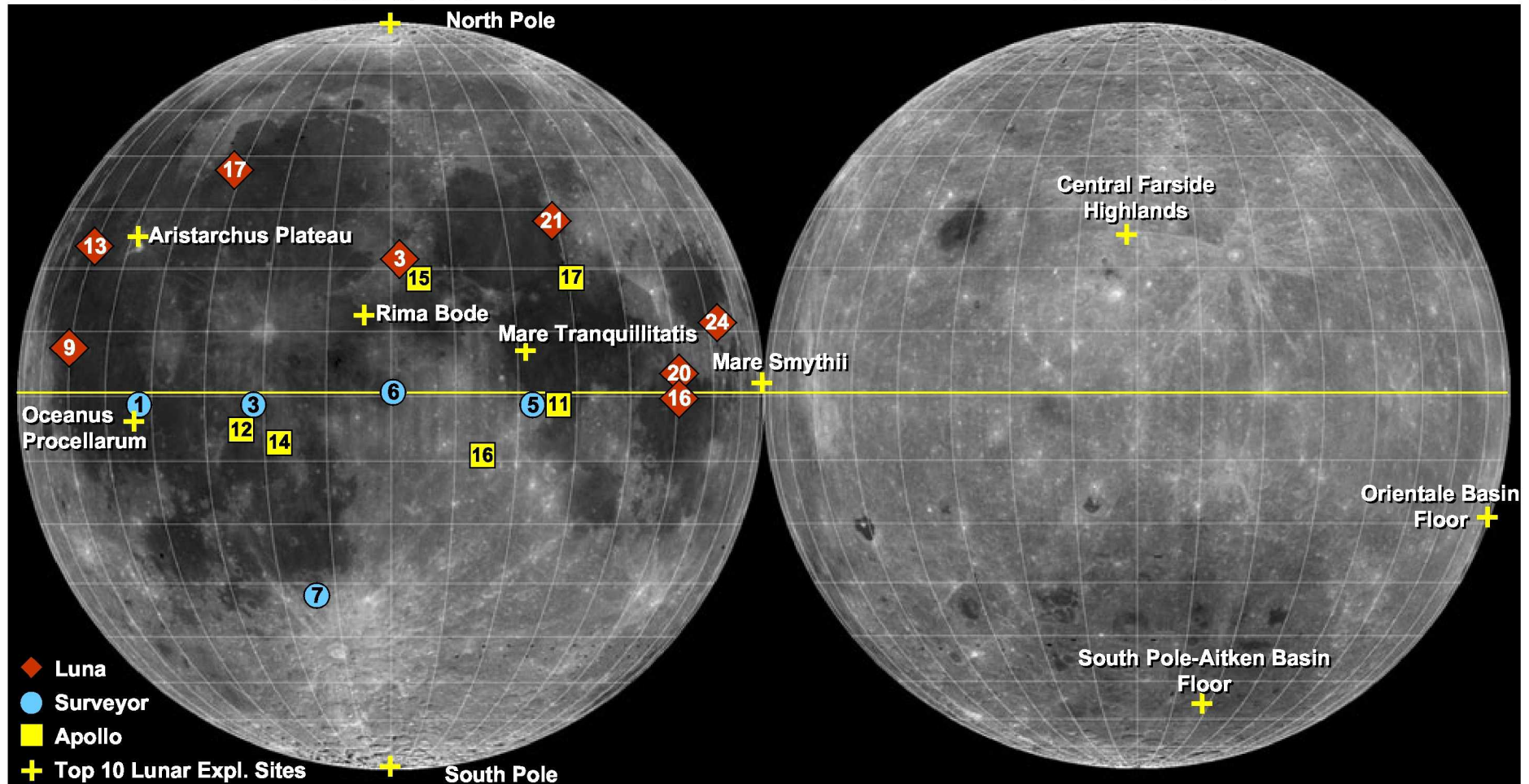
Near Side

Far Side

	Name	Latitude	Longitude	Classification
A.	South Pole	89.9° S	180° W	Polar
B.	Far-side South Pole Aitken Basin Floor	54° S	162° W	Other
C.	Orientale Basin Floor	19° S	88° W	Nearside
D.	Oceanus Procellarum	3° S	43° W	Nearside
E.	Mare Smythii	2.5° N	86.5° E	Nearside

	Name	Latitude	Longitude	Classification
F.	W/NW Mare Tranquillitatis	8° N	21° E	Nearside
G.	Rima Bode	13° N	3.9° W	Nearside
H.	Aristarchus Plateau	26° N	49° W	Nearside
I.	Central Far Side Highlands	26° N	178° E	Farside
J.	North Pole	89.5° N	91° E	Polar

