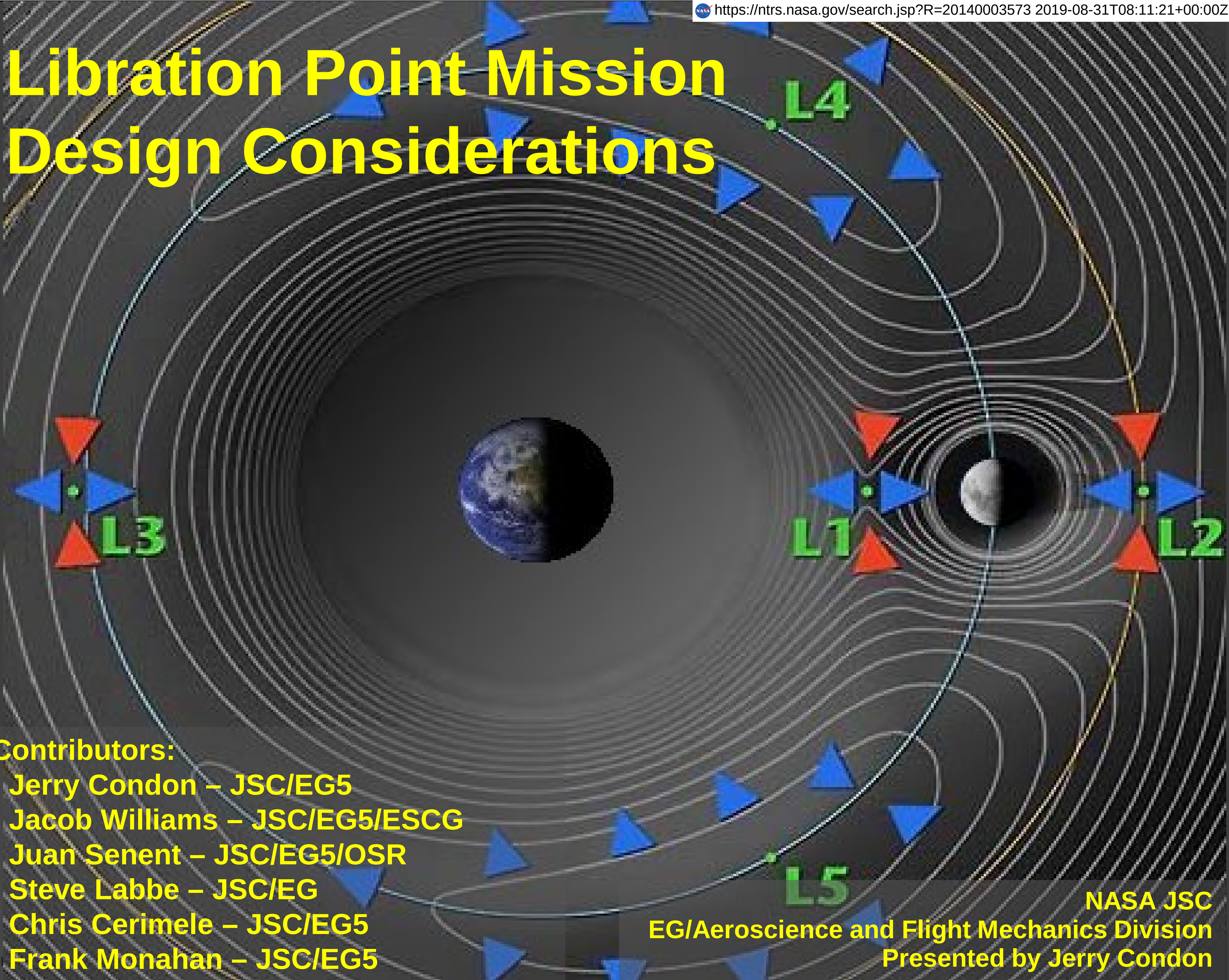
Libration Point Mission Design Considerations



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Presented by Jerry Condon

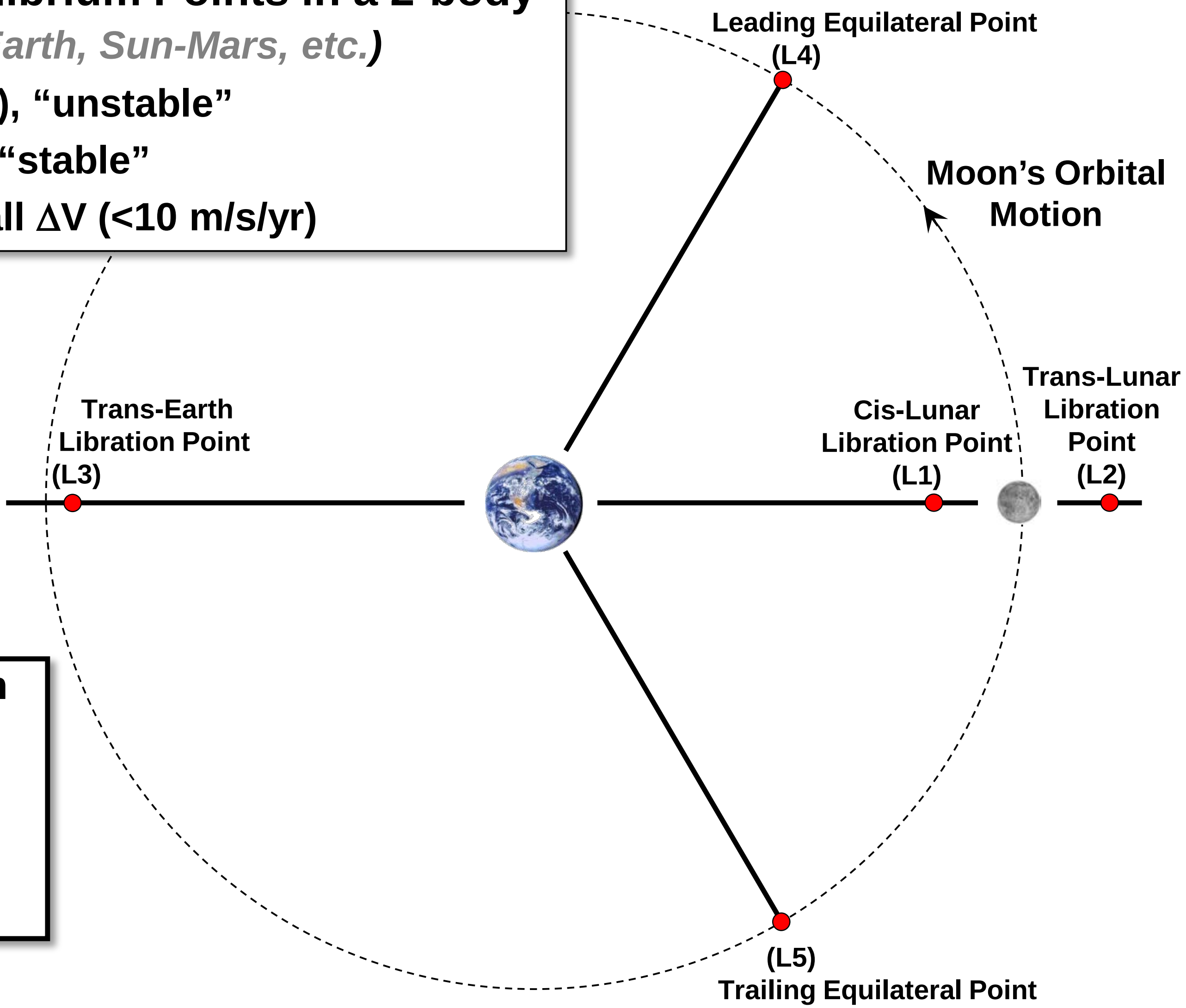
Outline

- Earth-Moon Libration Point System
- Halo Orbit Families & Geometries
- Libration Point Transfer Options
- Transfer Option Performance Example (ΔV, time)
  - Direct
  - Lunar Flyby
  - Low Energy (Manifold)
- L2 Halo Orbit Special Considerations
  - Orbit Maintenance
  - Communication
  - Rendezvous
  - Launch Opportunities
  - Aborts

Earth-Moon Libration Points

Libration Points are Equilibrium Points in a 2-body system (*Earth-Moon, Sun-Earth, Sun-Mars, etc.*)

- **Collinear points (L1, L2, L3), “unstable”**
- **Equilateral points (L4, L5), “stable”**
- **Station-keeping – very small ΔV (<10 m/s/yr)**



Earth-to-Moon – 384,400 km
L1 – 57,731 km from Moon
L2 – 64,166 km from Moon
L3 – 381,327 km from Earth
L4 & L5 – 384,400 km from Earth and Moon

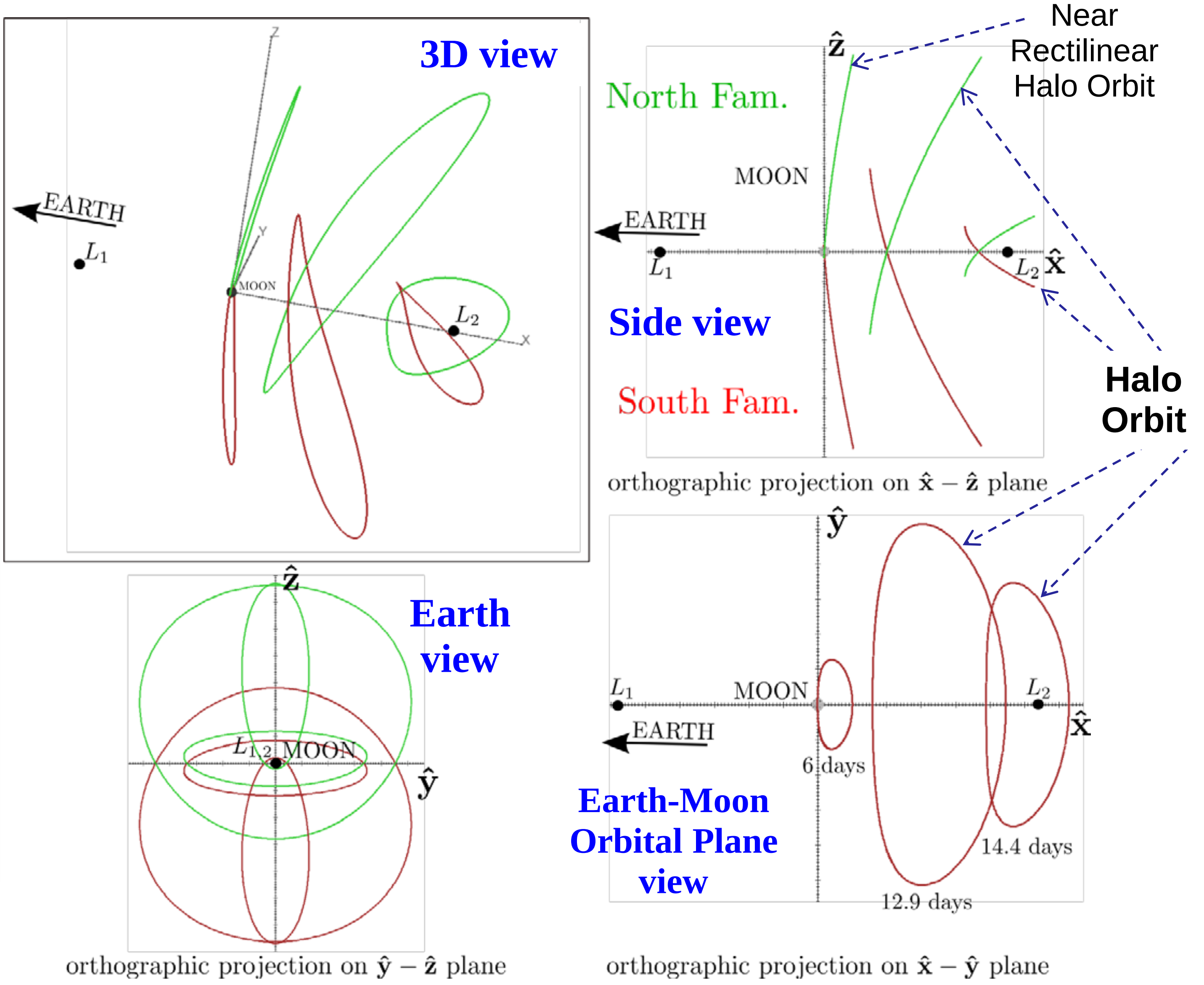
Halo Orbit Families and Geometry

Halo Orbit:

- A periodic, 3-D orbit near the L_1, L_2, L_3 Lagrange Pt.
- Spacecraft travels in a closed, repeating path near the Lagrange point
- Not technically orbiting the Lagrange point
- Tend to be unstable, station keeping is required

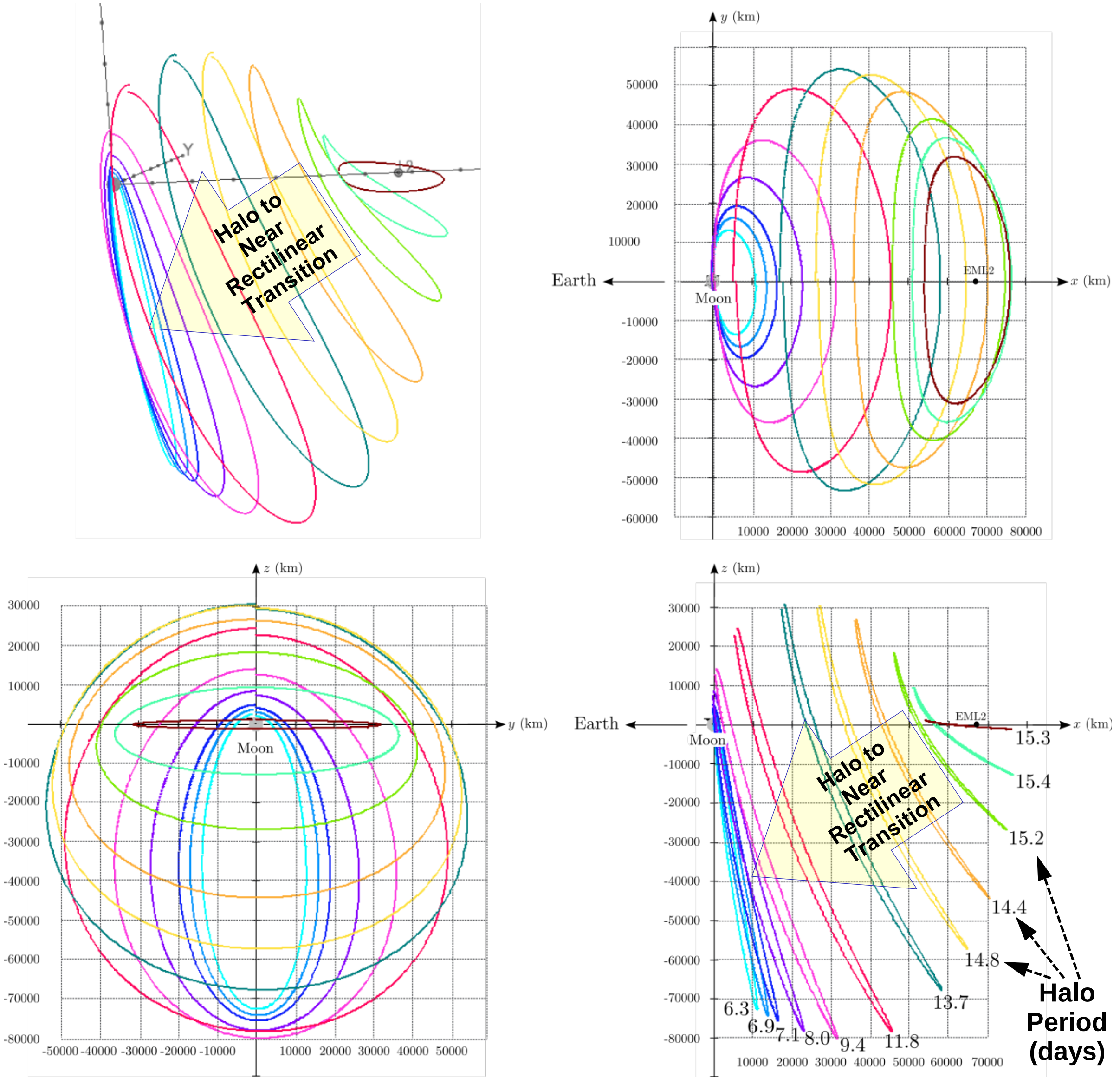
Rule of thumb:

- *Period of halo is ~1/2 of the period of the primaries*
- *In the Earth-Moon system, a halo period is ~14 days*



Earth/Moon-L2 Halo South Family

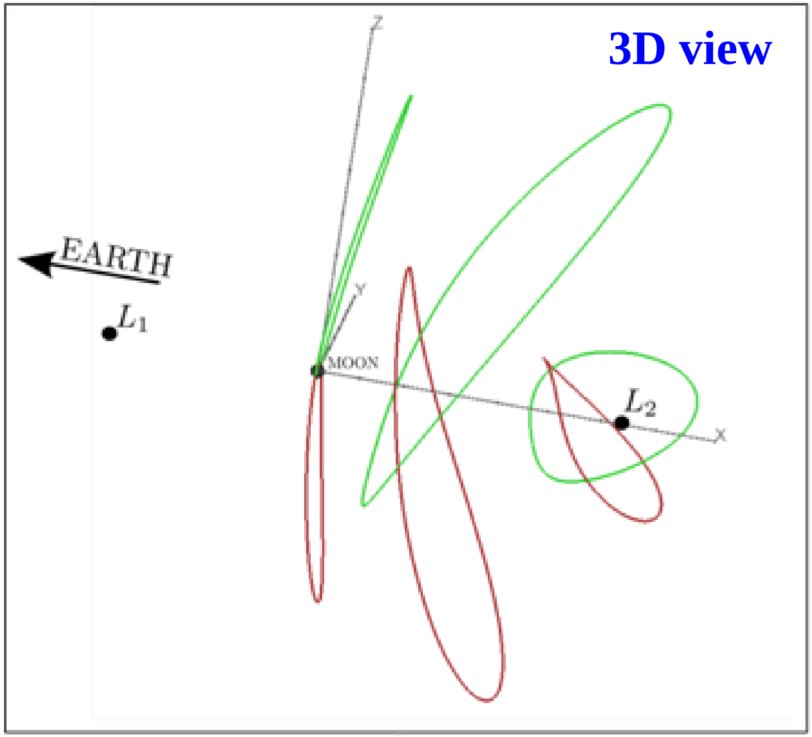
- ❑ Halo amplitudes range from approximately 0 to 30,000 km (north) and up to 80,000 km (south)
- ❑ As the Z-amplitude (Az) of the halo is increased, there is a transition to a near rectilinear halo with a 6-7 day period.
- ❑ Near rectilinear halos are less unstable and require less orbit maintenance*
- ❑ Lower Az halos are more unstable*, but more amenable to transition to and from weak stability boundary manifold trajectories



* Multibody Orbit Architectures for Lunar South Pole Coverage. D. Grebow, M. Ozimek, and K. Howell

L2 Halo Selection Considerations

- L2 or L2 Halo Orbit selection dependent on various parameters besides transportation
  - o Environmental considerations in L2 halo orbits
    - – Thermal, Radiation, MMOD
  - o Orbit maintenance costs
  - o Best halo for Earth communication and visibility
  - o Best halo for Lunar South Pole visibility
  - o Science considerations in L2 Halo Orbits
  - o Excursions to LLO or alternate Halo Orbits

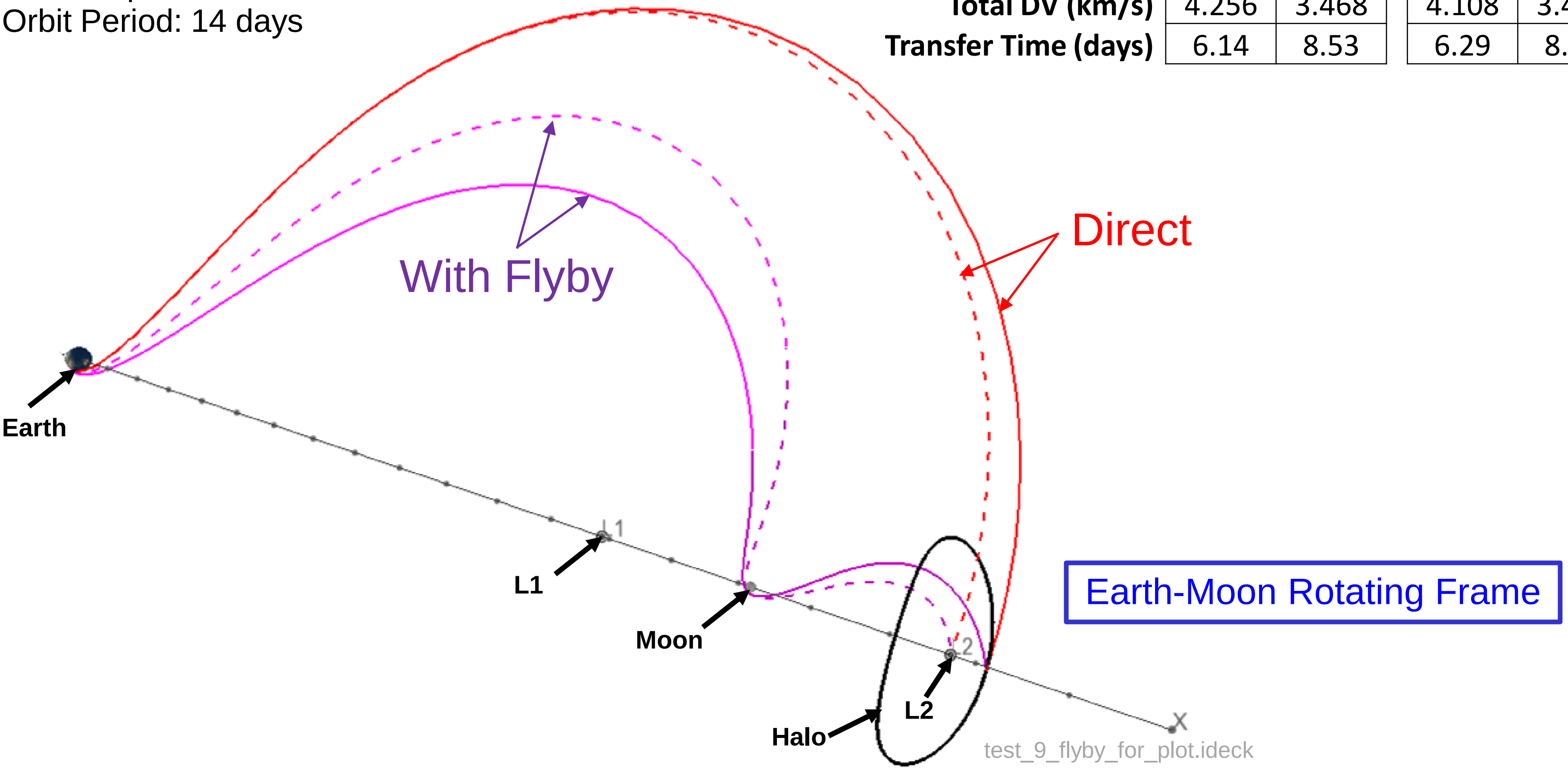


LEO to Earth-Moon L2
Direct and Flyby – Example Cases

LEO inclination: 28.5 deg

L2 Halo orbit:
Max. Amplitude in x-axis: 11,904 km
Max. Amplitude in y-axis: 34,672 km
Max. Amplitude in z-axis: 10,000 km
Orbit Period: 14 days