High Priority Lunar Exploration Sites
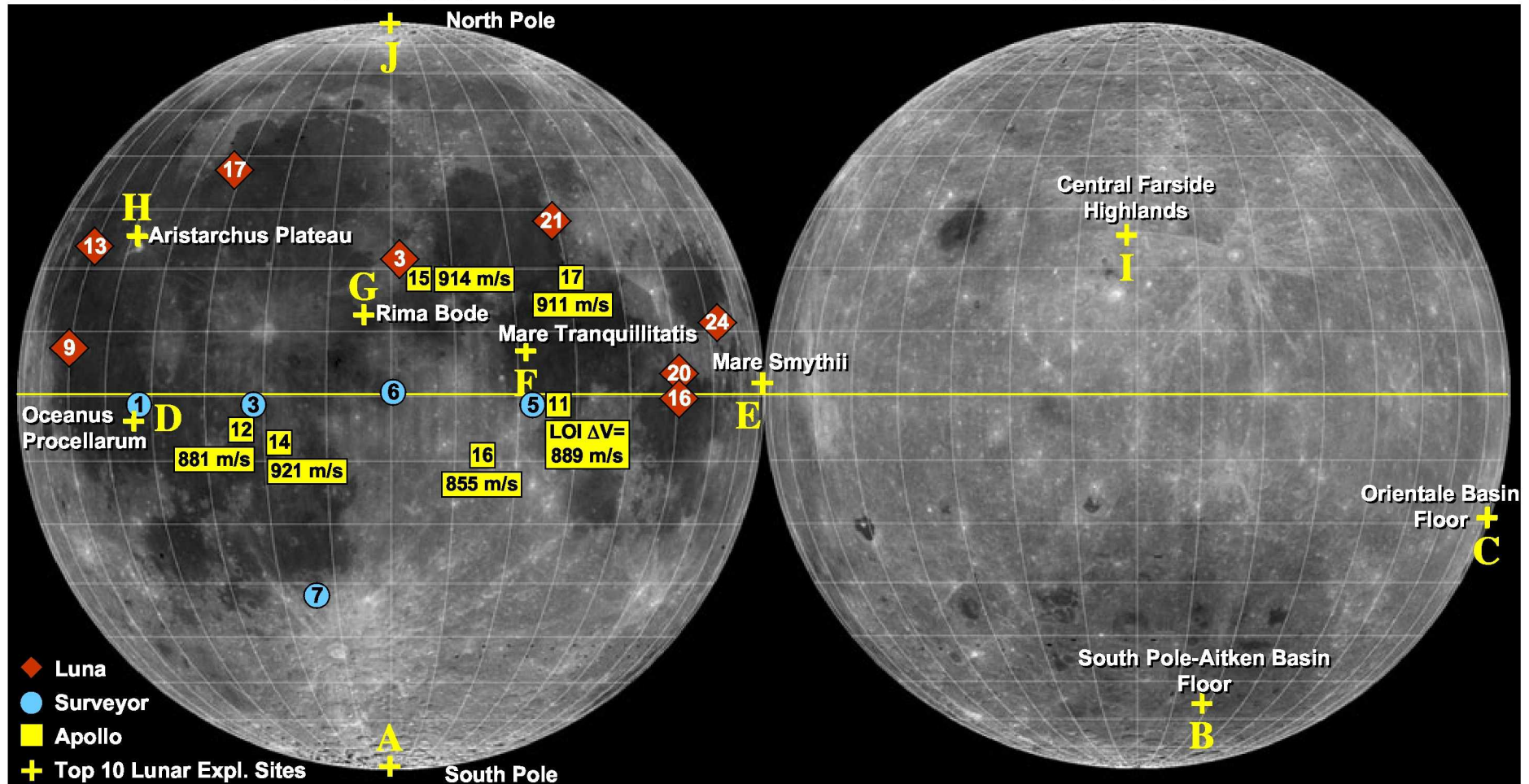
Near Side

Far Side

Site:	Description	Latitude	# LOI burns
A	South Pole	89.9° S	1
B	Far-side South Pole Aitken Basin Floor	54° S	3
C	Orientale Basin Floor	19° S	1 (probably) or 3 (unlikely)
D	Oceanus Procellarum	3° S	1 (probably) or 3 (unlikely)
E	Mare Smythii	2.5° N	1

Site:	Description	Latitude	# LOI burns
F	W/NW Mare Tranquillitatis	8° N	1 (probably) or 3 (unlikely)
G	Rima Bode	13° N	1 (probably) or 3 (unlikely)
H	Aristarchus Plateau	26° N	1 (probably) or 3 (possible)
I	Central Far Side Highlands	26° N	1 (probably) or 3 (possible)
J	North Pole	89.5° N	1

22

Geometry of the **Earth**-Moon System
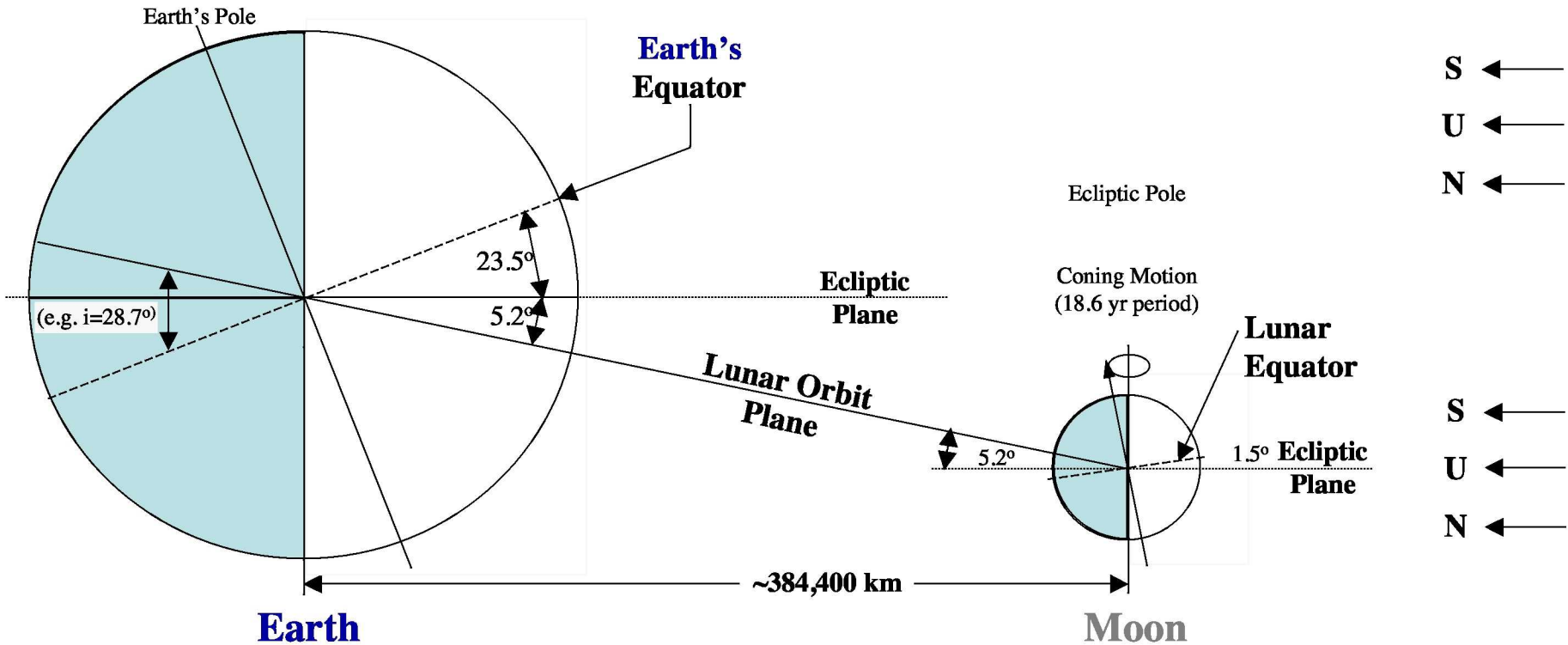
Translunar Injection and Coast

◆ **Other Views: Moon-centered** (top)**; Rotating** (bottom)
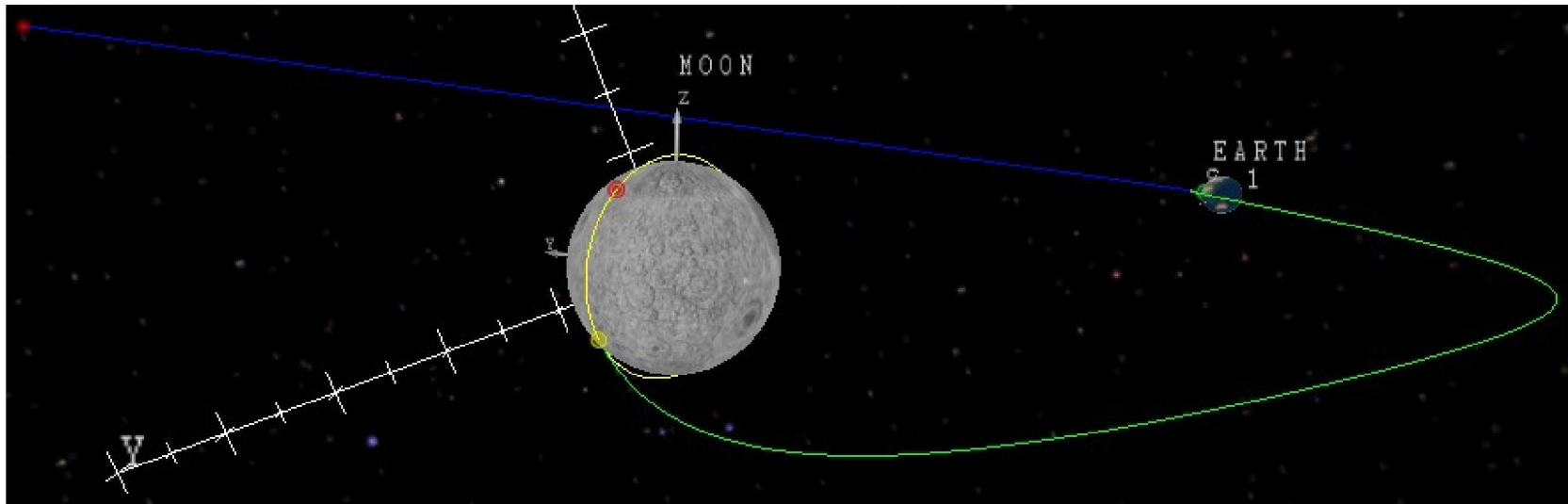
Trans-lunar coast trajectory (Moon-fixed frame)
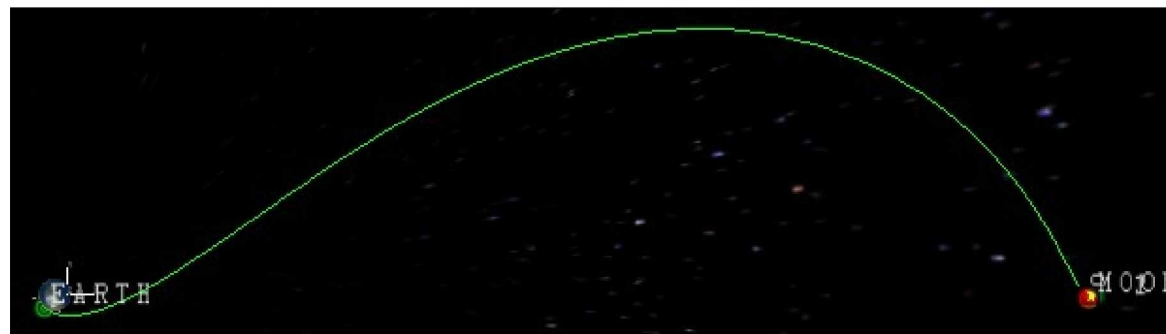
Trans-lunar coast trajectory (Rotating frame)