Destination	L2		L2 Halo	
Mission Type	direct	flyby	direct	flyby
LEO DV (km/s)	3.149	3.134	3.151	3.133
Lunar Periapsis DV (km/s)	0.000	0.186	0.000	0.175
Capture DV (km/s)	1.107	0.148	0.957	0.109
Total DV (km/s)	4.256	3.468	4.108	3.416
Transfer Time (days)	6.14	8.53	6.29	8.35



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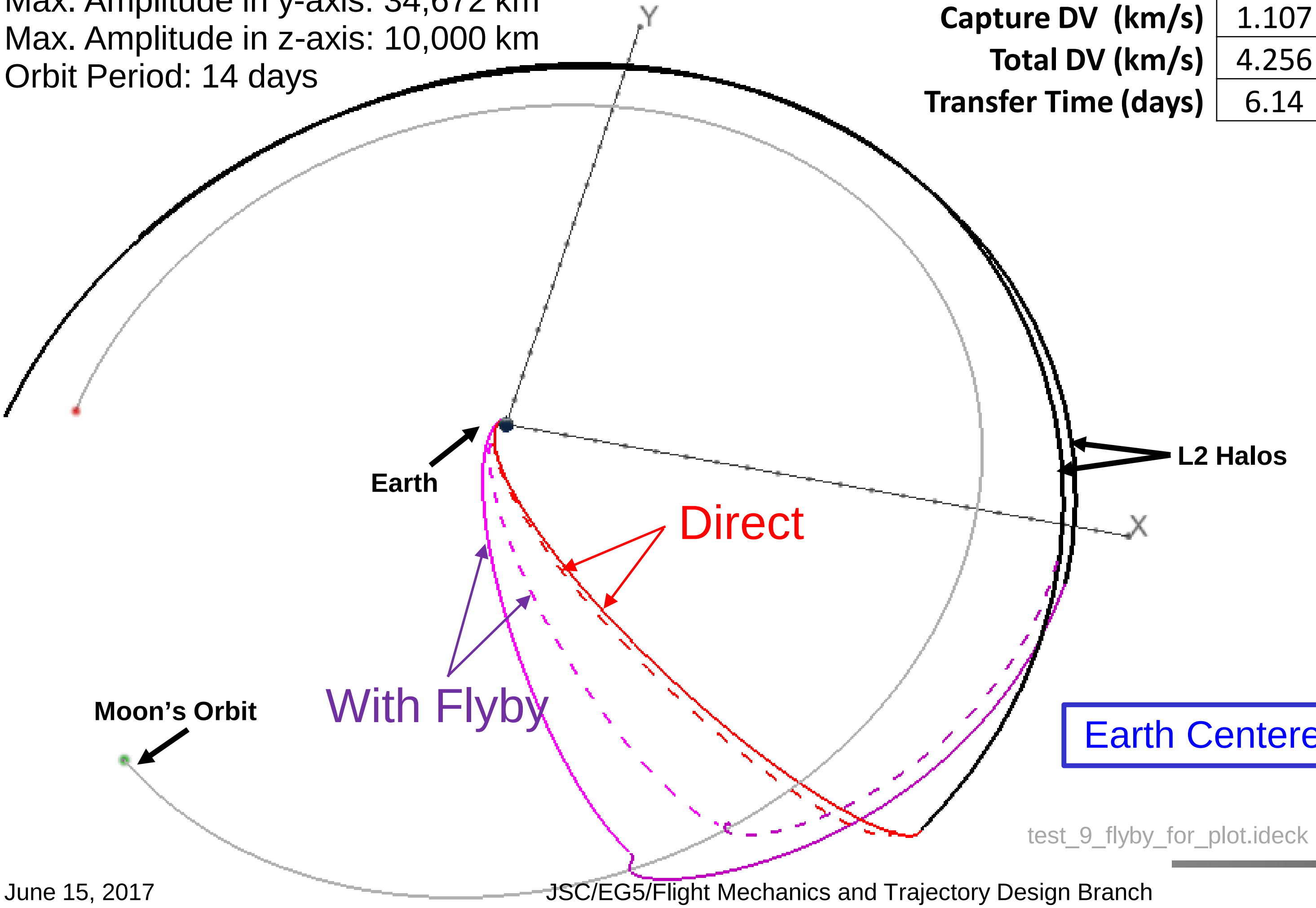


Earth Centered Ecliptic Frame

Earth Orbit to EM-L2 – Minimized Arrival ΔV

<u>Example Case-Low Energy Trajectory Design</u>

Performance

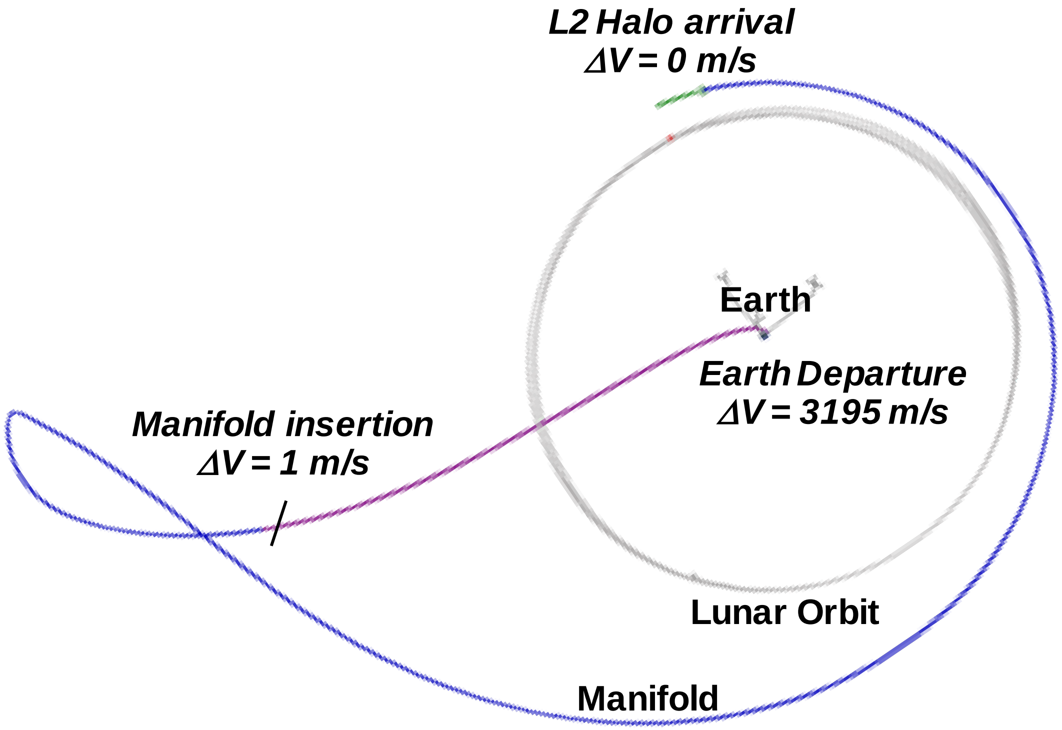
- Earth departure ΔV = 3195 m/s
- Manifold insertion ΔV = 1 m/s
- Total flight time = 103 days

Flight Profile

- After Earth launch, depart from a 28.5°, 185 km circular altitude LEO parking orbit, (ΔV = 3195 m/s)
- Achieve energy (C3) to reach the L2 Halo manifold insertion point
- Reach manifold insertion point (Earth departure + 10 days)
- Insert onto manifold (ΔV = 1 m/s)
- Coast on a trajectory taking the s/c 1-2 million km where the Sun's gravity field guides the trajectory to the L2 Halo arrival point
- Reach L2 Halo arrival (Earth departure + 103 days)

Comments

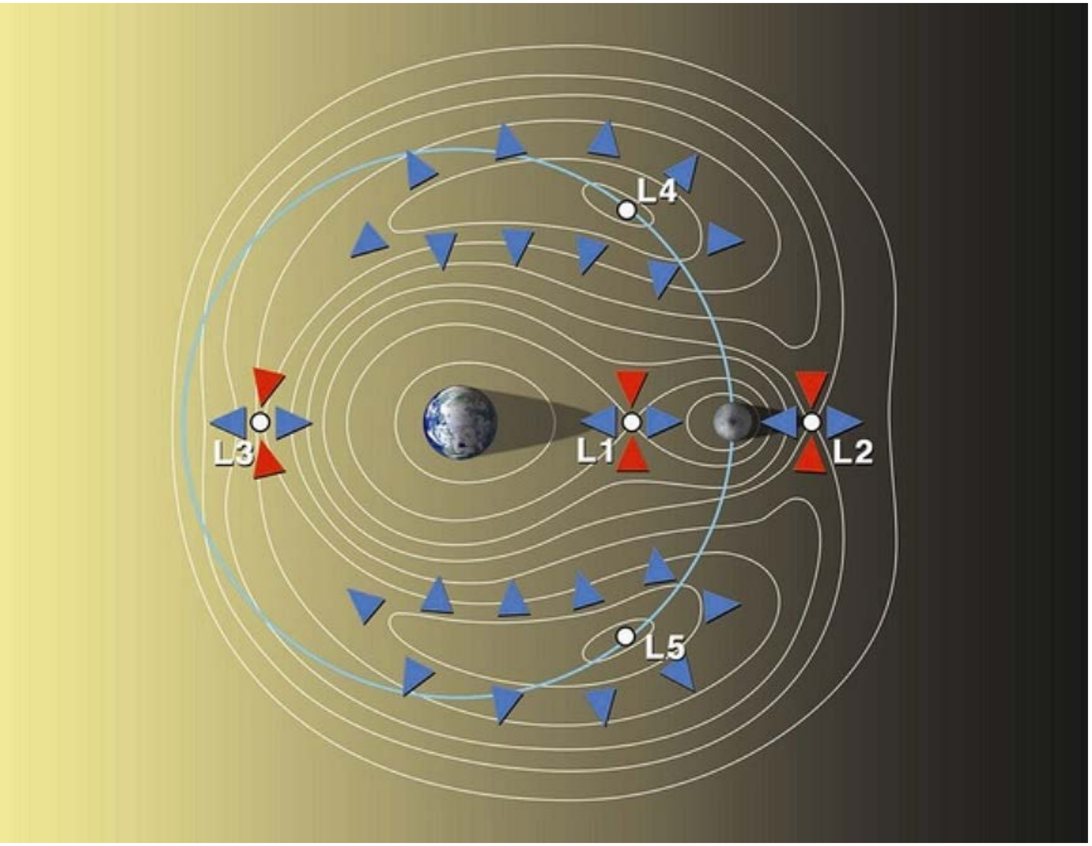
- Other low energy trajectory types (lunar flyby, lunar/earth flyby)



Low Energy Trajectory Design Example

Low Energy Transfer Option

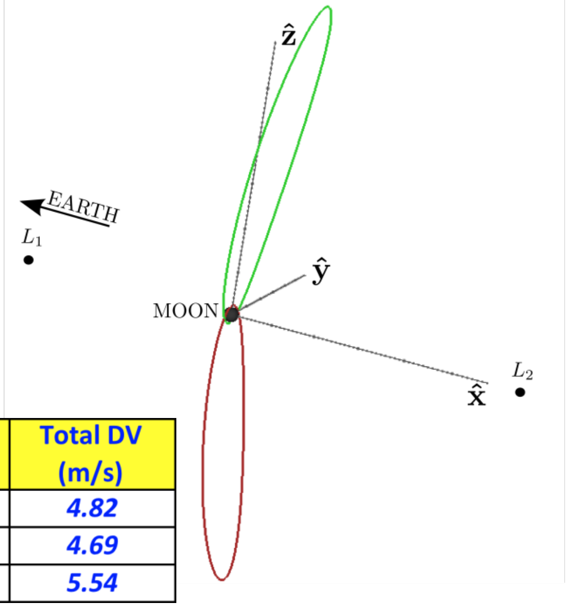
- Invariant manifolds (stable and unstable) lead into and out of L1 & L2
- Connect manifolds to construct low energy transfers to halo or other periodic orbits
- Can use this technique to generate trajectories from Earth to halo orbits in the Earth-Moon (as well as Sun-Earth) system with an extremely small arrival ΔV requirement
- Earth departure ΔV can be slightly higher than crewed direct or lunar flyby trajectory
- Opens selection to low ΔV capability buses
- Longer trip time – uncrewed flights only
- May be limited opportunities



Orbit Maintenance for Halo Orbit Families

- Station keeping at L2 for 1 year can be as low as <5-10 m/s/year – Near-rectilinear halo orbit

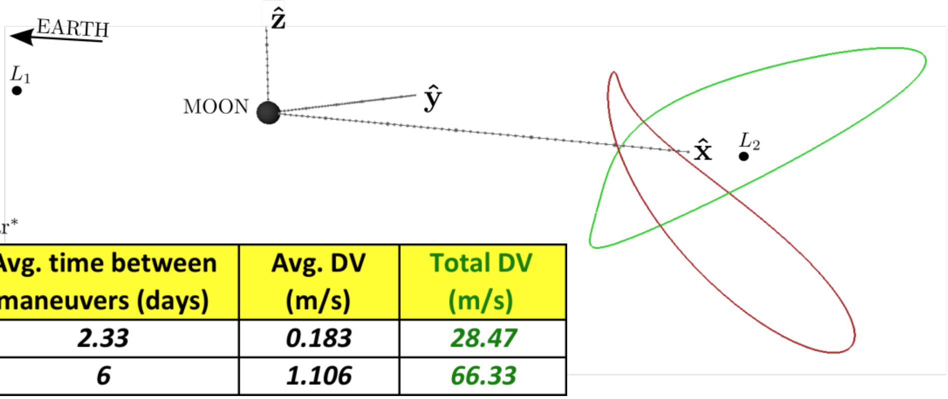
- Type of Halo Orbit impacts costs – can increase to <25-30 m/s/year



Station Keeping for 1 year*

Orbit Type	Libration Point	Period (days)	No. of Maneuvers	Avg. time between maneuvers (days)	Avg. DV (m/s)	Total DV (m/s)
Near-rectilinear halo	*L2*	*7*	*86*	*4.2*	*0.057*	*4.82*
Near-rectilinear halo	*L2*	*8*	*55*	*6.4*	*0.086*	*4.69*
Near-rectilinear halo	*L2*	*8*	*55*	*6.4*	*0.101*	*5.54*

Values are representative



Station Keeping for 1 year*

Orbit Type	Libration Point	Period (days)	No. of Maneuvers	Avg. time between maneuvers (days)	Avg. DV (m/s)	Total DV (m/s)
Halo	*L2*	*14*	*156*	*2.33*	*0.183*	*28.47*
Halo	*L1*	*12*	*60*	*6*	*1.106*	*66.33*

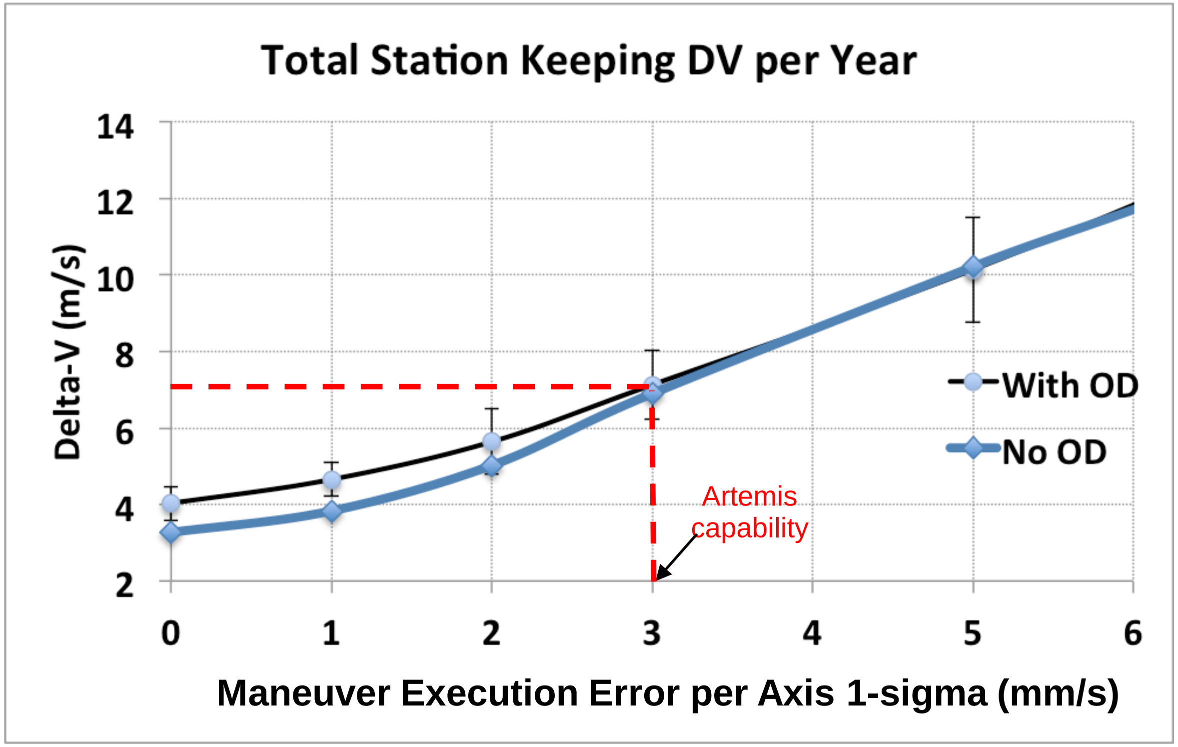
*Multibody Orbit Architectures for Lunar South Pole Coverage. D. Grebow, M. Ozimek and K. Howell

Orbit Maintenance ΔV Cost In EM-L2 Halo

Courtesy: Jeff Parker, JPL

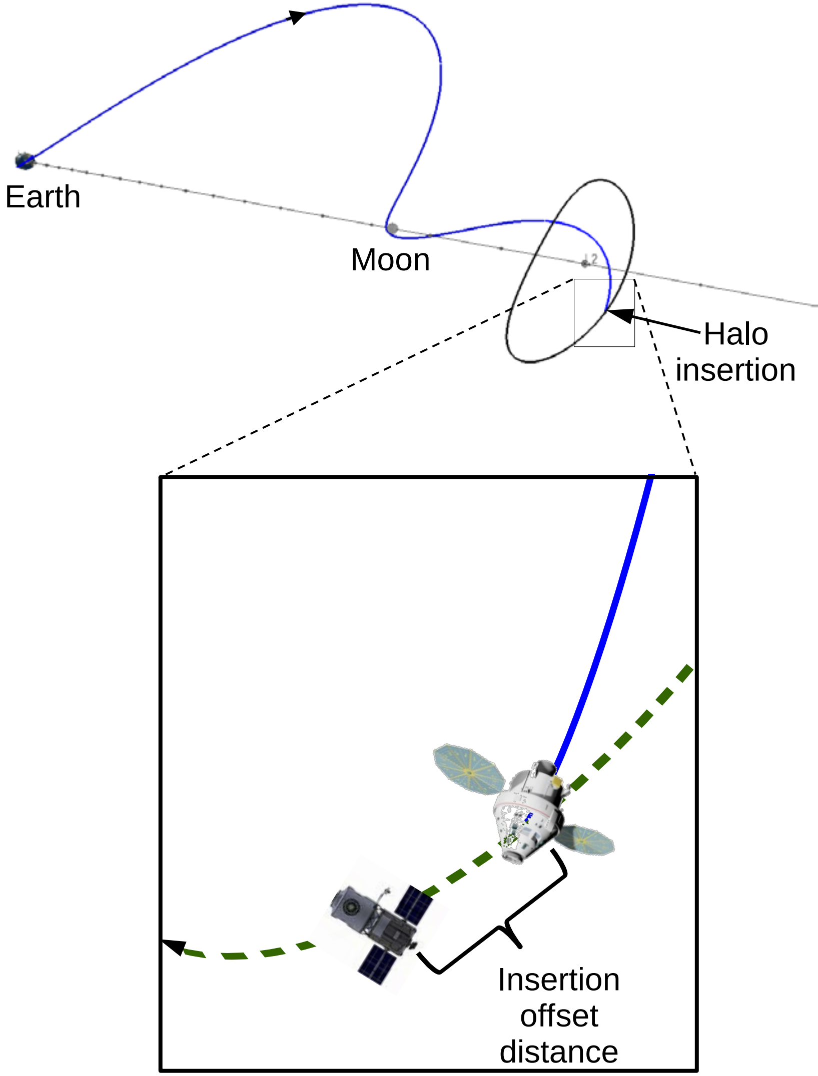
- ☐ Station keep ΔV depends upon:
  - Control law
  - Maneuver execution error
  - Navigation Orbit Determination (OD)
- ☐ With a good control law, navigation accuracy and execution precision will dominate the station keeping ΔV cost