Trans-Lunar Coast -- Trajectory Correction Maneuvers

Event / Description	LDAC-1 ΔVs (Ltd., m/s)	LRAC-2 ΔVs (3σ, m/s*)	Apollo 14 ΔVs (ABTN, m/s)
Trajectory/Midcourse Correction (TCM/MCC)-1	2	21.5 (MET: TLI + 2 to 3)	21.7 (GET: TLI + 28:08)
TCM / MCC-2	0	2.0 (MET: TLI + ~24)	0
TCM / MCC-3	0	3.7 (MET: TLI + ~62)	0
TCM / MCC-4	0	2.6 (MET: LOI - 8)	1.1 (GET: TLI + 74:38)
Totals (Statistical: RSS'd, Actual: Summed):	2	25.4	22.8

GR&A for LRAC-2 TCM Analyses:

- ♦ **FLAK = un-Fortunate Lack of Acceleration Knowledge**
- ♦ **Maneuver execution errors (Gates Method)**
- ♦ **Dynamic modeling errors**
- ♦ **Measurement noise and modeling errors including**

* Derived from Mean, +1σ, and 95% ΔVs
ABTN -- Apollo By The Numbers, NASA SP-2000-4029; Apollo 14: Jan. 31, 1971, 4:03PM ET to Feb. 9, 4:05PM ET

"Apollo By The Numbers" (NASA SP-2000-4029)

Apollo Midcourse (M/C) Corrections -- Historical

Mission:	Outbound M/C 1: (m/s)	Outb M/C 2: (m/s)	Subtotal (m/s)	Inbound M/C 1: (m/s)	Inb M/C 2: (m/s)	Subtotal (m/s)	Total M/C (m/s)
Apollo 8	6.2	0.4	6.6	1.5	0.0	1.5	8.1
Apollo 9	-----	-----	0.0	-----	-----	0.0	0.0
Apollo 10	15.0		15.0	0.7		0.7	15.7
Apollo 11	6.4		6.4	1.5		1.5	7.8
Apollo 12	18.8		18.8	0.6	0.7	1.3	20.2
Apollo 13	7.1	11.5	18.6	2.4	1.0	3.4	21.9
Apollo 14	21.7	1.1	22.7	0.2	0.0	0.2	22.9
Apollo 15	1.6	1.6	3.3	1.7	0.0	1.7	5.0
Apollo 16	3.8		3.8	1.0	0.7	1.8	5.6
Apollo 17	3.2		3.2	0.6		0.6	3.8

AVERAGE =	10.9
SQRT(DEVSQ) =	22.0
3*average deviation = 3*AVEDEV(Ap8,Ap10:Ap17) =	20.9
Standard DEViation:	7.8

(Inbound AVERAGE = 2.9)

Lunar Orbit Insertion

◆ MAVERIC Simulation:

- Pre-LOI (C_3 = ~1.0 km^2/s^2)

- LOI (MAVERIC was 894 m/s)

- ~1 rev in LLO

- DOI (~6 minutes)

- Powered Descent (PD),
  & Touchdown (TD)

MAVERIC: Pre-LOI to LLO orbit & DOI

— LOI to LLO to DOI to TD
— DOI Burn
— LOI Burn
— Moon's Surface
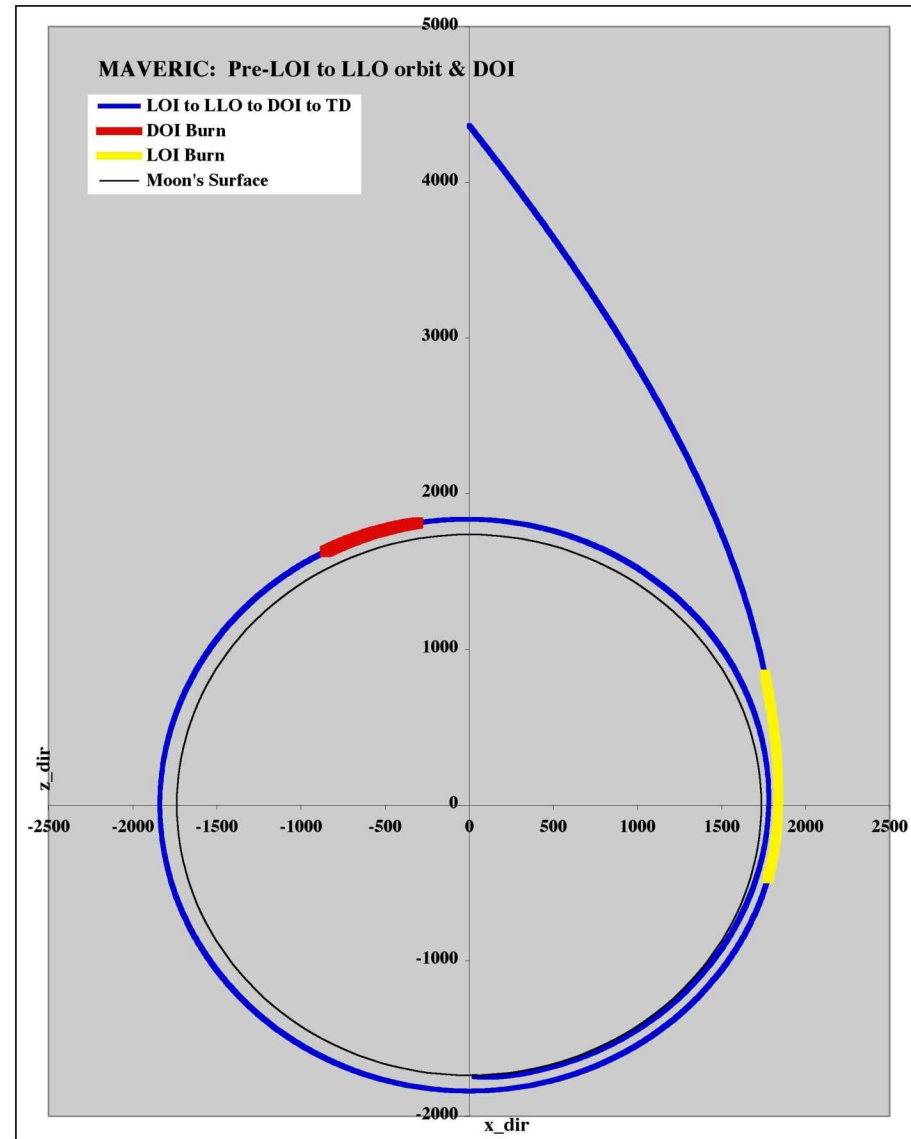
Pre-LOI thru Touchdown Trajectory (MAVERIC 3-DOF)

Example 24-hour Parking Orbits (~to scale)

Possible plane
change location
(4 examples/locations)
(inclination/node)

Other possible
orbit orientation

Possible plane
change location
(node/inclination)

$h_{apolune} =$
~15,925 km
(a = 9751 km)

b = 5698 km

Possible plane
change location
(node)

c = e*a=
7913 km

$h_{LEO} =$
100 km
(r = 1838)

Altitude at f=90°/270° =
~1592 km
(r = 3330 km)

Possible plane
change location
(inclination)

Lunar Equator

$R_{Moon} =$
1738 km
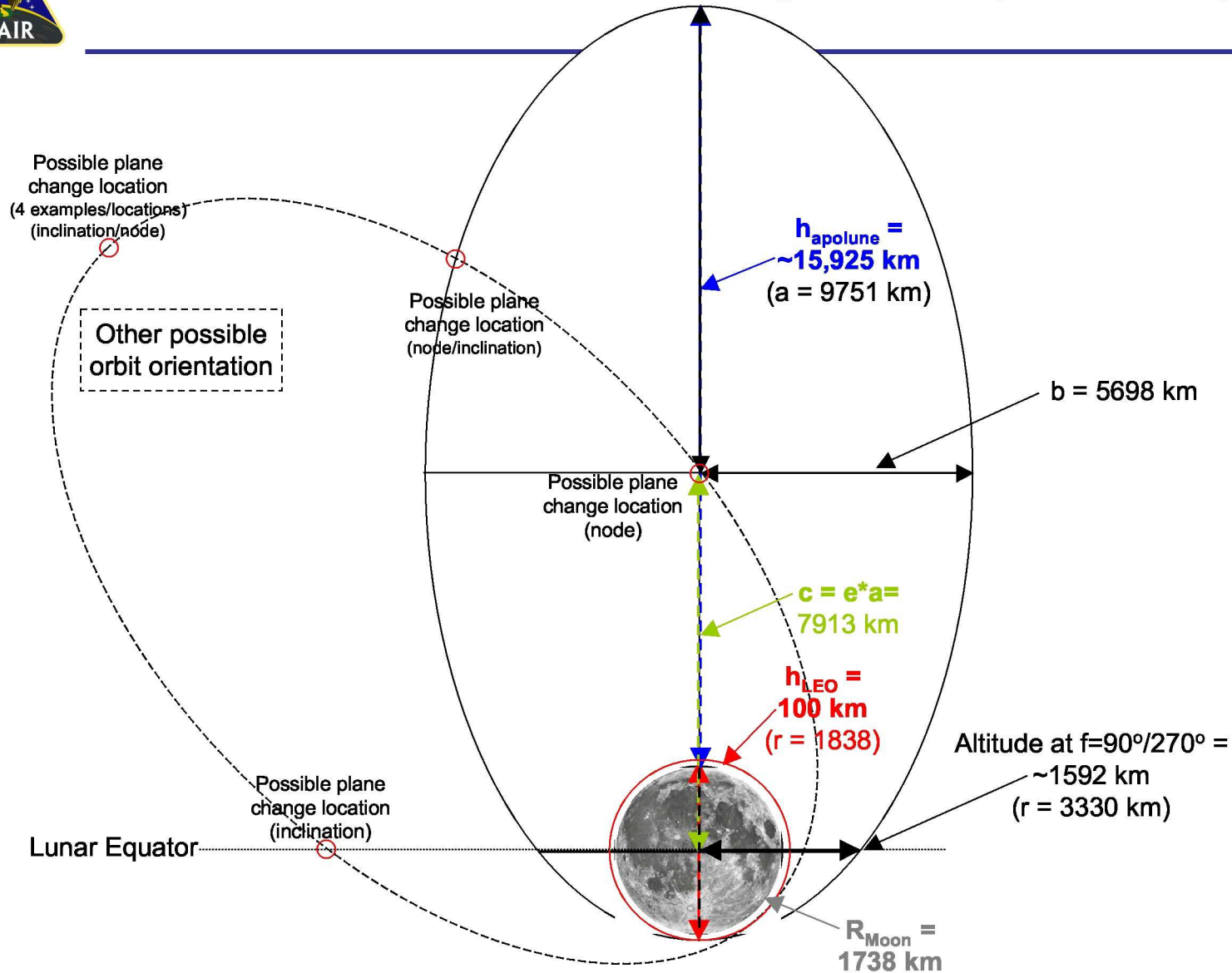
◆ **Altitude Maintenance for 4 additional days of LLO loiter (DM RCS)**

- 90 km minimum results in 10 m/s, 176 kg of RCS propellant for 5 days LLO loiter