- ☐ Assumptions
  - 24 maneuvers over 12 revs (~2 maneuvers / rev)
- ☐ Reference EM-L2 halo z-amplitude = 10 km



Earth-Moon L2 Halo Orbit Rendezvous

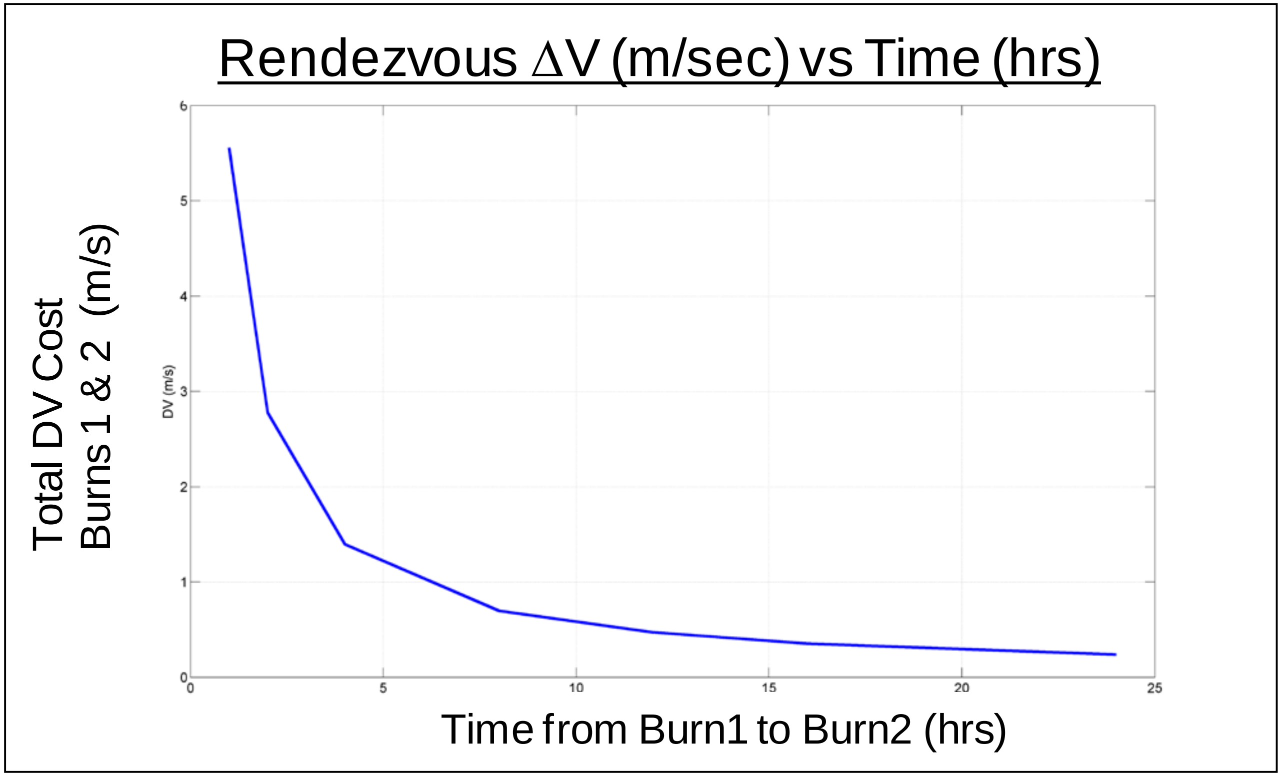
- ☐ The EM-L2 Halo rendezvous mission begins at Earth launch.
- ☐ The MPCV can be launched daily to target the EM-L2 halo itself
- ☐ There exists an optimal LEO to EM-L2 halo insertion location that occurs once during the halo period (around 14 days).
- ☐ Launching at a time designed to insert the MPCV onto the halo at this time will provide a minimum ΔV requirement.
- ☐ Launching at a time away from this optimal time will incur additional MPCV ΔV cost.
- ☐ MPCV inserts onto EM-L2 halo at a selected offset distance.
- ☐ MPCV later performs maneuver(s) to close this distance, ultimately docking with the Waypoint Spacecraft (WSC)



Earth-Moon L2 Halo Orbit Rendezvous – Example Methodology –

Assumptions

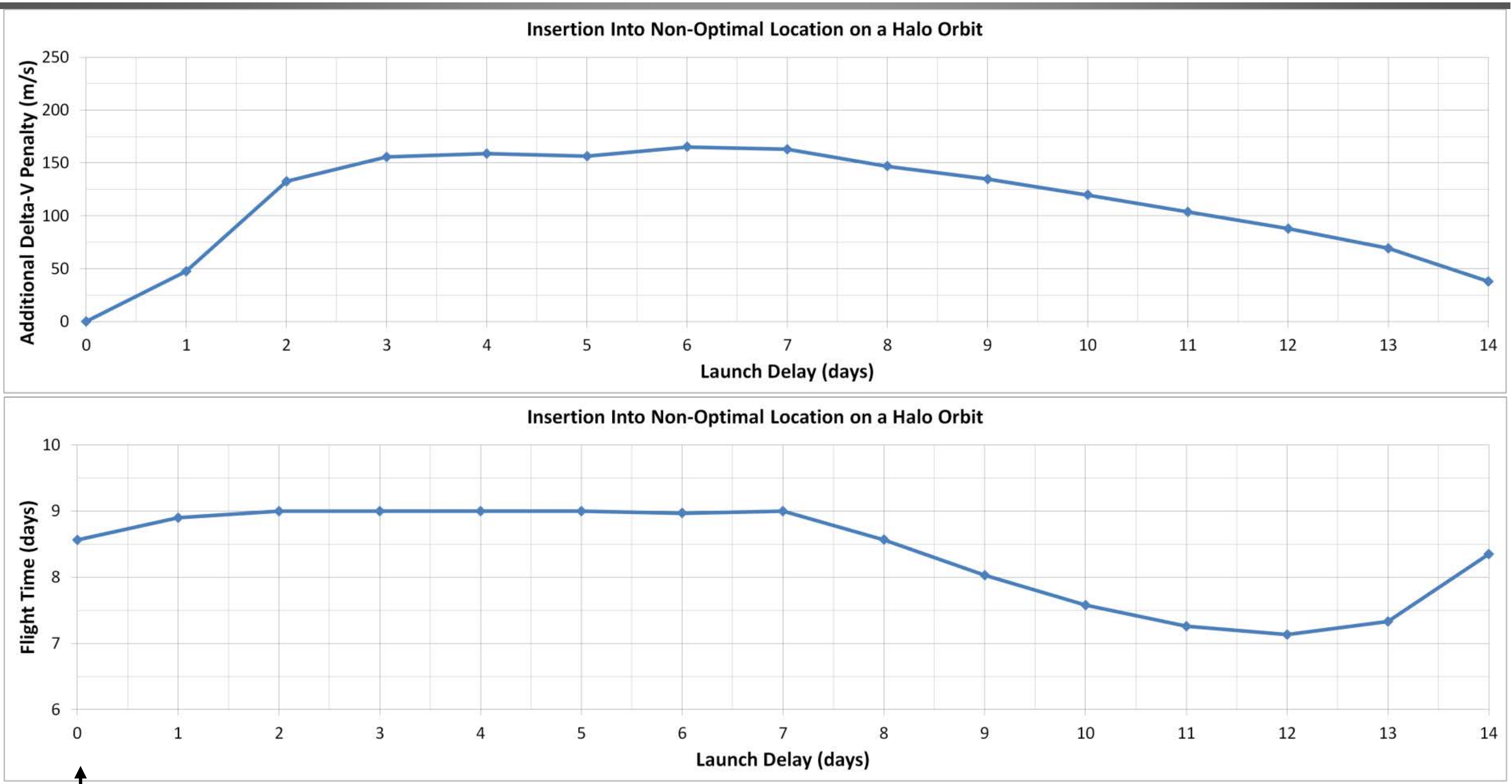
- Target s/c in halo orbit about L2
- Chaser s/c inserts onto (target) halo, trailing by 10 km
- Chaser executes a 2-manuever sequence to close the distance
- Chaser burn 1 maneuver closes distance between Chaser and target over a selected duration



Effect of Launch Delays on MPCV ΔV Requirement
Introduction/Methodology

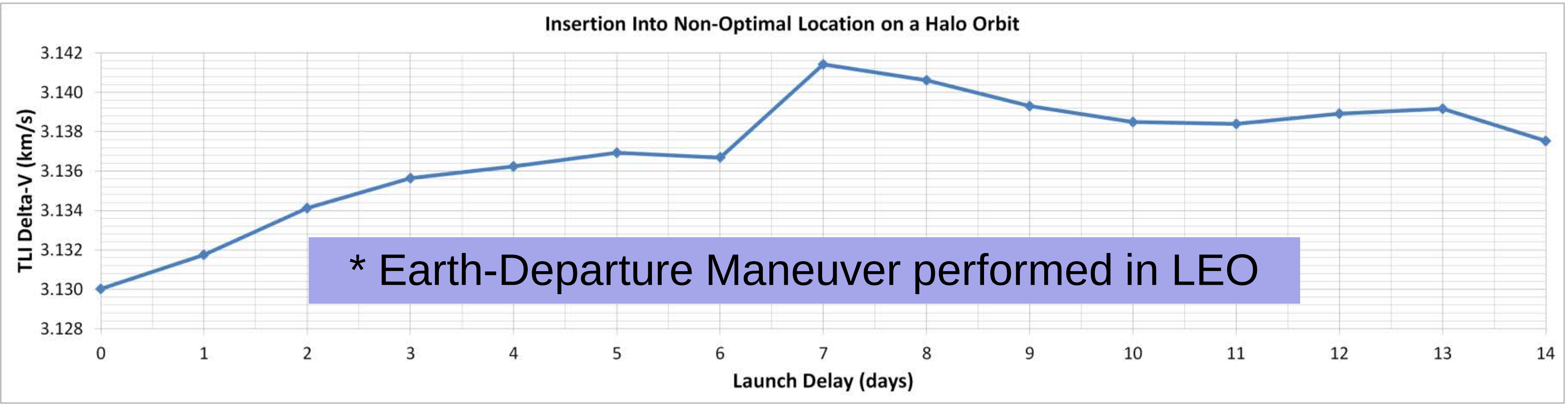
- ❑ Vary the departure epoch (Earth departure maneuver is delayed) from the nominal (minimum Δv) mission.
- ❑ For the nominal mission, the WSC is at the optimal insertion offset point at the optimal (minimum DV) MPCV insertion time.
  - The nominal mission assumes zero launch delay.
- ❑ Rendezvous occurs at a new location to minimize Δv and limit the flight time to 9 days.
- ❑ The x-axis in the plots are days past the nominal TLI.

Effect of Launch Delays on MPCV ΔV Requirement Results

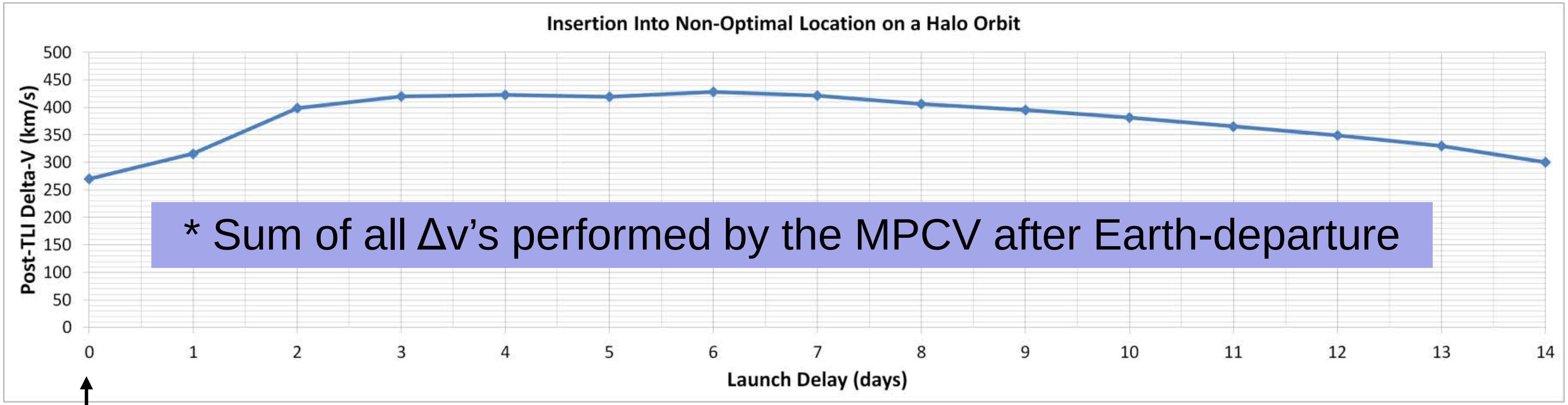


Nominal Departure

Effect of Launch Delays on MPCV ΔV Requirement Results



* Earth-Departure Maneuver performed in LEO

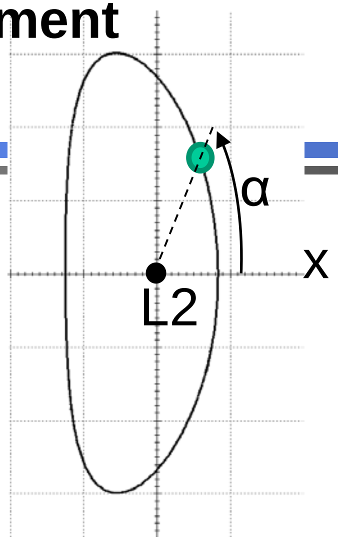


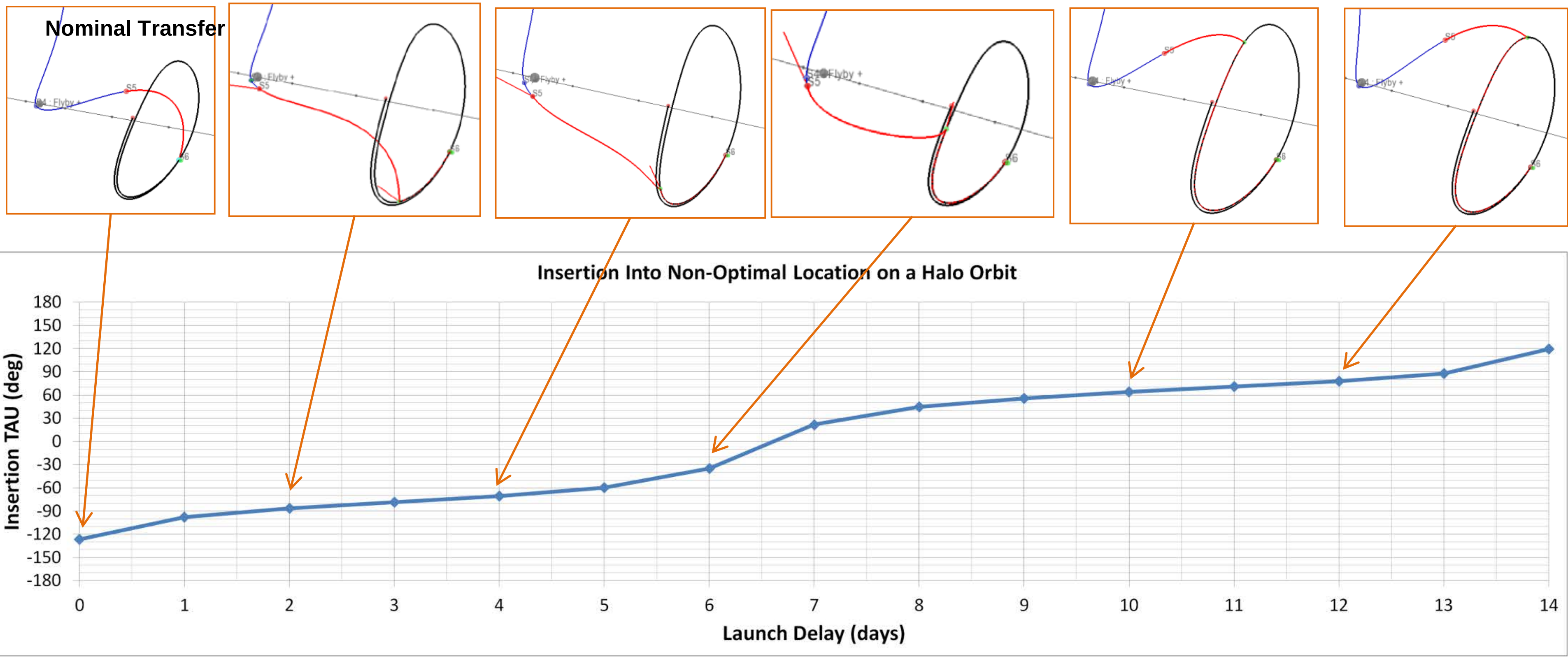
* Sum of all Δv's performed by the MPCV after Earth-departure

Nominal Departure

Effect of Launch Delays on MPCV ΔV Requirement Results

- In this plot, τ is really (180 – α), since we are using real (not osculating) halo orbits.
- α is the insertion right ascension (in the Earth-Moon rotating frame, centered at the L2 point).





Effect of Launch Delays on MPCV ΔV Requirement Summary

❑ It turns out that freeing the Moon flyby altitude and flyby maneuver true anomaly constraints result in significant savings for the worst off-nominal cases.

  - About 75 m/s for the +4 day case
  - The worst-case additional ΔV cost is 165 m/s (for a 6-day delay)

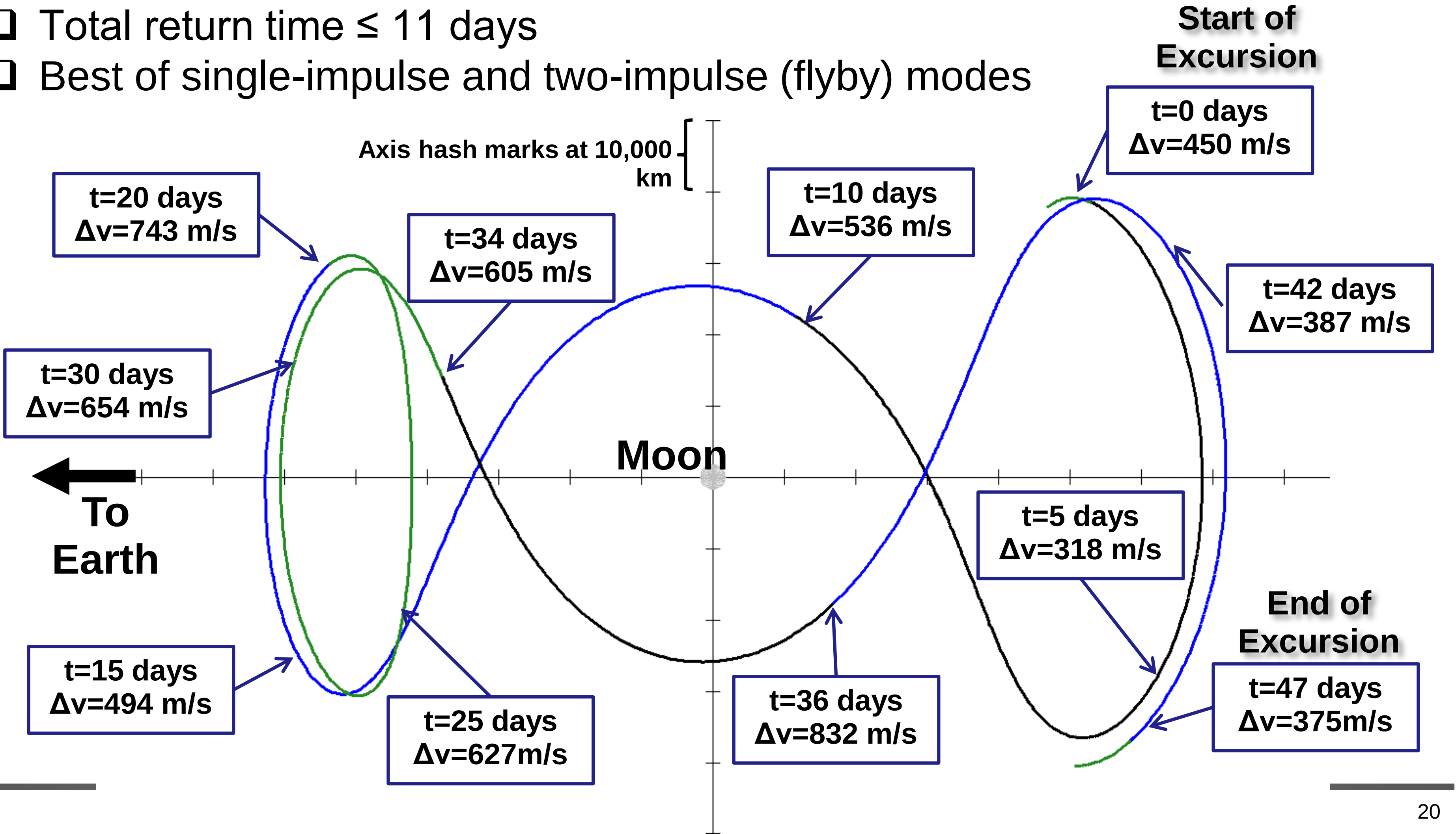
❑ This analysis does not account for delays in LEO after launch (it is assumed that TLI is always performed in the optimal LEO for the departure epoch).

❑ There may be cases where a multiple-maneuver insertion sequence may reduce the cost [haven’t looked at this yet].

❑ This analysis also assumes that the WSC is a totally passive vehicle. A further study could assume that it is capable of also performing maneuvers to produce a more favorable alignment.