◆ **Boil-off Propellant (DM MPS)**

- New analysis indicates 100-138 kg of LH_2 boil-off in needed for mission

◆ **Attitude Maintenance (DM RCS)**

- Analysis indicates 30 kg of ACS prop is needed per day in LLO for loiter
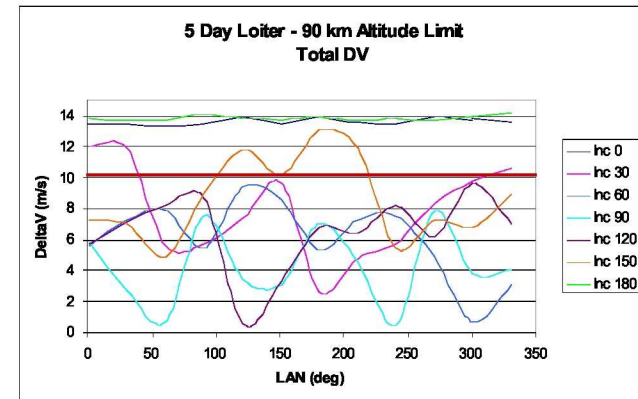
Time in LLO, days	Prop Budget for $\theta = 40°$, kg
1	30
2	60
3	90
4	120
5	150
5 (LLO) +1 (24-hr)	~156

5 Day Loiter - 90 km Altitude Limit Total DV

Several Days
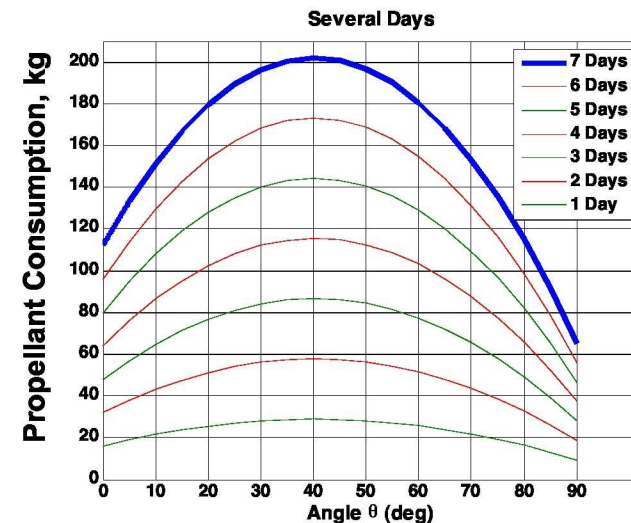
Ascent: Why Is the DMZ So Long?

Piloted Ascent:

- Derived from previous apside plots

- RCS assist to complete MECO can help only the last few seconds (chart § following)

- Note the very smooth energy addition all the way to MECO

Lower apside is not >0 until ~400 seconds into ascent (~99% of it)

Y-axis: Perilune / Apolune Alt., km

Inner plot Y-axis: Apside ("Apolune") alt., km (100 to -1800)

Inner plot X-axis: AM Ascent Time (sec) (0, 100, 200, 300, 400)
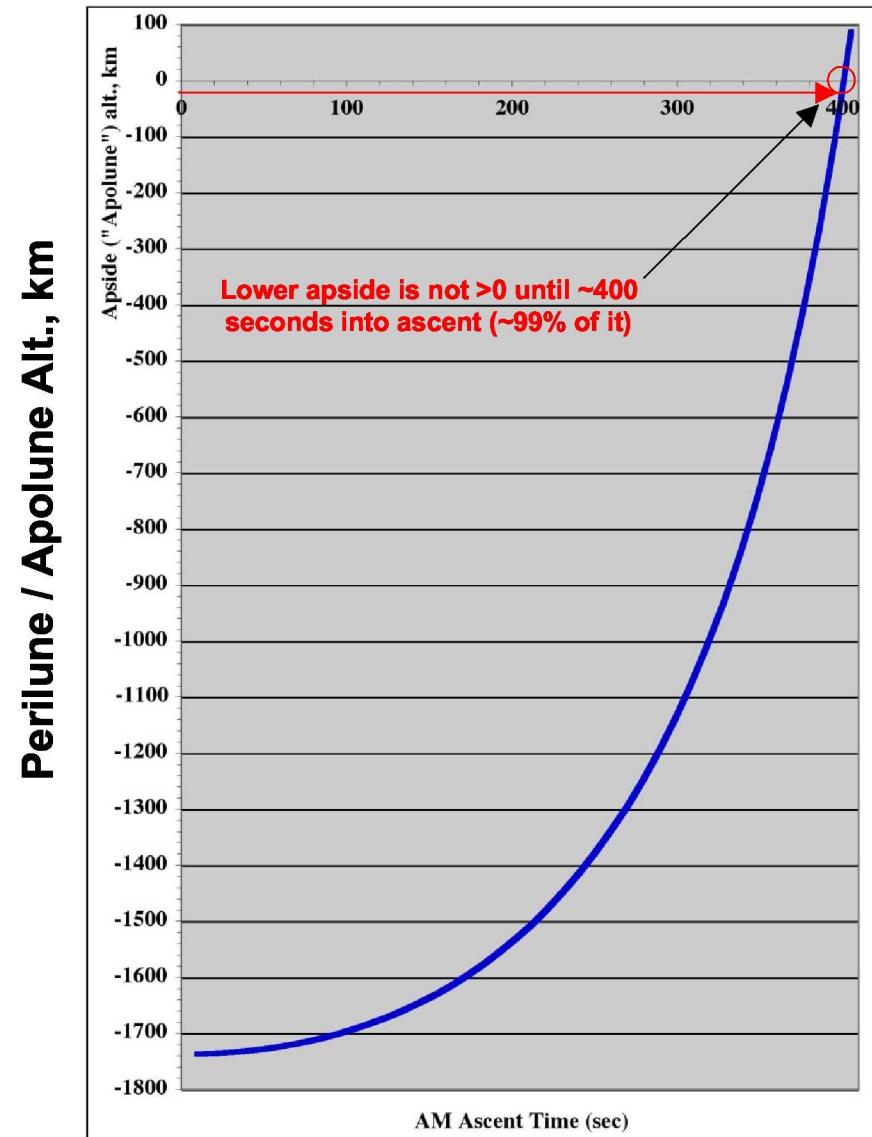
Ascent Time (to MPS burn-out/1st perilune), **sec**

Post-ascent Rendezvous, Prox. Ops & Docking

100 km x 100 km

TPI

H_{a-} = 86 km
H_{a+} = 100 km

3

Coast ~46 min

NCC 2

1

h_p = 15 km

0

Powered Ascent, ~7 min

15.24 km x 75 km

Coast ~43 min

Prox Ops & Docking

Braking / TPF

5 4
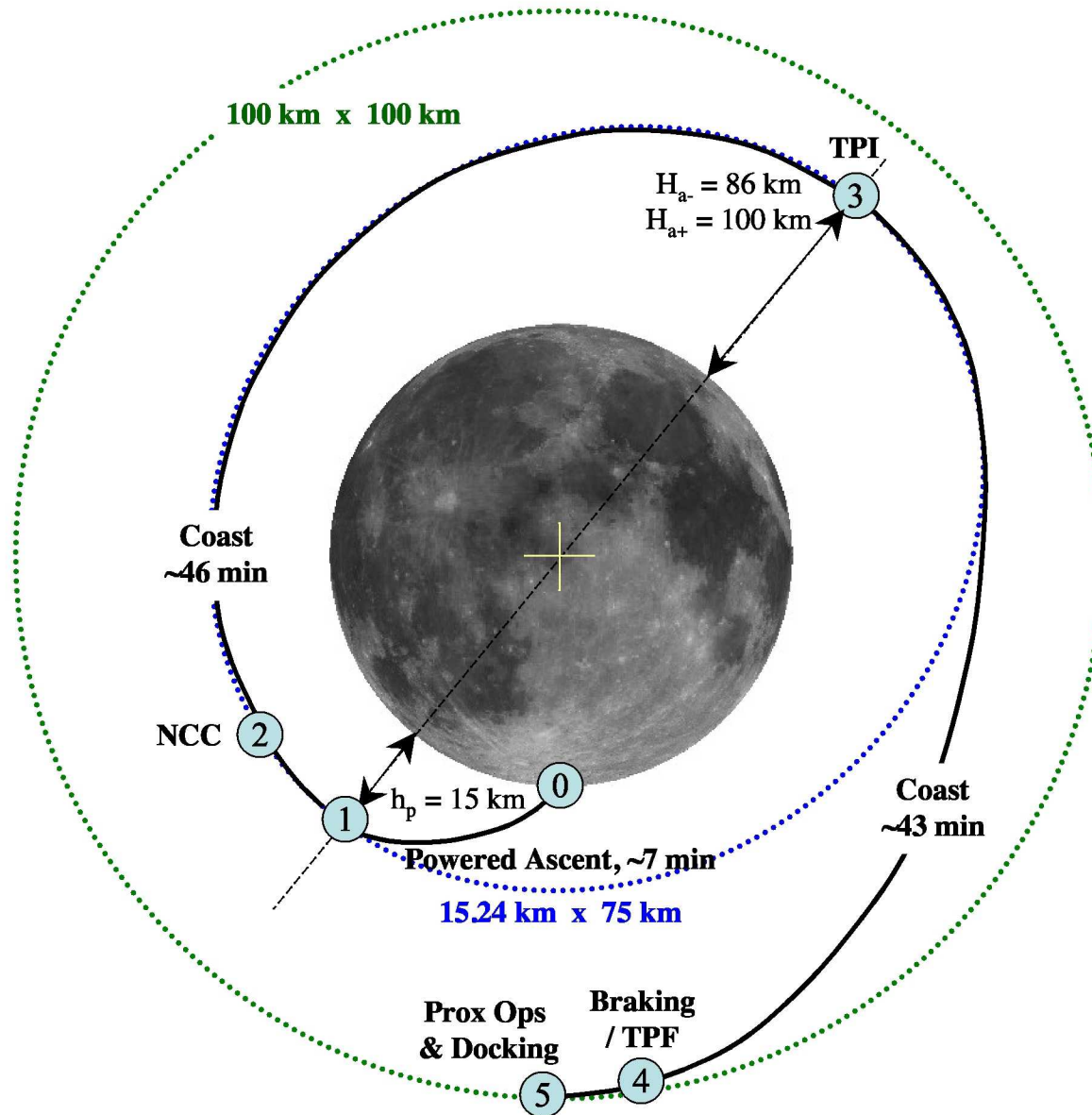
0 - Liftoff from surface

1 - Completion of powered ascent

2 - Correction burn

3 - TPI burn

4 - R-bar acquisition (2 km below)