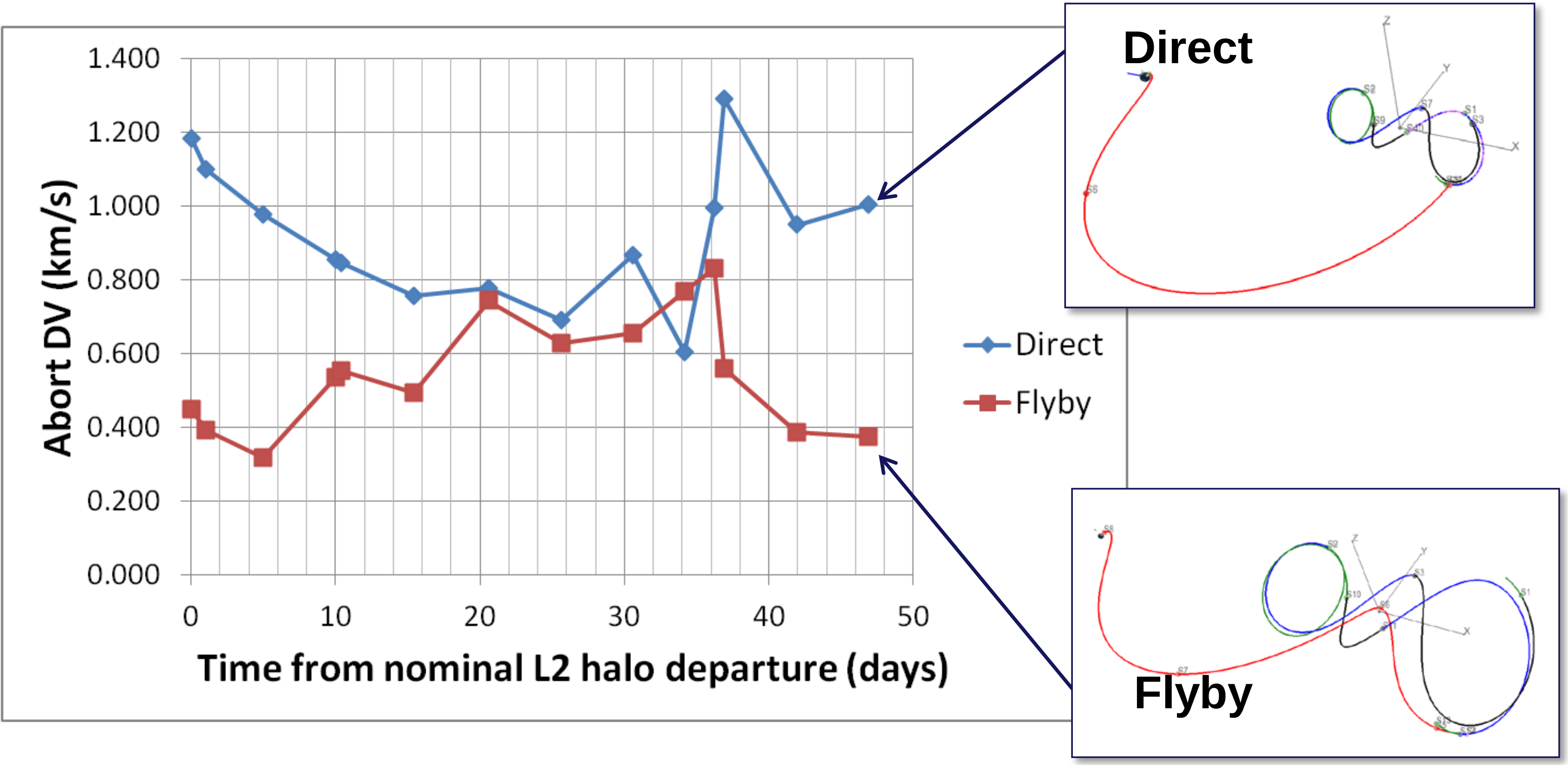
Aborts

- ☐ Earth Entry Interface (EI) Target:
  - Altitude: 121.9 km
  - Flight path angle: -5.86
- ☐ Total return time ≤ 11 days
- ☐ Best of single-impulse and two-impulse (flyby) modes



Aborts

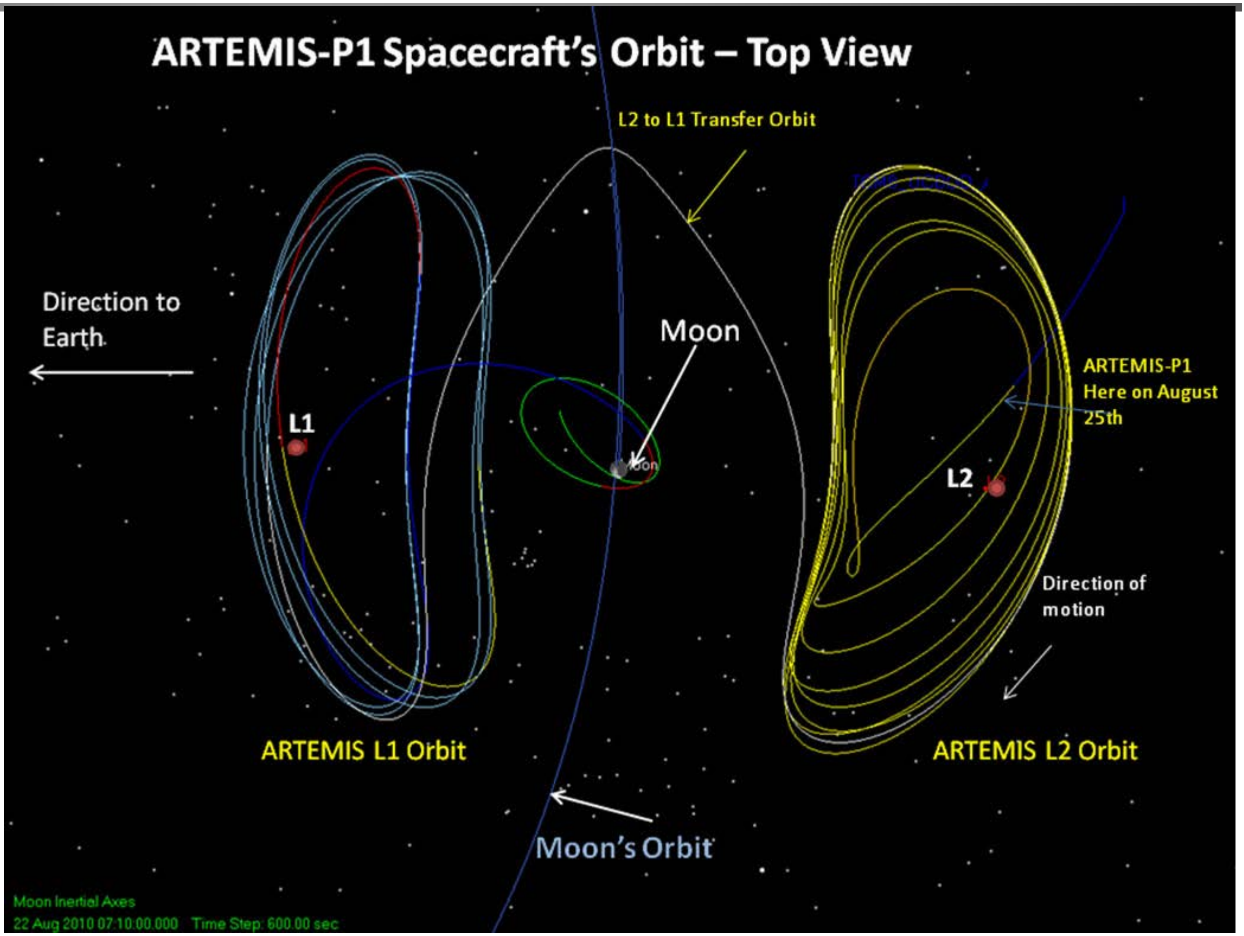
- ☐ For most cases, the flyby departure will be cheaper (but there are cases where a direct return is cheaper).



Maneuver between L2 and L1 Halo orbits – ARTEMIS Mission Example –

ARTEMIS Mission

- Spacecraft “P1” reached vicinity of L2 Aug. 2010, “P2” reached L1 Oct. 2010
- Frequent orbit maintenance (every week) required, but maneuvers were small - *equivalent to <100 m/s/year*
- P1 maneuvered from L2 to L1 Jan 2011 – *ΔV negligible, 10-day transfer*
- P1 maneuvered (90 m/s) from L1 “lissajous” to lunar orbit June 2011, P2 joined it in July (120 m/s)



	Time at L2	L2-L1 Transfer	Time at L1
P1	131 days	10 days	154 days
P2	-	-	255 days

END PRESENTATION

BACKUP

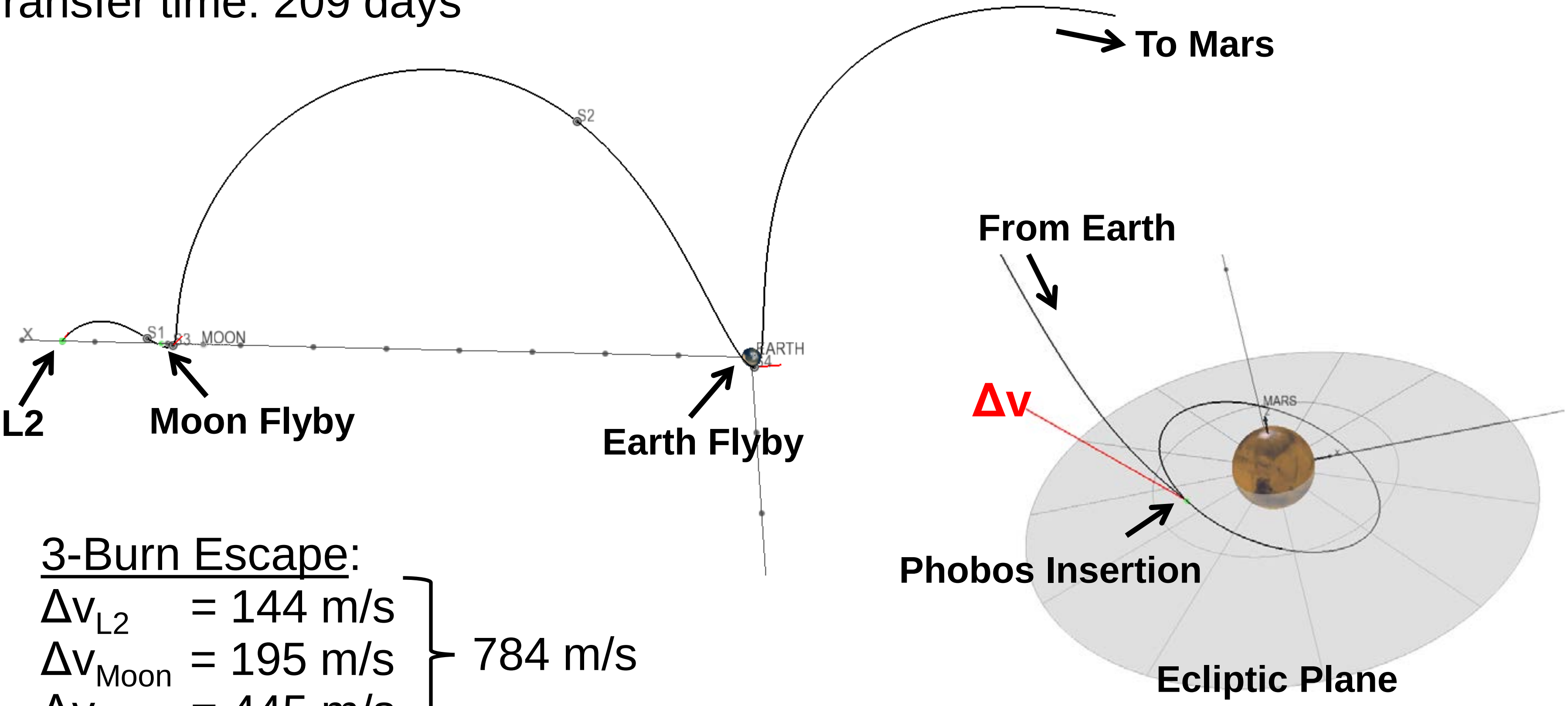
Earth-Moon Libration Point Mission Design and Performance

L2 TO MARS

Earth-Moon L2 to Phobos Orbit Insertion

Example Trajectory

Transfer time: 209 days



3-Burn Escape:

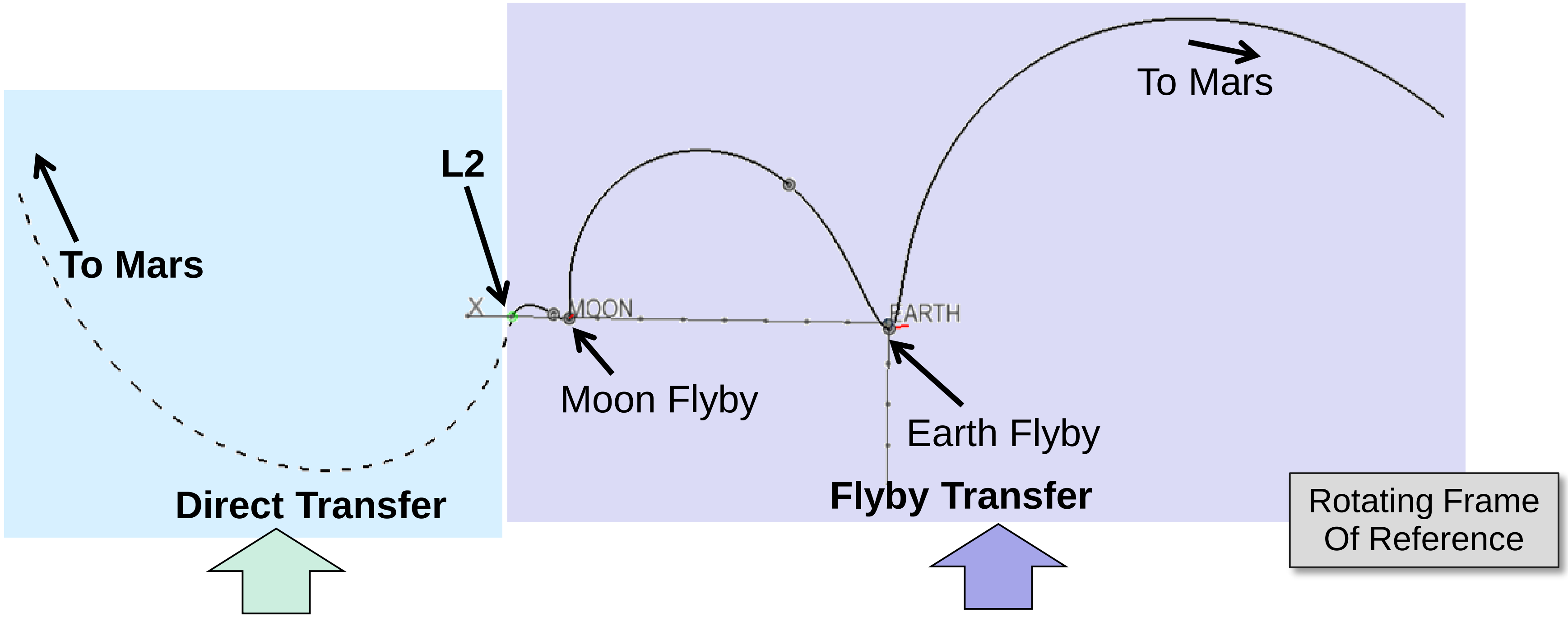
$$\left.\begin{array}{ll} \Delta v_{L2} & = 144 \text{ m/s} \\ \Delta v_{Moon} & = 195 \text{ m/s} \\ \Delta v_{Earth} & = 445 \text{ m/s} \end{array}\right\} 784 \text{ m/s}$$

$\boldsymbol{\Delta v_{Mars} = 2.135 \text{ km/s}}$

$\Delta v_{Total} = 2.921 \text{ km/s}$

Earth-Moon L2 to Phobos Orbit Insertion

- ☐ Compare to direct transfer from L2



To Mars

L2

To Mars

Moon Flyby

Earth Flyby

Direct Transfer

Flyby Transfer

Rotating Frame Of Reference

Δv_{Escape} = 2.228 km/s
Δv_{Mars} = 2.282 km/s
Δv_{Total} = $\underline{4.511 \text{ km/s}}$

Δv_{Escape} = 784 m/s
Δv_{Mars} = 2.135 km/s
Δv_{Total} = $\underline{2.921 \text{ km/s}}$

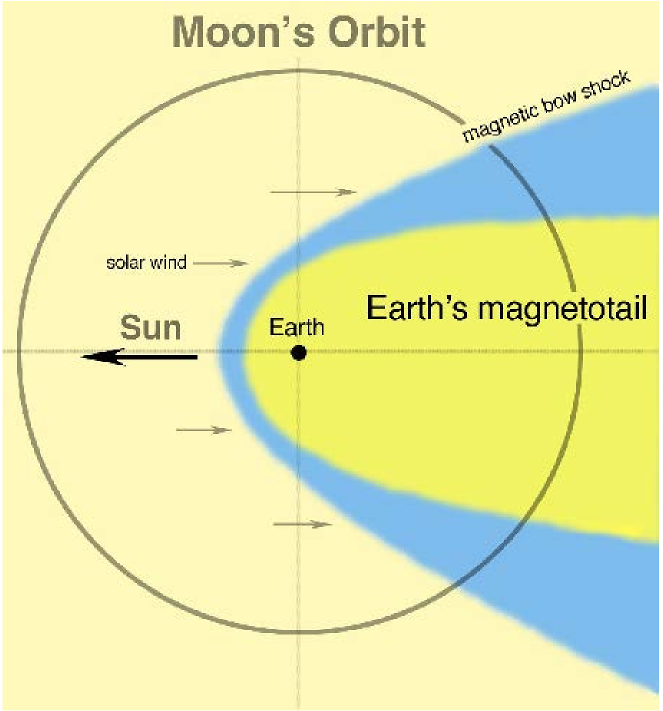
Earth_to_Mars_with_Lunar_flyby_v6+direct.ideck

Natural Environments at Earth-Moon L2 Ionizing Radiation

- Major sources of radiation
  - Galactic Cosmic Rays
  - Solar Particle Events
  - Magnetosphere

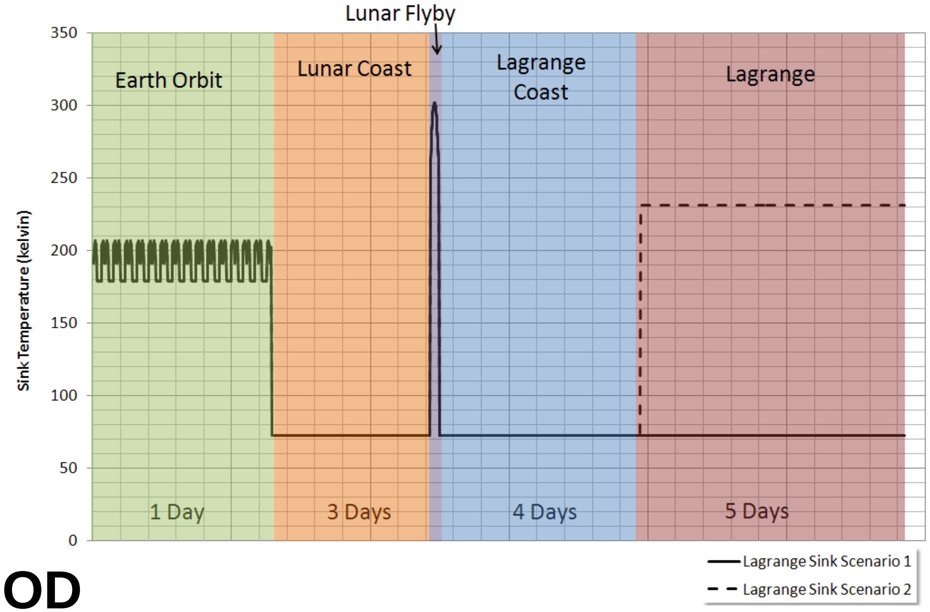
- Shielding Strategy required to protect crew.
  - Mission duration and shielding strategy determine risk
  - Short Duration free space missions (<30 days) can be conducted within current NASA Standards and risk models

- Spacecraft hardware assessment required to ensure surface charging and ionizing radiation levels at L2 are within existing hardware certification levels.



Natural Environments at Earth-Moon L2
Thermal & Micro-Meteoroid

- **Thermal**
  - Thermal environment ranges from 70 - 230 Kelvin depending on exposure to the sun
  - Thermal environment not considered an architectural driver for L2 missions



Radiator sink temperature at L2 is invariant. It can either be very cold due to deep space or at a constant sink of 230 Kelvin (1 sun)

- **MMOD**
  - Man made orbital debris not a major factor at L2 / L2 Halos, tends to “wash out” of location / orbit
  - Meteoroid risk is influenced by Earth focusing (gravitational) factor and Earth shadowing while in Earth orbit
    - Meteoroid risk far from Earth is typically less compared to meteoroid risk in LEO
  - MMOD environment not considered an architectural driver for L2 missions.

Earth-Moon L2 Halo Orbit Rendezvous
– Example Methodology –

Assumptions

- Target s/c in halo orbit about L2
- Chaser s/c inserts onto (target) halo, trailing by 10 km
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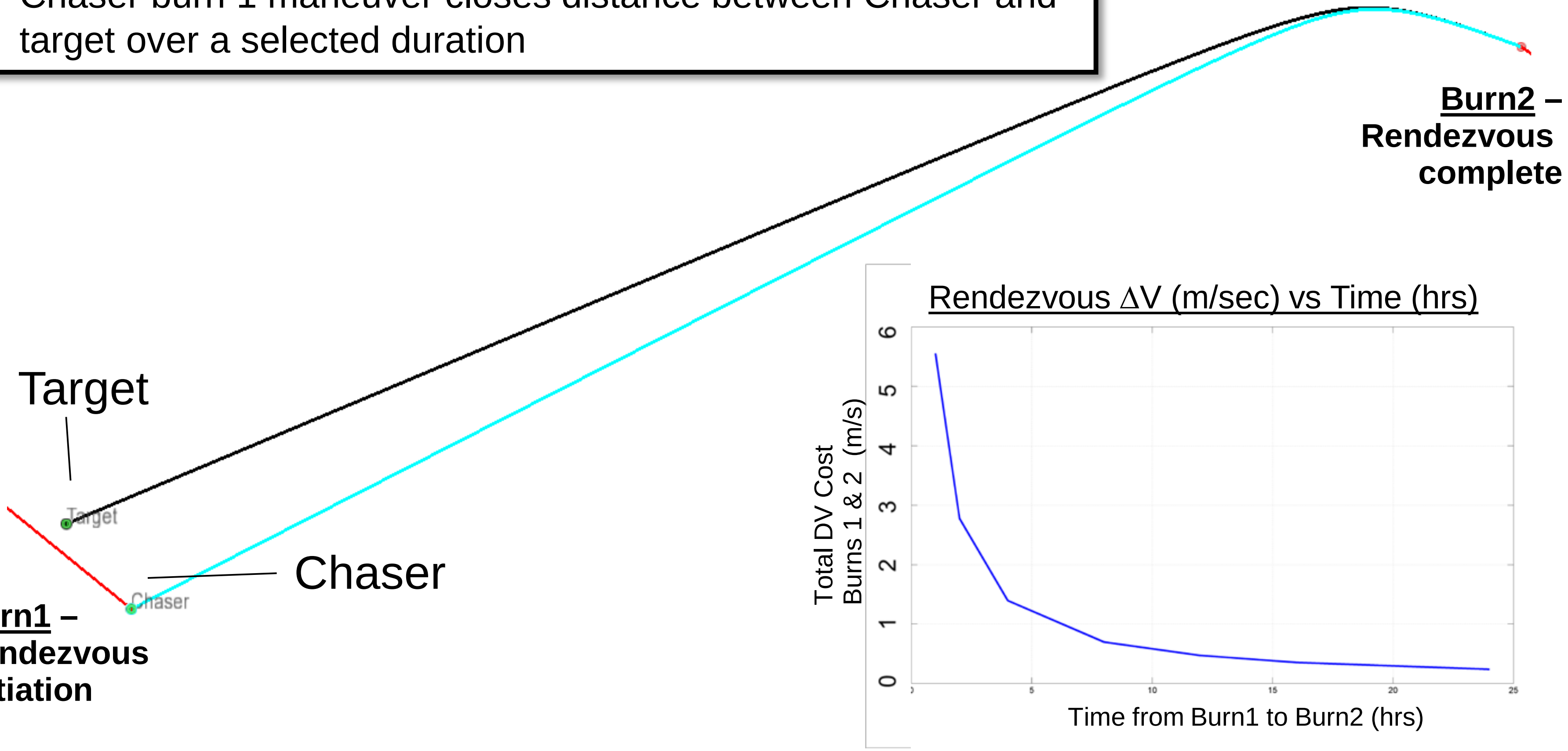


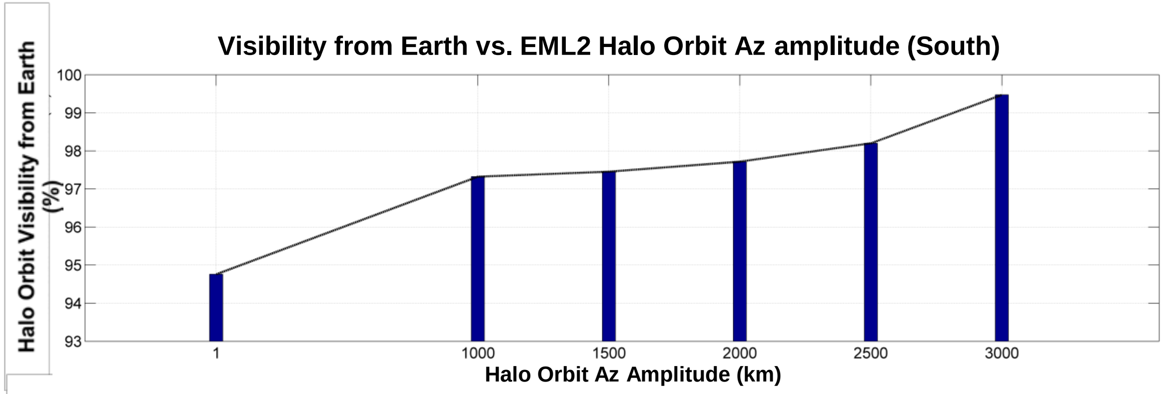
Earth Visibility from Halo Orbit
– Communications –

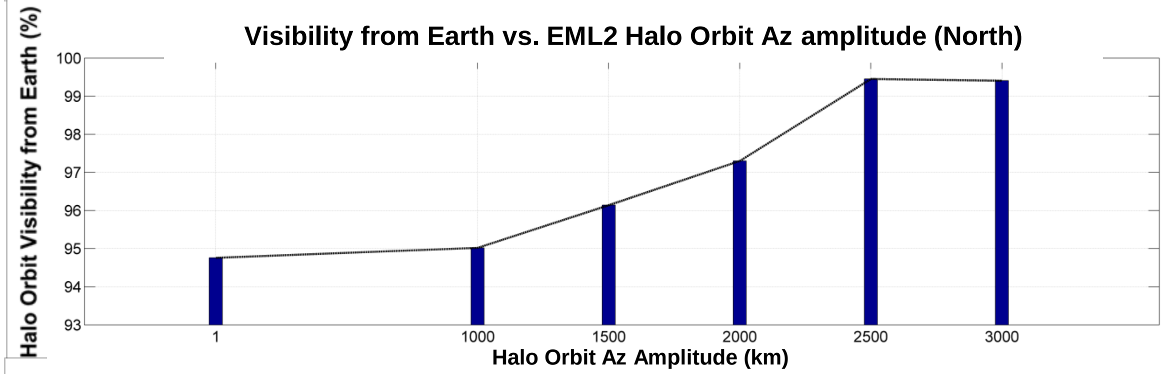
Assumptions

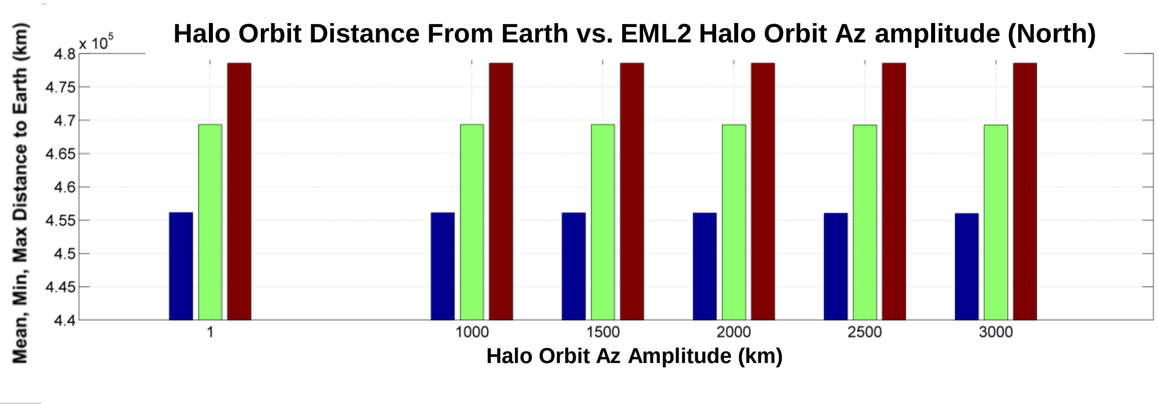
- Visibility based on ability to view at least one of three DSN sites from the spacecraft on the halo
- There are North and South halo orbit families

Observations

- Visibility to Earth is affected by halo orbit amplitude (94%+)
- Visibility to Earth is essentially independent of distance from the Earth


Visibility from Earth vs. EML2 Halo Orbit Az amplitude (South)


Visibility from Earth vs. EML2 Halo Orbit Az amplitude (North)


Halo Orbit Distance From Earth vs. EML2 Halo Orbit Az amplitude (North)

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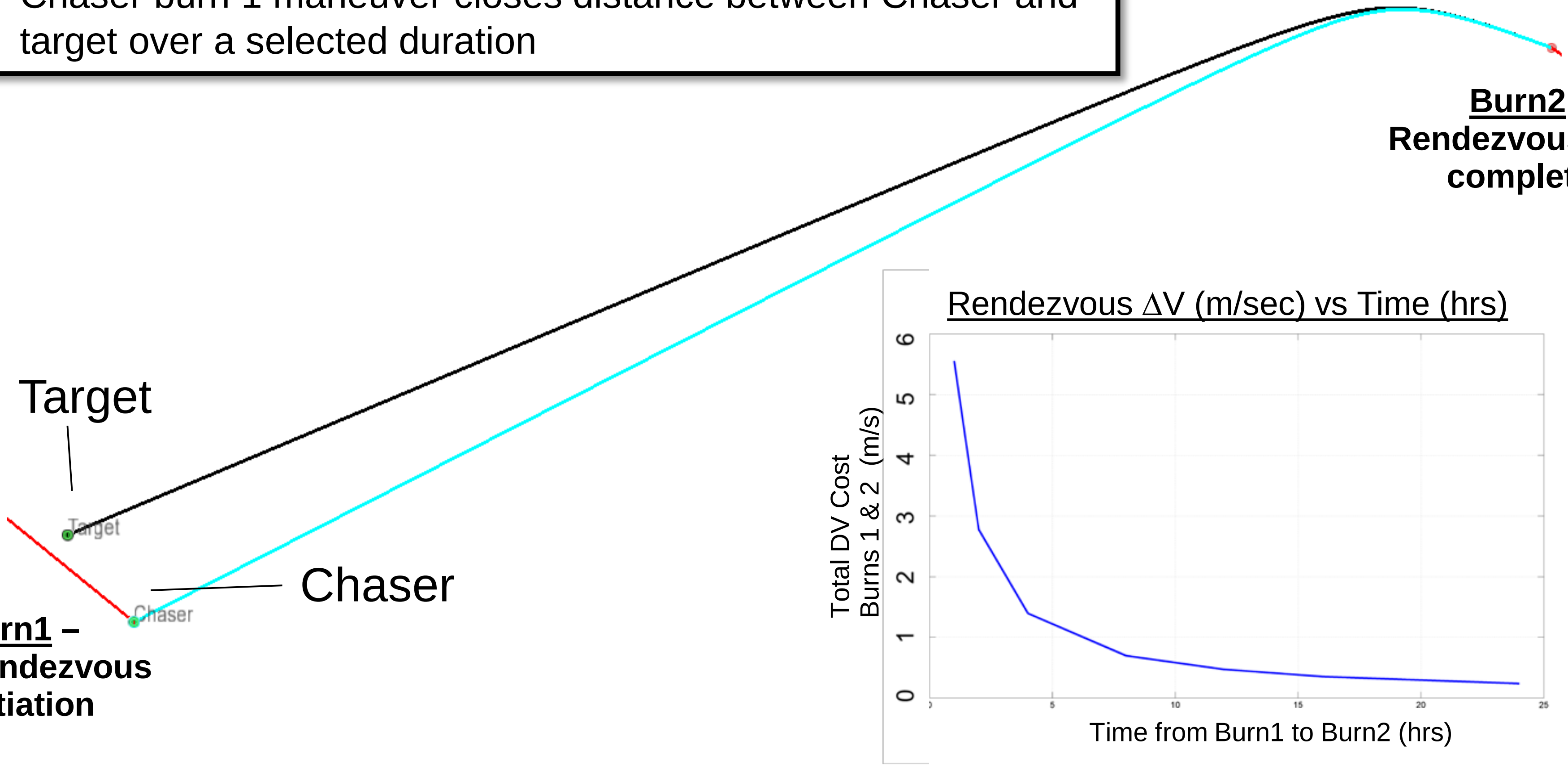


Burn2 – Rendezvous complete

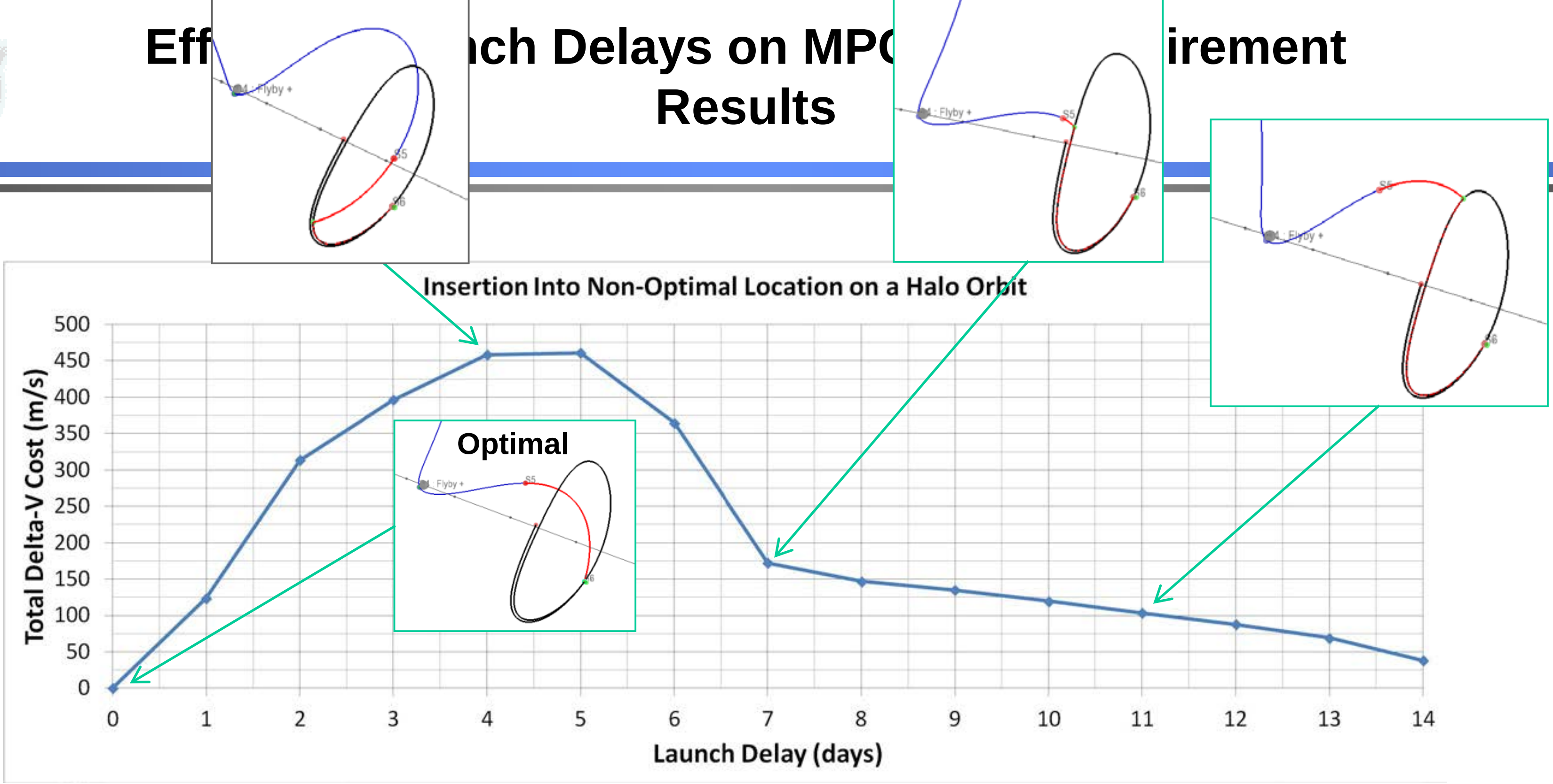
Target

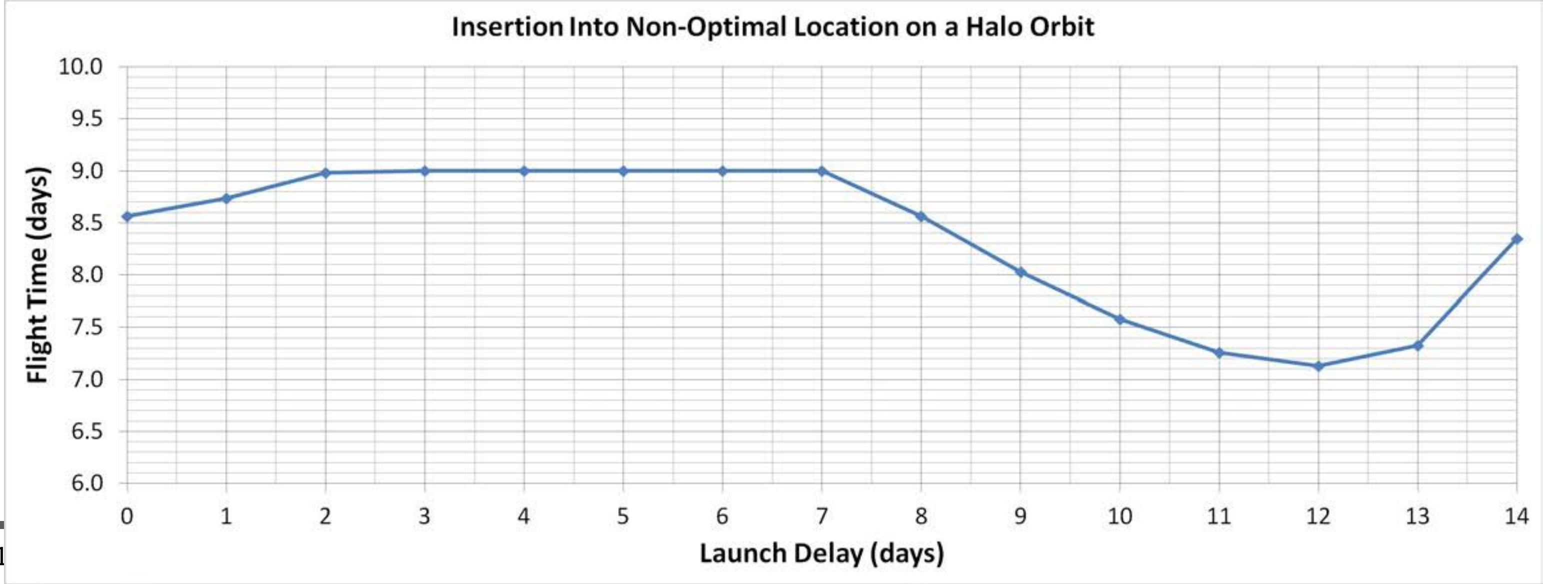
Chaser

Burn1 – Rendezvous initiation

Rendezvous ΔV (m/sec) vs Time (hrs)