5 - Proximity Ops & Docking

RPOD ΔVs + Dispersions Results

| Maneuver | Nom. ΔV | 1σ ΔV Dispersion | | | | | | ΔV+3*(all errors) |
		FLAK	Orion State Error	Thrust / I$_{sp}$	Injection	Misalignment	All Errors	
NCC:	1.744	0.000	*1.690*	*0.000*	*0.942*	0.000	*1.932*	7.540
TPI:	18.903	*0.080*	**2.655**	0.081	*1.005*	0.019	**2.875**	27.528
MC-1:	0.000	0.040	0.079	*0.096*	0.010	*0.165*	*0.209*	0.627
MC-2:	0.003	0.065	*1.623*	0.172	0.305	0.101	*1.805*	5.418
MC-3:	0.001	0.098	0.164	0.235	0.018	0.028	0.265	0.796
R-bar 1	0.411	0.075	0.155	0.151	0.015	0.024	0.176	0.939
R-bar 2	0.687	0.017	0.003	0.017	0.001	0.002	0.026	0.765
R-bar 3	0.468	0.011	0.002	0.015	0.000	0.003	0.019	0.525
R-bar 4	0.382	0.006	0.001	0.014	0.000	0.002	0.017	0.433
R-bar 5	0.312	0.004	0.001	0.014	0.000	0.002	0.016	0.360
R-bar 6	0.256	0.004	0.001	0.014	0.000	0.001	0.016	0.304
R-bar 7	0.212	0.004	0.001	0.014	0.000	0.001	0.015	0.257
R-bar 8	0.177	0.003	0.001	0.014	0.000	0.001	0.015	0.222
R-bar 9	0.150	0.003	0.001	0.014	0.000	0.001	0.015	0.195
R-bar 10	0.129	0.003	0.001	0.014	0.000	0.001	0.015	0.174
R-bar 11	0.115	0.002	0.001	0.014	0.000	0.001	0.014	0.157
R-bar 12	0.233	0.002	0.001	0.013	0.000	0.001	0.014	0.275
R-bar 13	0.124	0.003	0.006	0.027	0.000	0.006	0.028	0.208

Nominal ΔV: 24.315 (no dispersions)   **Total Required 3σ RPOD ΔV (Budget = 30 m/s):** 46.723

R123: Summary of RSS'd ΔVs To Be Used in Prop Sizing

**LDAC-4 Kick-off: Summary of Dispersion ΔVs -- CFP-1P**   **Blue/Bold = Inputs**

1st few dozen variables (of the "100s" possible)	ECRV Noise (1X) FLAK Mean m/s	ECRV 1σ m/s	White Noise (wh) FLAK Mean m/s	White 1σ m/s	Indiv. or Subtotal (M+3σ) Total (1X) m/s	Total (wh) m/s	Running Total: Σ(Means)+3*RSS(σ's) Total (1X) m/s	Total (wh) m/s
TCM-1 (Nav., exec., FLAK)			7.45	4.69		21.51		21.5
TCM-2 (Nav., exec., FLAK)			0.9	0.4		2.02		22.4
TCM-3 (Nav., exec., FLAK)			1.6	0.7		3.67		24.2
TCM-4 (Nav., exec., FLAK)			1.1	0.5		2.63		25.4
... Prop system change ...								
LOI (Nav., exec., FLAK)	0.01	0.9	0.02	0.9	2.6	2.7	2.6	2.7
... Prop sys. & large mass change ...								
Post-LOI TCM/clean-up			3.0	1.3		7.0		7.0
... Prop system change ...								
Plane Change Isp							TBD	
Plane Change Mis-align							TBD	
... Prop system change ...								
DOI (Nav., exec., FLAK)	0.5	1.5	0.1	0.3	4.9	1.0	4.9	1.0
... Prop system change ...								
*PDI I.C.'s							0.0	
*PD Thrust	----	3.0					9.0	
*PD Isp	----	2.8					12.3	
*PD prop mass	----	2.5					14.4	
*PD Xcg	----	2.9					16.8	
*PD Ycg	----	2.9					18.8	
*PD Zcg	----	1.4					19.3	
*PD Ixx	----	2.7					20.9	
*PD Iyy	----	2.8					22.6	
*PD Izz	----	2.8					24.1	
*PD Ixy	----	2.7					25.4	
*PD Ixz	----	2.8					26.8	
*PD Iyz	----	3.0			28.2		28.2	53.0
... Prop system change ...								
DM RCS Isp/PD							TBD	
DM RCS Mis-align/PD							TBD	
.. Prop sys. & large mass change ..								
**Ascent Thrust	----	0.9					2.8	
**Ascent Isp	----	0.03					2.8	
**Ascent m0	----	1.4					5.1	
**Ascent Mis-align	----				5.1		5.1	3.7
... Engine change ...								
RPOD, nominal	24.315	----						
RPOD FLAK	----	0.5						
RPOD CEV State Error	----	7						
RPOD Thrust/Isp	----	1.0						
RPOD Injection	----	2.3						
RPOD Misalignment	----	0.4			22.41		46.723	

Annotations: RSS (groupings on left); LDAC-4 Kickoff Buy-back (25.4, 7.0); Not bought back (4.9); Previous (LDAC-1) (53.0); LRAC-2 buy-back (2.6, 5.1, 46.723).

* (Guidance & Control only)   ** (Guidance only)

To be used in sizing   LDAC-4   3/18/2010

33

Bio data: Larry Kos
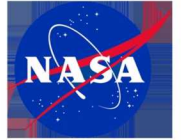
♦ **<u>Education</u>:**
- Master of Aerospace Engineering, Auburn University, 1996

♦ **<u>Work Experience</u>:**
- Marshall Space Flight Center (MSFC),1982-present
  - MSFC Technical Lead, Human Mars Mission, 1996-1999
  - Agency Technical Lead, Integrated In-Space Propulsion Technology Plan, '01-'04
  - Agency Technical Lead, Low-Thrust Trajectory Tool Development, 2003-2006
  - Altair Integrated Performance Group Lead, 2007-2009
  - Human Exploration Mission Designer/Analyst, 1993-present

♦ **<u>Awards</u>:** More than 20 NASA awards including --
- Agency-level: NASA Commendations
- MSFC-level: Director's Commendations, Space Flight Awareness Awards, Special Service Awards, Certificates of Appreciation, On-the-Spot Awards, Time-off Awards, Group Achievement Awards

♦ **<u>Publications</u>:**
- Mr. Kos has published ~20 papers, including papers and NASA TMs on the Exploration activities above
- Featured/invited presentations at LOGSA ('98) & JANNAF ('03) conferences