Eff ch Delays on MPC irement
Results





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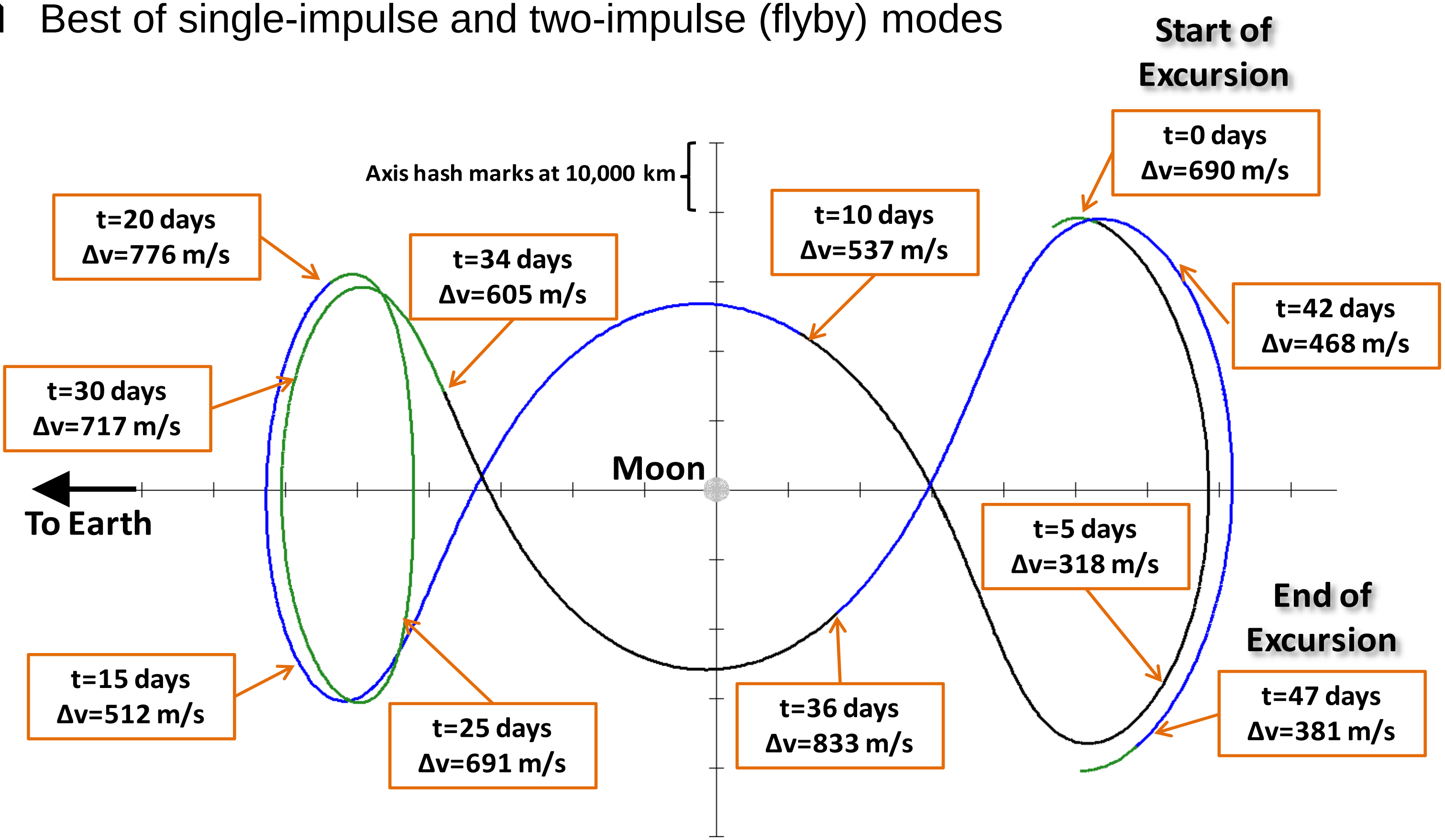
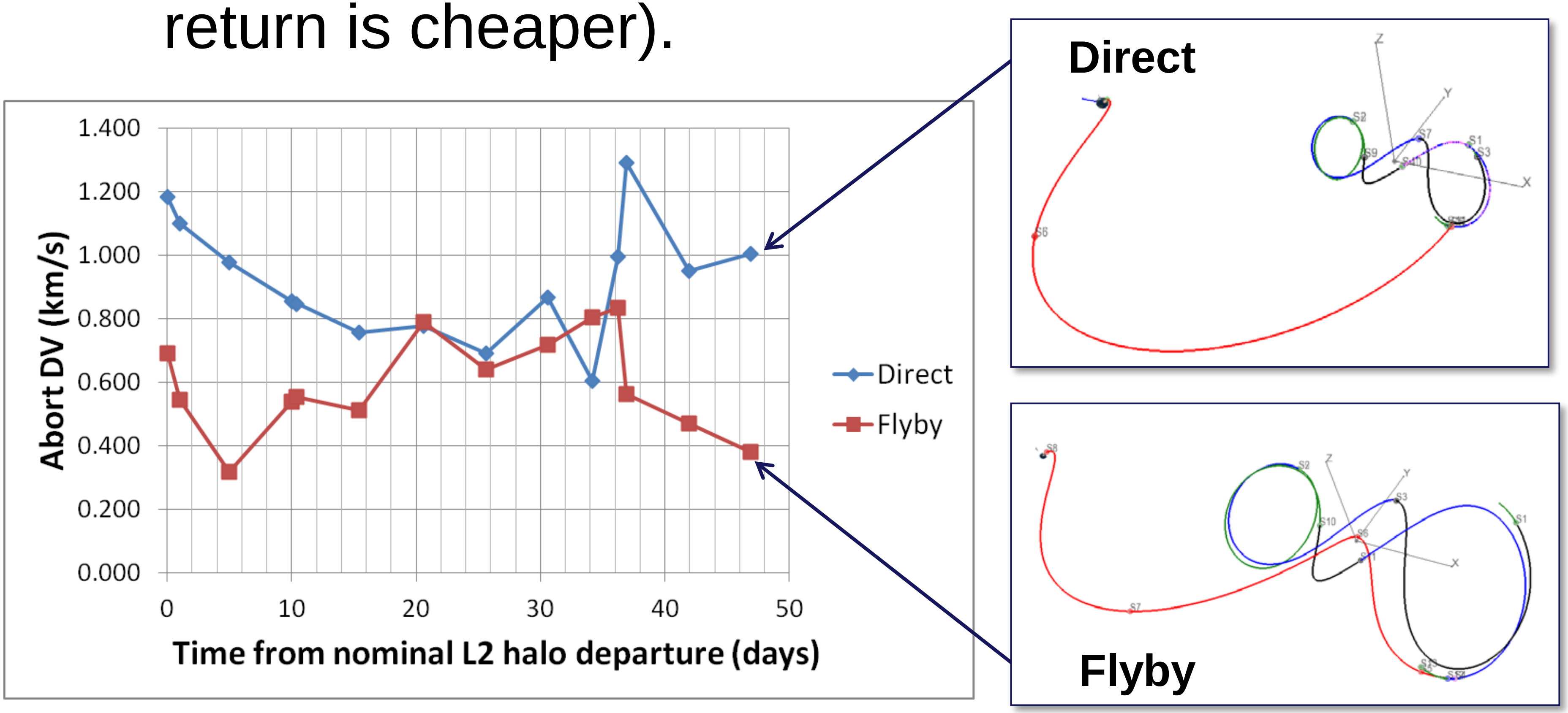


Abort Assessments

- ☐ For most cases, the flyby abort will be cheaper (but there are cases where a direct return is cheaper).



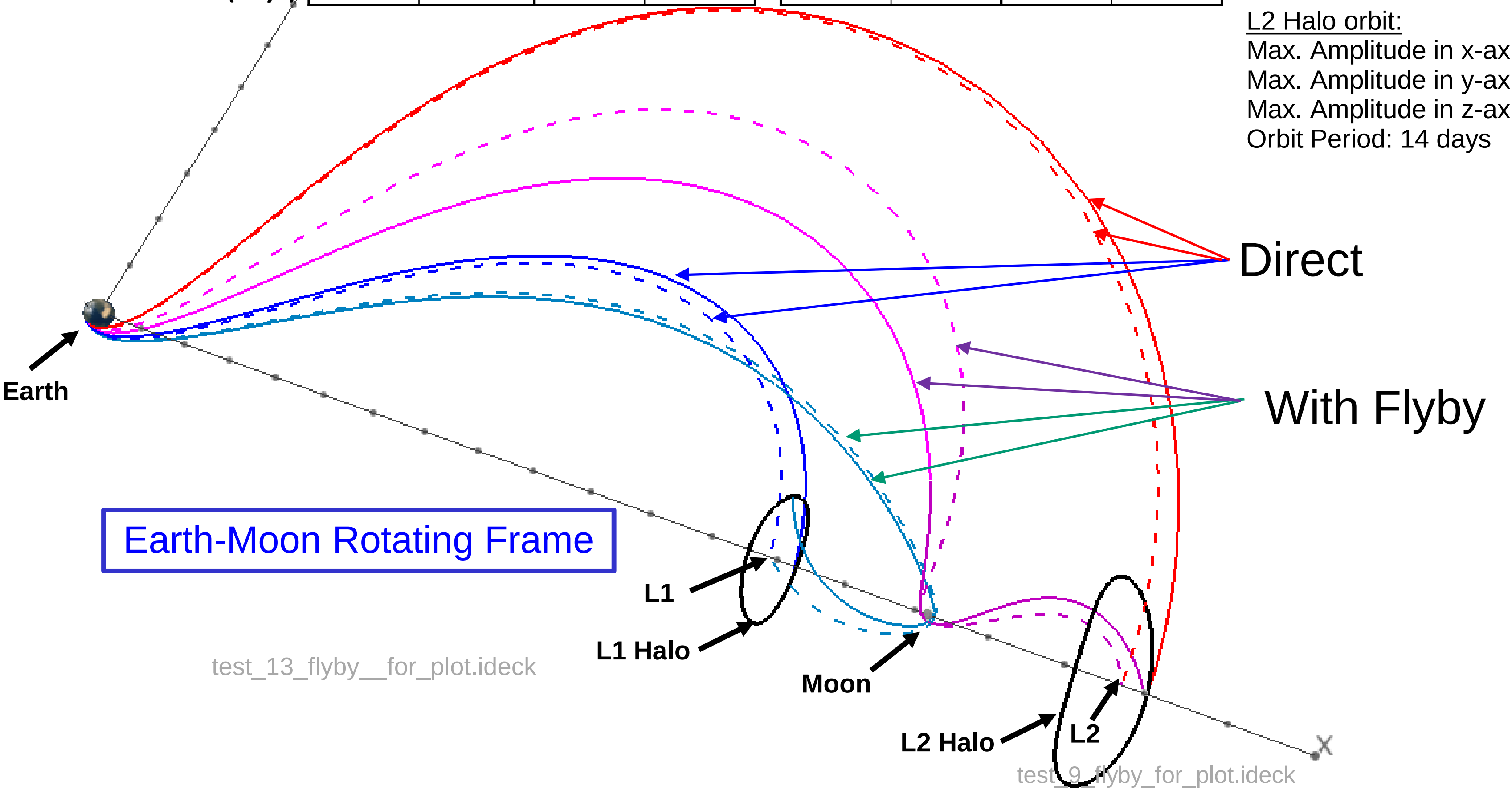
LEO to Earth-Moon L1 & L2
Direct and Flyby – Example Cases

Destination	L1		L1 Halo		L2		L2 Halo	
Mission Type	direct	flyby	direct	flyby	direct	flyby	direct	flyby
LEO DV (km/s)	3.116	3.154	3.118	3.155	3.149	3.134	3.151	3.133
Lunar Periapsis DV (km/s)	0.000	0.298	0.000	0.274	0.000	0.186	0.000	0.175
Capture DV (km/s)	0.716	0.411	0.605	0.287	1.107	0.148	0.957	0.109
Total DV (km/s)	3.832	3.863	3.723	3.716	4.256	3.468	4.108	3.416
Transfer Time (days)	3.76	4.38	3.94	4.83	6.14	8.53	6.29	8.35

LEO inclination: 28.5 deg

L1 Halo orbit:
Max. Amplitude in x-axis: 6,537 km
Max. Amplitude in y-axis: 23,445 km
Max. Amplitude in z-axis: 10,000 km
Orbit Period: 12 days

L2 Halo orbit:
Max. Amplitude in x-axis: 11,904 km
Max. Amplitude in y-axis: 34,672 km
Max. Amplitude in z-axis: 10,000 km
Orbit Period: 14 days



ASTEROID UTILIZATION MISSION

Status 1

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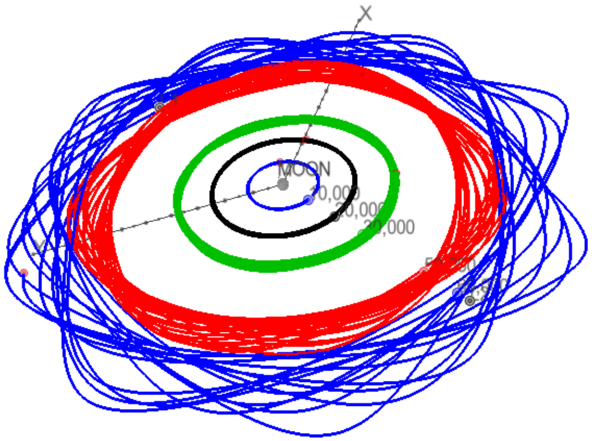
Contributors

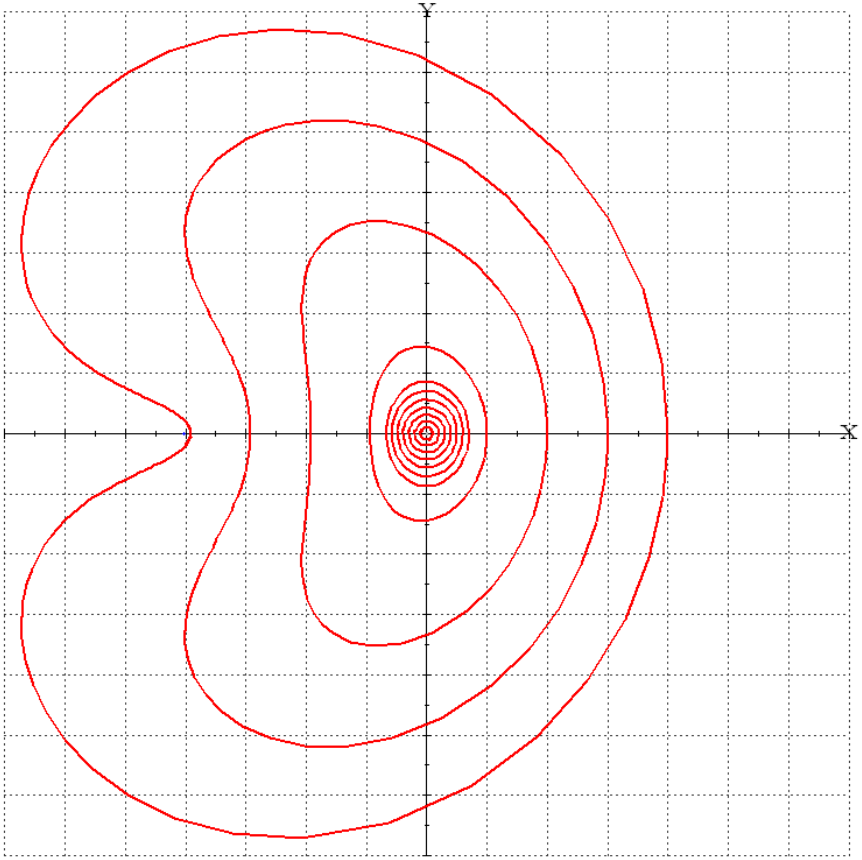
Jerry Condon	**JSC/EG5**	**Integrated Performance Lead**
Jacob Williams	**JSC/ESCG**	**Trajectory/Performance**
David Lee	**JSC/EG5**	**Asteroid Capture - Consult**

Overview of Slides

1. **DRO Tutorial**
2. **MPCV Transfer Example Cases**
3. **Overview of Trade Studies to Assess MPCV Accessibility to the Asteroid Targets**

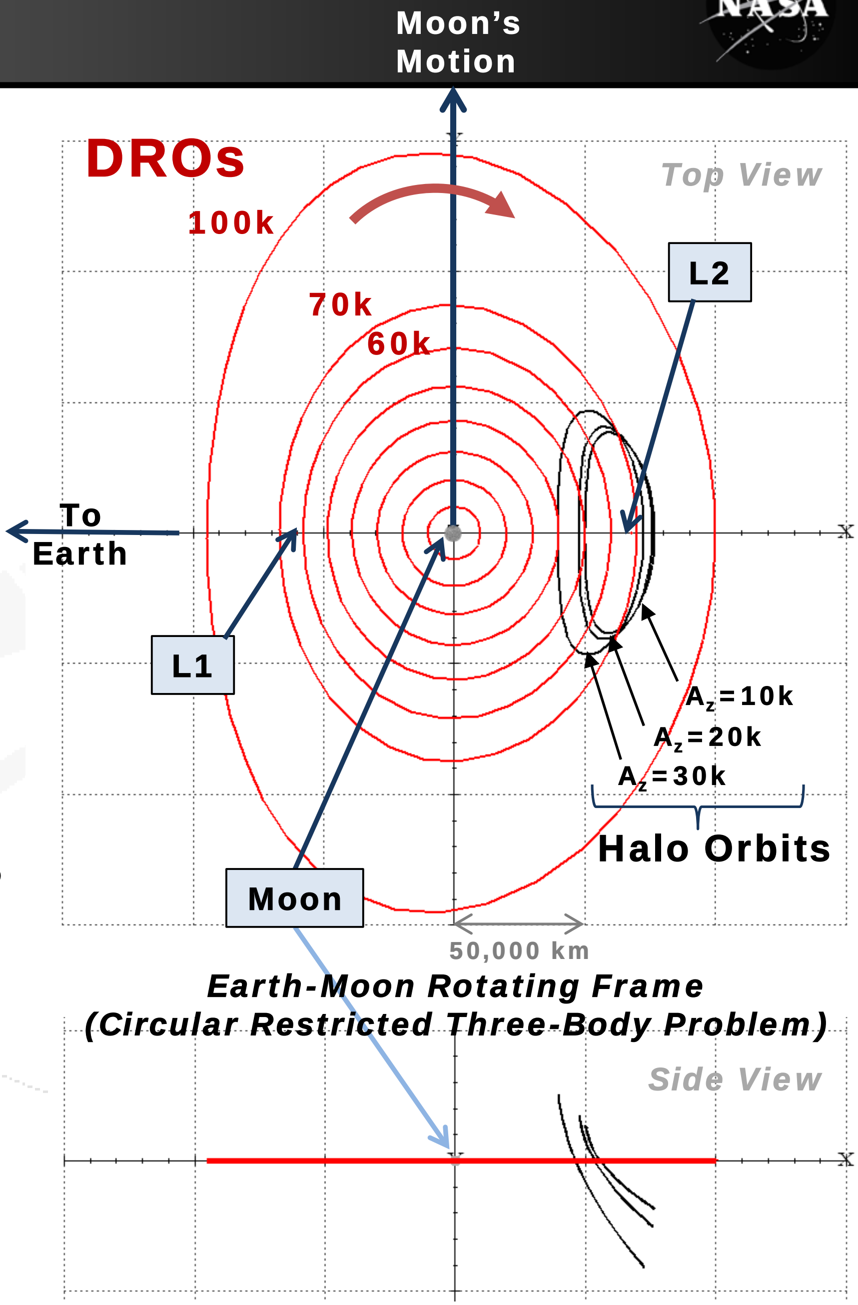
DRO Tutorial





Background

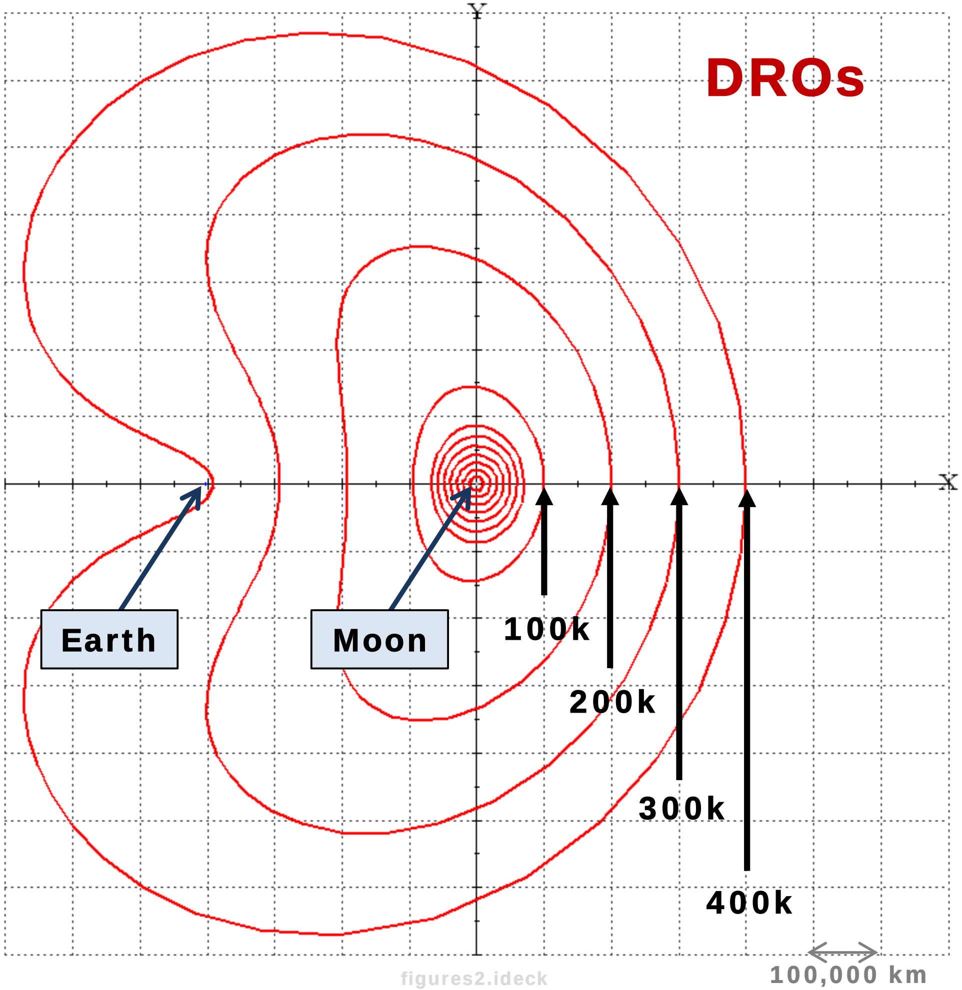
- A Distant Retrograde Orbit (DRO) is another family of stable orbits found in the circular restricted three-body problem (CRTBP).
- In the Earth-Moon rotating frame, they look like circular to elliptical orbits around the Moon (with the Moon at the center of the orbit).
- DROs orbit the moon in a retrograde direction from moon's orbit/rotation.
- Some DROs are very stable over long periods of time (≥100 years) with no orbit maintenance, even with a "real" force model.



Earth-Moon Rotating Frame (Circular Restricted Three-Body Problem)

DROs In The CR3BP

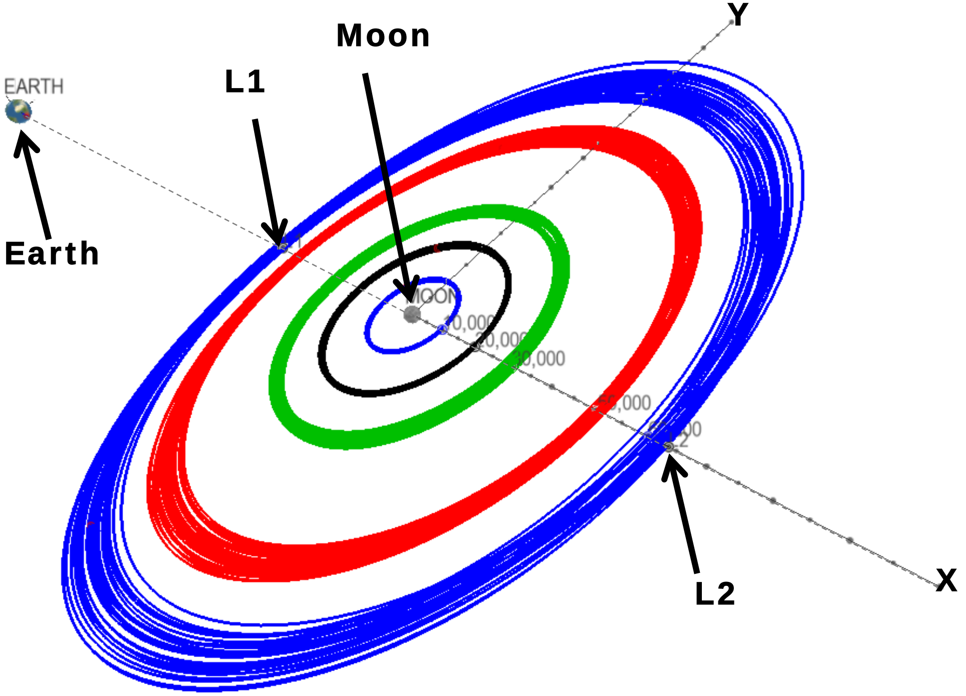
- **DRO shapes transition with increases in altitude**
  - Nearly circular near the Moon
  - Transitioning to an elliptical shape
  - Then becomes more kidney-shaped
  - Then becomes more cardioid shaped
- **In a “real” system, 50,000 to 70,000 km altitudes are stable for > 100 years**
  - Additional work required to determine stability of higher altitudes



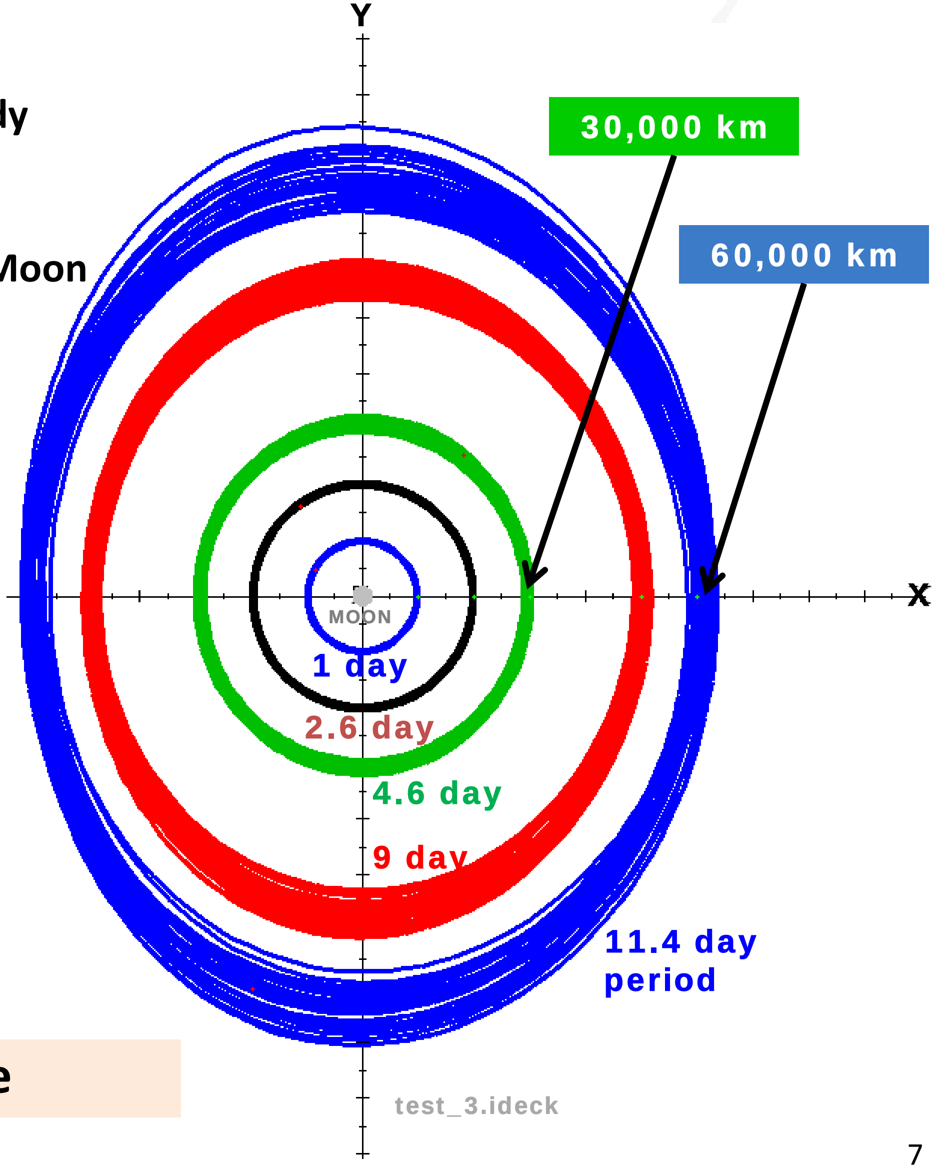
Earth-Moon Rotating Frame
(Circular Restricted Three-Body Problem)

Example DRO’s (Real Force Model)

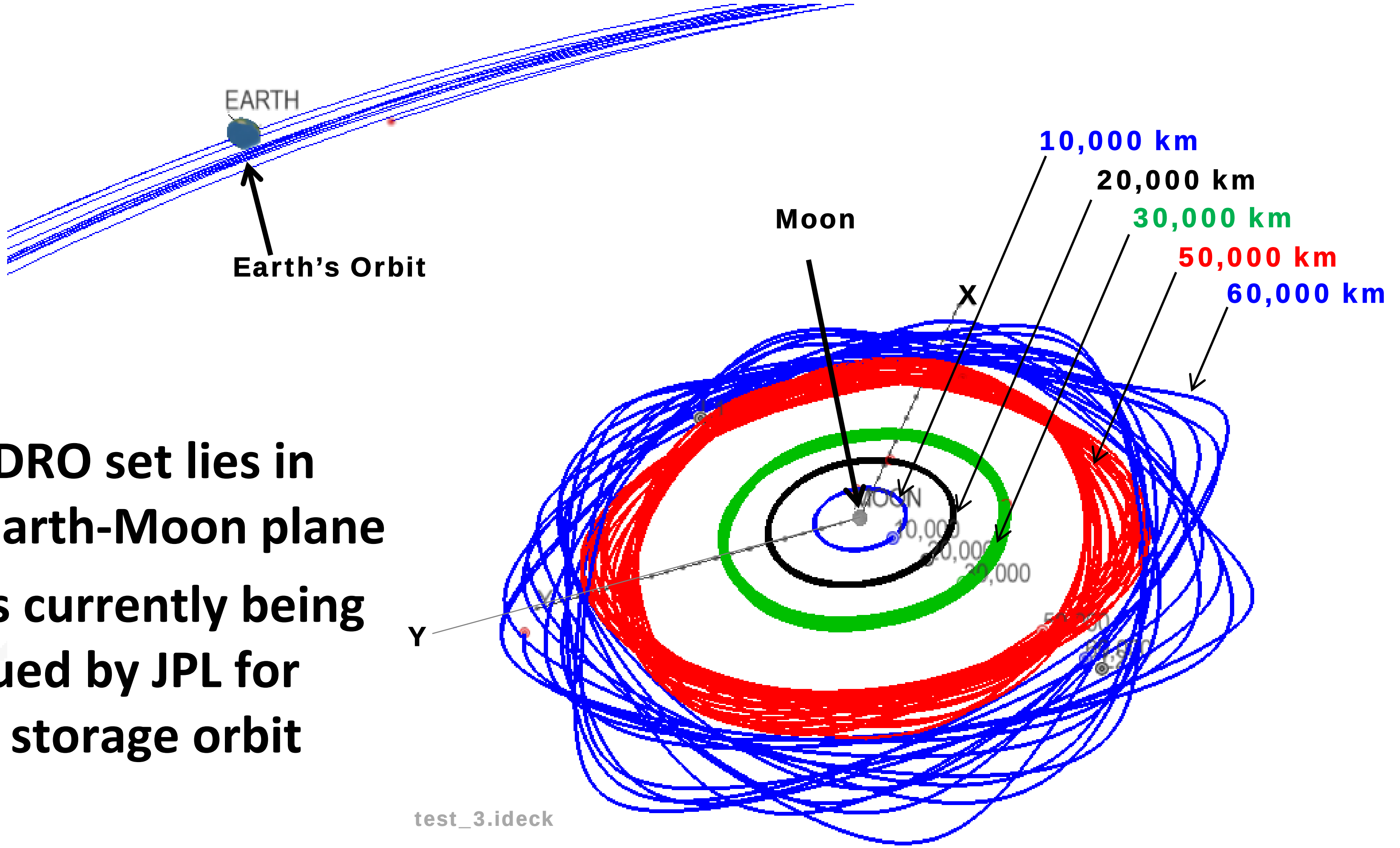
- Classified by x-axis crossing radius (b) - like a semi-minor axis: 10,000 – 60,000 km shown here
- Stable for long periods (300 days shown here)
- DROs are periodic in Circular Restricted Three Body Problem (CRTBP)
- DROs are quasi-periodic in the real force model
- Increasing period with increasing altitude above Moon



Earth-Moon Rotating Frame



Example DRO's: Inertial Frame (Moon-Centered)

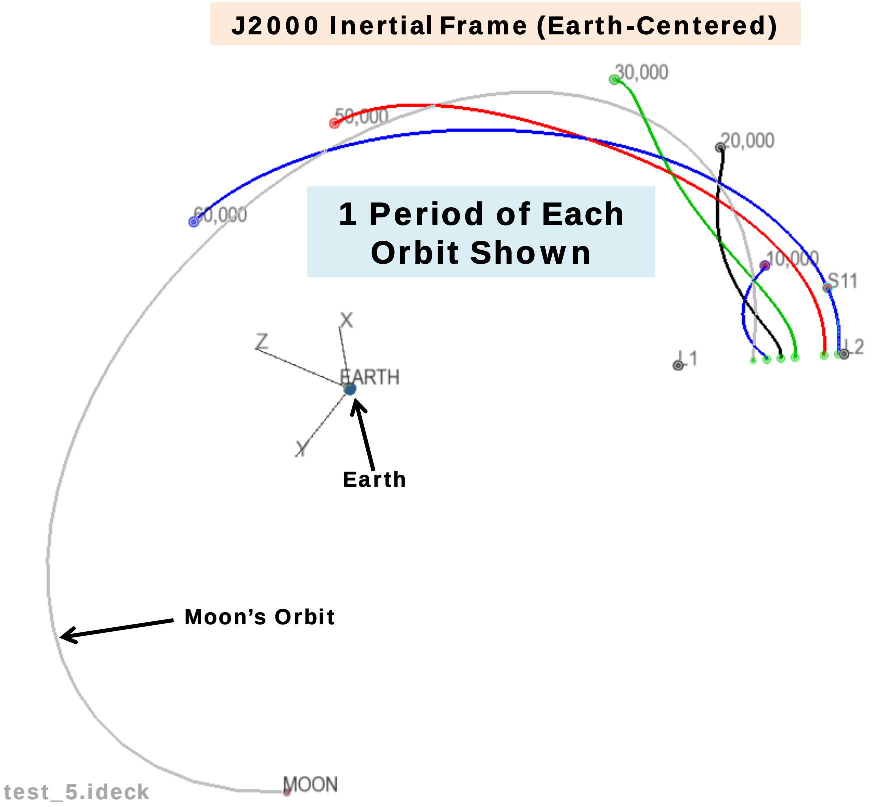


- This DRO set lies in the Earth-Moon plane
- DROs currently being pursued by JPL for ARM storage orbit

J2000 Inertial Frame (Moon-Centered)

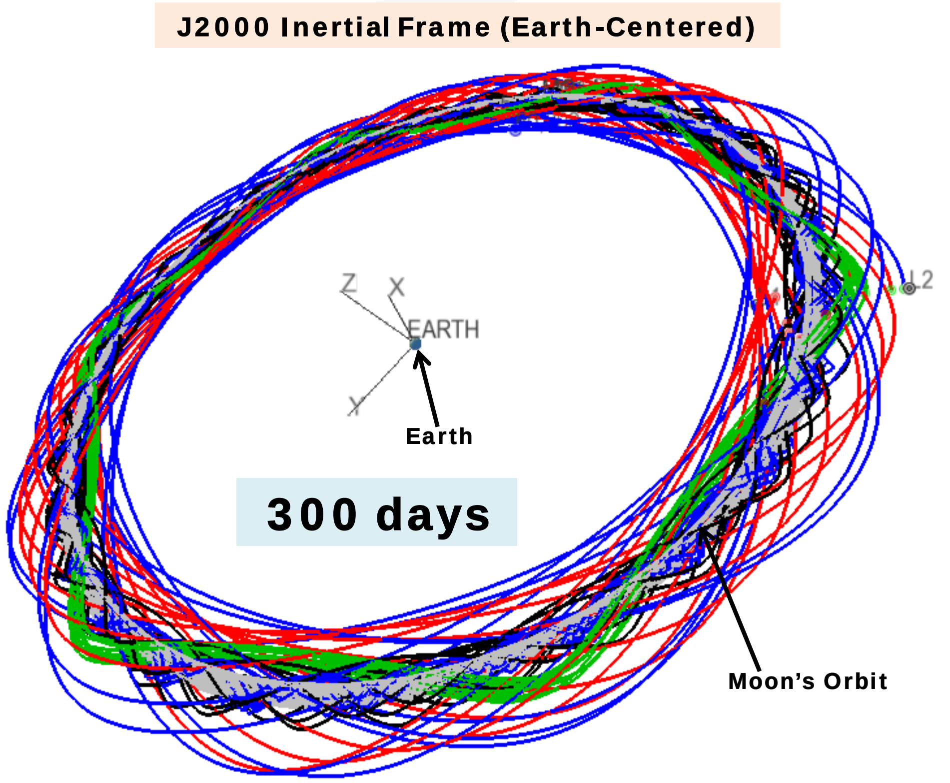
Example DRO's: Inertial Frame (Earth-Centered)

- **From an Earth perspective, a DRO will reside in the general vicinity of the moon and so a spacecraft in a DRO will orbit Earth about every 28 days**
- **A spacecraft in a DRO will have the same approximate position, rotational velocity, and inclination as that of the moon**



J2000 Inertial Frame (Earth-Centered)

1 Period of Each Orbit Shown



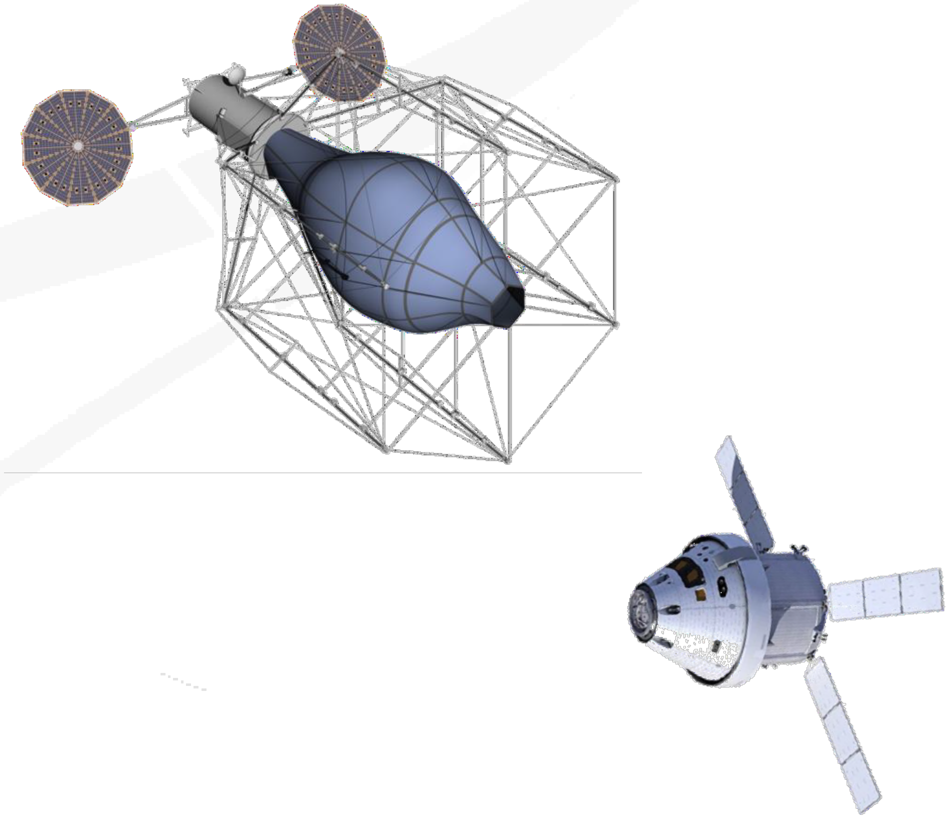
J2000 Inertial Frame (Earth-Centered)

300 days

Summary

- “Lunar” DROs cycle in the vicinity of the moon with a range of altitudes
- DROs are generally stable (particularly lower altitude DROs)
- They can propagate for many years (some cases > 100 years) without maintenance
- In an Earth-Moon rotating frame, DROs are a quasi-periodic orbit (circle, ellipse, kidney – shaped)
- From Earth perspective, DROs approximately follow the lunar orbit motion

Round-Trip MPCV Mission TO DRO (Example Cases)



Overview

- **This assessment contains a round-trip MPCV mission to a distant retrograde orbit (DRO) around the Moon.**
  - In support of a rendezvous mission with a pre-emplaced asteroid
- **This is a feasibility assessment (i.e. just a single case) and not a trade study.**
  - It is not a best or worst case.
  - Mission performance requirements and opportunities would be better revealed by trade studies

Assumptions

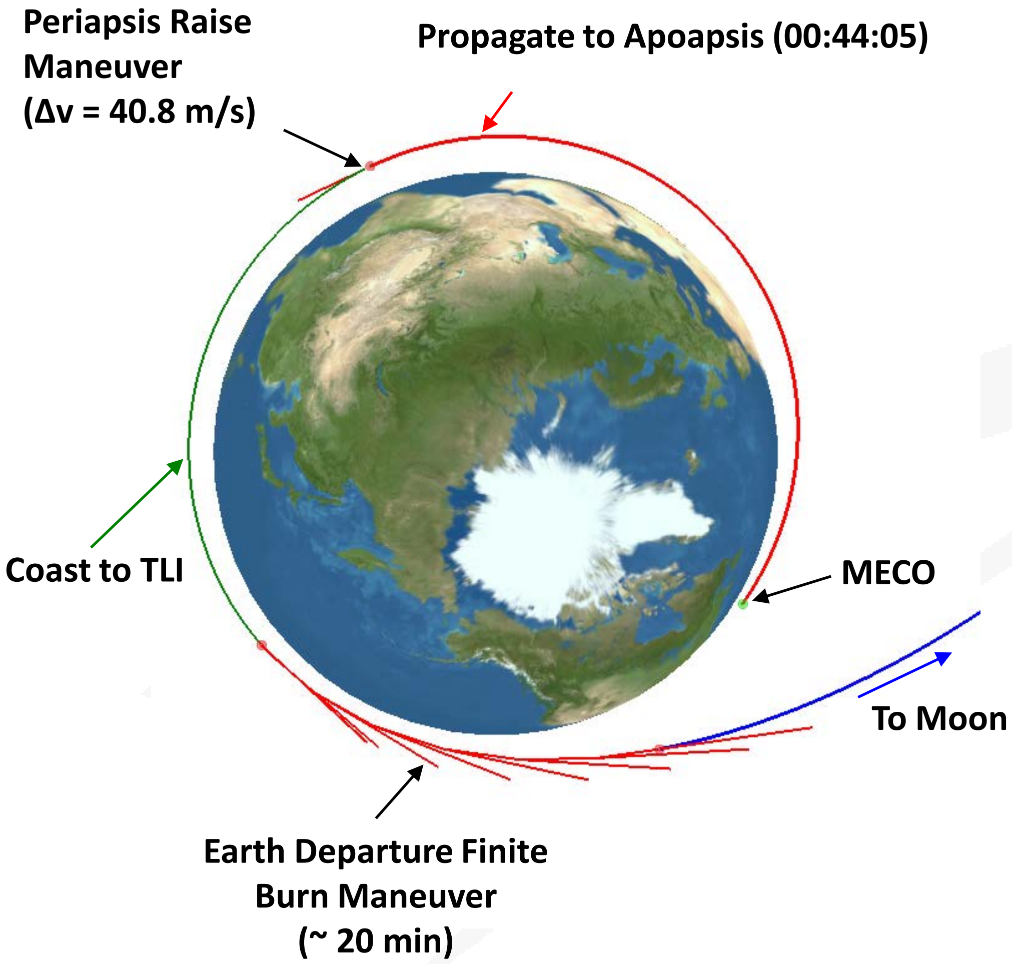
- **Impulsive (Δv) maneuvers (optimized)**
- **DRO size (x-axis crossing distance) = 62,000 km**
- **Outbound:**
  - iCPS MECO state (see next slide)
  - iCPS Capability:
    - Total Δv: 2,900 m/s
    - 40.77 m/s used for the PRM (raises perigee from 40.7 to 185 km, with an 1806 km apogee)
    - Estimate about 80 m/s gravity for losses,
    - So, the impulsive limit for TLI is 2,779.23 m/s
  - iCPS Earth departure maneuver, followed 30 seconds later by an MPCV departure maneuver (if necessary to complete TLI)
  - Powered lunar flyby (minimum 100 km altitude)
- **Inbound:**
  - Direct return (no flyby)
  - EI Altitude: 121.92 km
  - EI FPA: -5.86 deg
- **Total mission duration ≤ 21 days (optimized)**
- **At least one day stay in the DRO (optimized)**

Assumptions: MECO State

- **MECO State (EPM_OF_DATE Frame, Earth-centered)**
  - Radius magnitude (km): 6535.55695654
  - Longitude (deg): -66.5737779
  - Geocentric latitude (deg): 27.5046213
  - Velocity magnitude (km/s): 8.20892236
  - Geocentric azimuth (deg): 97.76124785
  - Geocentric flight path angle (deg): 3.46249997

Source: 080212 Mass properties for the L2 Waypoint analysis_Gutkowski.xlsx

Assumptions: Earth Departure



Example: MECO at 2017-Dec-12 08:14:14 TDB

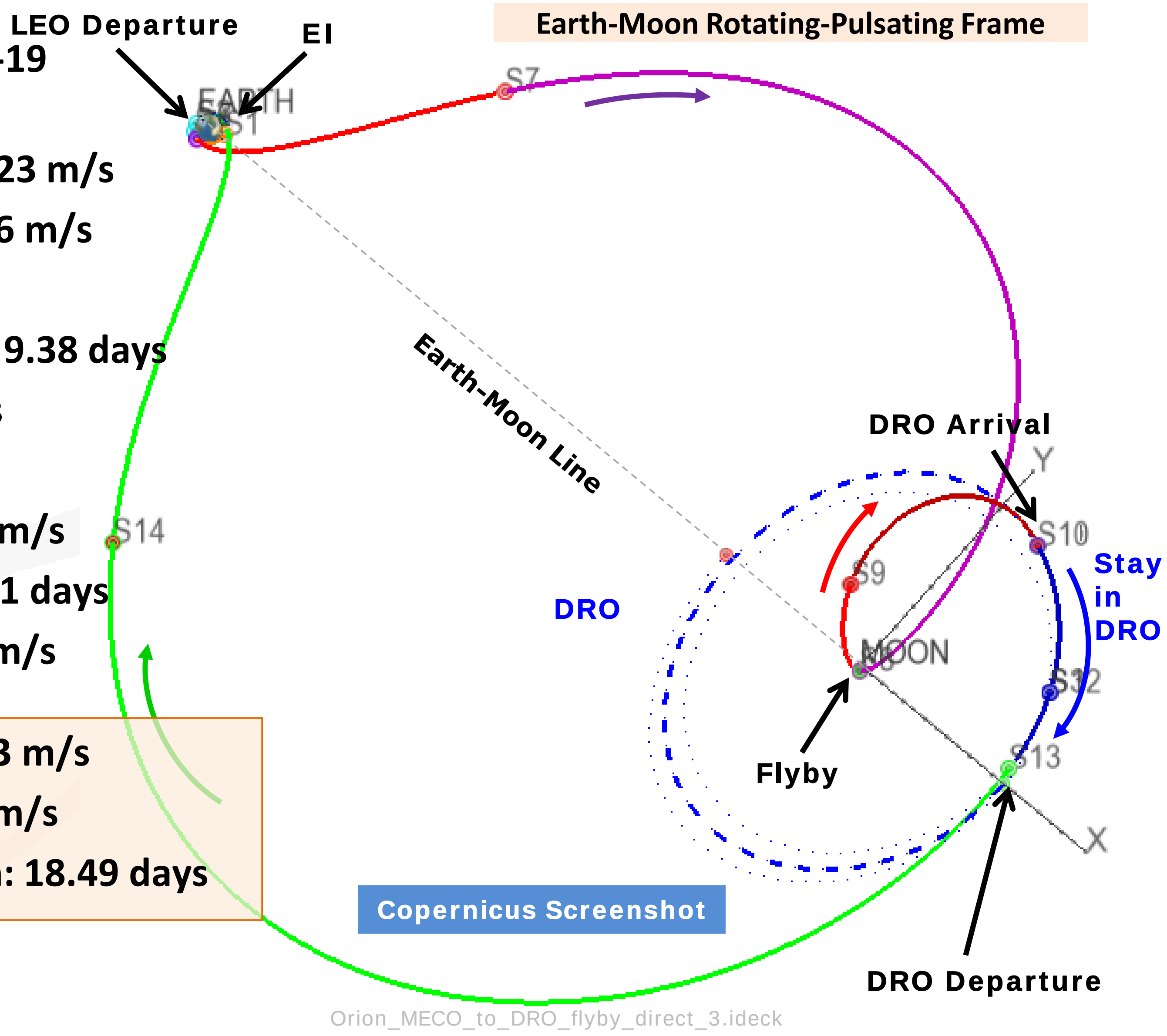
hybrid_8x8_20171122_240030_base.ideck

- **Fixed flight time from MECO to apoapsis (44 min).**
- **Fixed maneuver to raise periapsis to 185 km altitude (40.8 m/s).**
  - **The 185 km periapsis is the actual propagated value (not osculating).**
  - **8x8 GGM02C Earth gravity model**
- **Optimized coast to the Earth departure maneuver.**
- **Finite Burn Earth departure maneuver with optimized control law and burn duration:**
  - **VUW Frame**
  - $\alpha, \dot{\alpha}, \beta, \dot{\beta}$ **SOC control law**

Results (Impulsive Only)

- MECO Epoch: 2021-Jul-19 15:59:17 TDB
- iCPS Departure: 2,779.23 m/s
- MPCV Departure: 60.36 m/s
- Flyby : 179.1 m/s
- Outbound Flight Time: 9.38 days
- DRO Arrival: 124.6 m/s
- Stay Time: 2.49 days
- DRO Departure: 577.4 m/s
- Return Flight Time: 6.61 days
- Entry velocity: 11.00 km/s

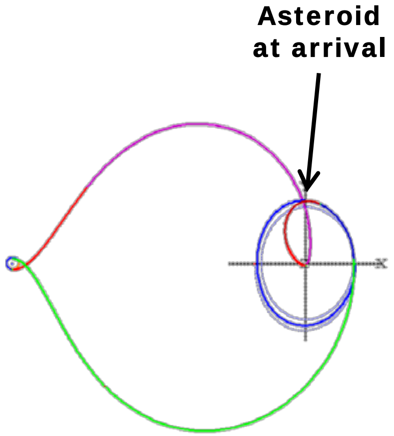
- Total iCPS Δv: 2,779.23 m/s
- Total MPCV Δv: 941.5 m/s
- Total Mission Duration: 18.49 days

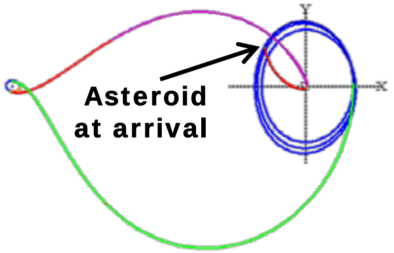


Phasing

- **Assume the asteroid is at the location of the optimal insertion on previous slide (the 2021-Jul-19 departure epoch).**
- **Subsequent opportunity (next month) will have a higher cost due to the non-optimal phasing (assuming the asteroid cannot be moved once placed in orbit)**
- **Trade studies will be necessary to assess asteroid accessibility over time for different sized DROs and asteroid insertion phase.**

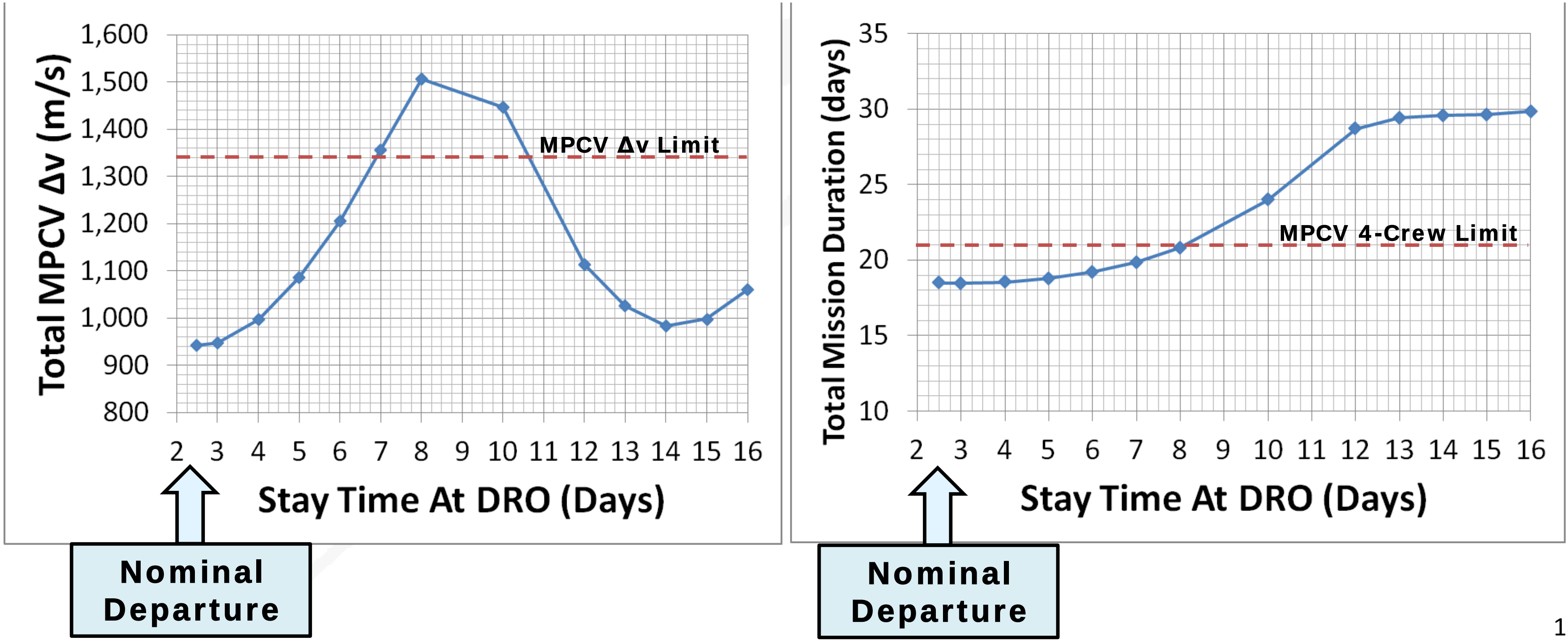
Departure Epoch	MPCV Cost (m/s)	Stay Time (days)	Total Mission Time (days)
2021-Jul-19 15:59:17	941.5	2.49	18.49
2021-Aug-14 12:01:35	1,209.7	4.94	16.58



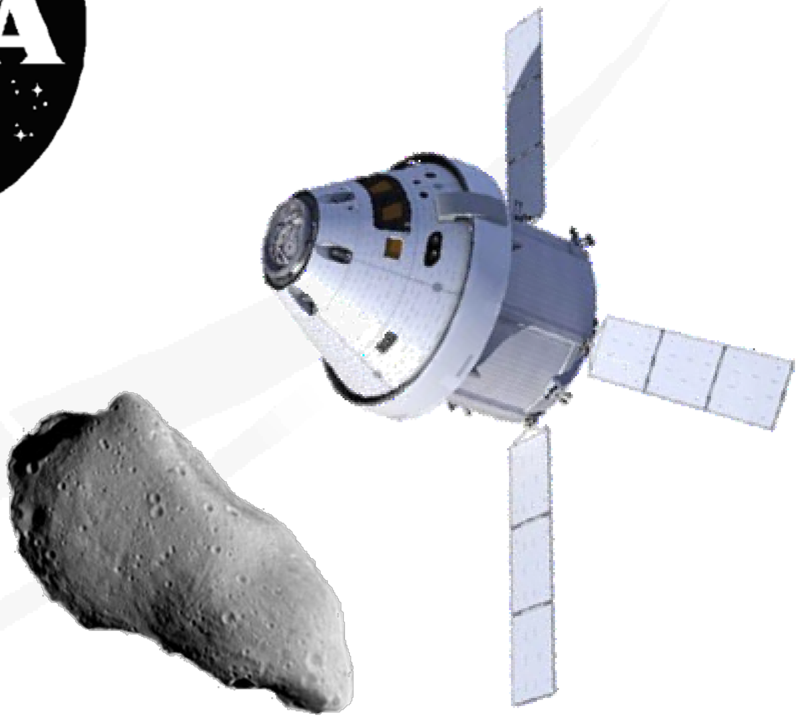


Delayed DRO Departure

- **Using the nominal case (MECO Epoch: 2021-Jul-19), delay the departure from the asteroid and re-optimized the Earth return.**
  - Direct return.
  - The EI constraint remains altitude and flight path angle only.



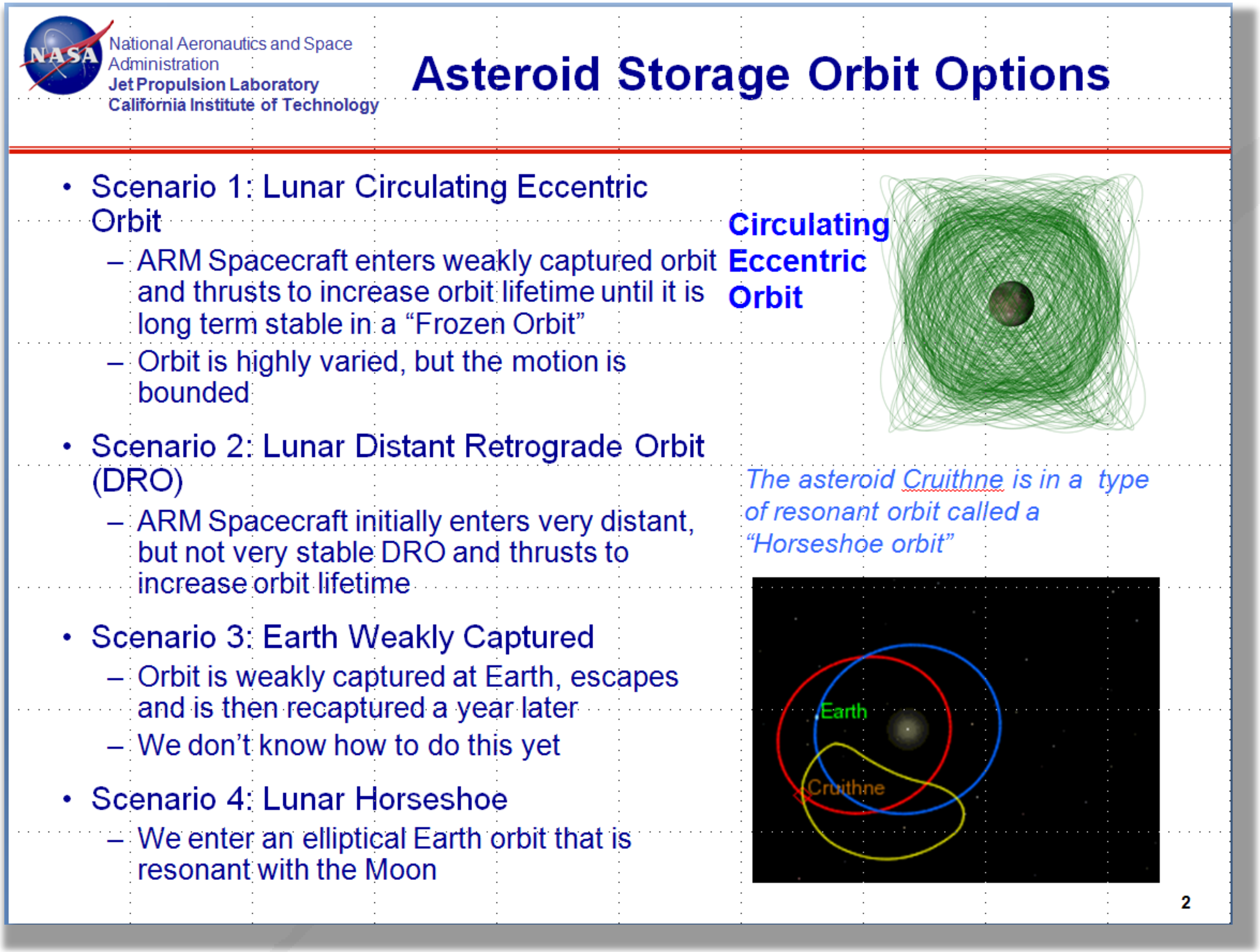
Delayed DRO Departure



Orion Performance to Asteroid Rendezvous Targets

Asteroid Storage Orbits

- **Currently monitoring JPL assessment of possible asteroid storage orbits**



National Aeronautics and Space Administration
Jet Propulsion Laboratory
California Institute of Technology

Asteroid Storage Orbit Options

- Scenario 1: Lunar Circulating Eccentric Orbit
  - ARM Spacecraft enters weakly captured orbit and thrusts to increase orbit lifetime until it is long term stable in a "Frozen Orbit"
  - Orbit is highly varied, but the motion is bounded
- Scenario 2: Lunar Distant Retrograde Orbit (DRO)
  - ARM Spacecraft initially enters very distant, but not very stable DRO and thrusts to increase orbit lifetime
- Scenario 3: Earth Weakly Captured
  - Orbit is weakly captured at Earth, escapes and is then recaptured a year later
  - We don't know how to do this yet
- Scenario 4: Lunar Horseshoe
  - We enter an elliptical Earth orbit that is resonant with the Moon

Circulating Eccentric Orbit

The asteroid Cruithne is in a type of resonant orbit called a "Horseshoe orbit"

Current JPL primary ARM Earth return target:

- **DRO**
- **TBD altitude > 60,000 km**

Secondary JPL ARM Target:

- **Earth-Moon L2**

Reference: Strange, N., "Lunar Storage Orbits", JPL Presentation, 1/25/13

Trade Space – Performance

- **DRO Performance**
  - Parameters*:
    - Epoch range: 2021 - 2025
    - DRO Altitude
      - 60,000, 70,000, 80,000 km
      - Based on orbit lifetime. Shorter lifetimes not considered, currently.
    - Mission time
      - 84 total crew days available
      - Assess mission times for crews of 2, 3, 4
    - Stay time at the asteroid
    - Direct vs LGA, Outbound/Inbound
      - Current study: LGA outbound with Direct inbound
    - Earth return targets: Altitude/FPA vs Entry Target-line

* Parameter sets are similar for lunar orbit and libration point targets. For an EML2H target, the “altitude” variation can be substituted with an “amplitude” of the halo.

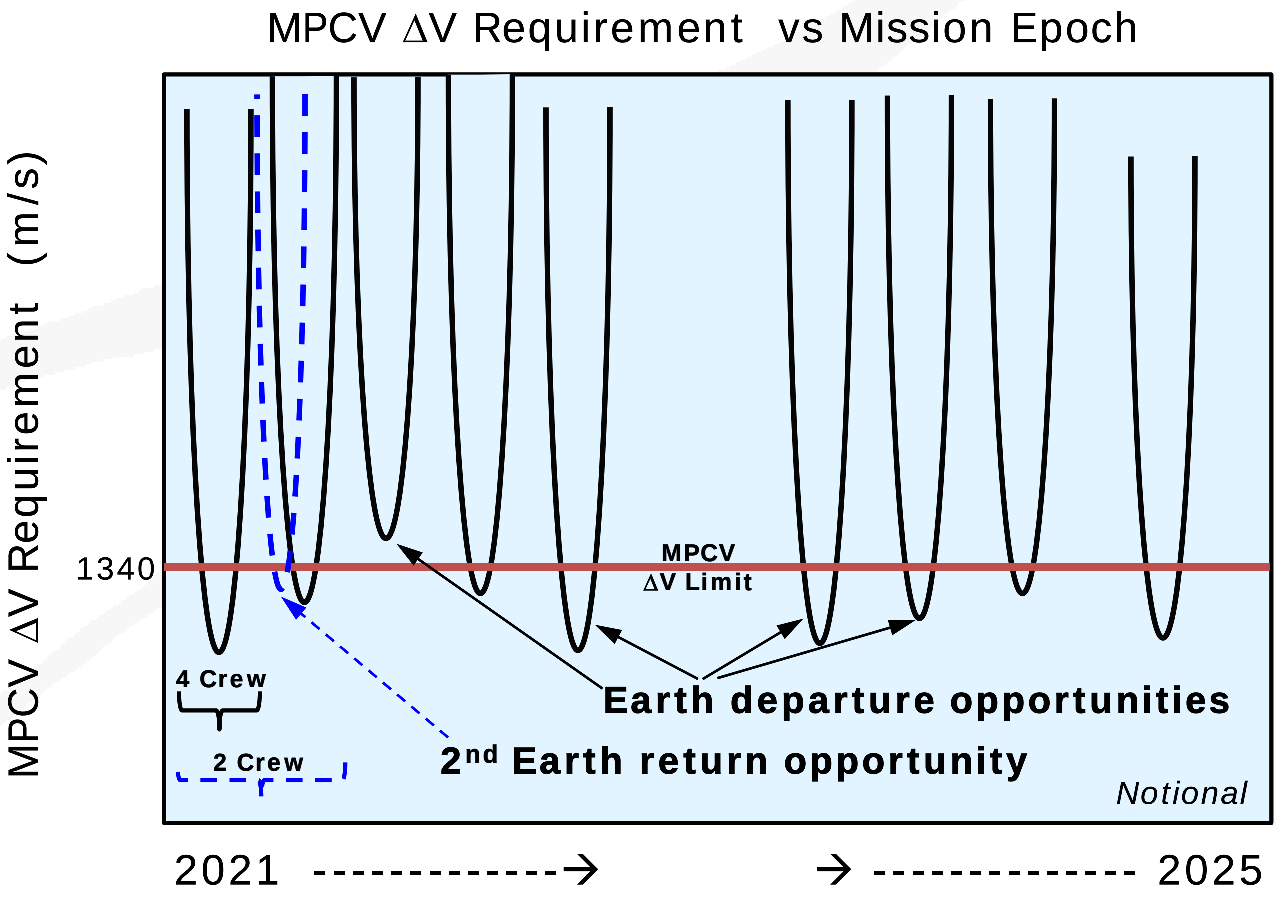
Trade Space – Performance

- **DRO Abort Performance**
  - Assessment of Earth return performance cost along outbound (Earth to Asteroid orbit) trajectory
    - Possibility of inclusion of a free-return on outbound
  - Assess aborts for Direct vs LGA (outbound and inbound) combinations

- **Assess MPCV mission opportunities**
  - Matching DRO orbit to resonate with lunar orbit
    - MPCV performance to DRO; Orbit lifetime

Trade Space – DRO Performance Output

- **Mission DV/propellant**
- **Frequency of opportunities**
- **Increased opportunities for reduced crew**
- **Launch windows**
- **Aborts**
- **Cargo missions?**



Future Work

- **Conduct MPCV Trades**
  - Performance cost for varying DRO characteristics
  - Mission opportunities to DRO
  - Mission duration vs. cost (for > 21 day max MPCV active life)
  - Launch window for DRO mission
- **Develop MPCV DRO rendezvous**
  - Far-field (and proximity operations)
- **Orbit lifetime**
- **Aborts**
- **Continue to assess MPCV performance to alternative asteroid storage orbits**

Acronyms

DRO	Distant Retrograde Orbits
FPA	Flight Path Angle
FPR	Flight Performance Reserve
iCPS	interim Cryogenic Propulsion Stage
LGA	Lunar Gravity Assist
MPCV	Multi-Purpose Crew Vehicle
OM	Orbit Maintenance
PRM	Perigee Raise Maneuver
TCM	Trajectory Correction Maneuver
SLS	Space Launch System

Backup

Round-Trip MPCV Mission To Libration Point (Example Cases)

1. EM-L2 Halo → EM-L2 Halo

MPCV

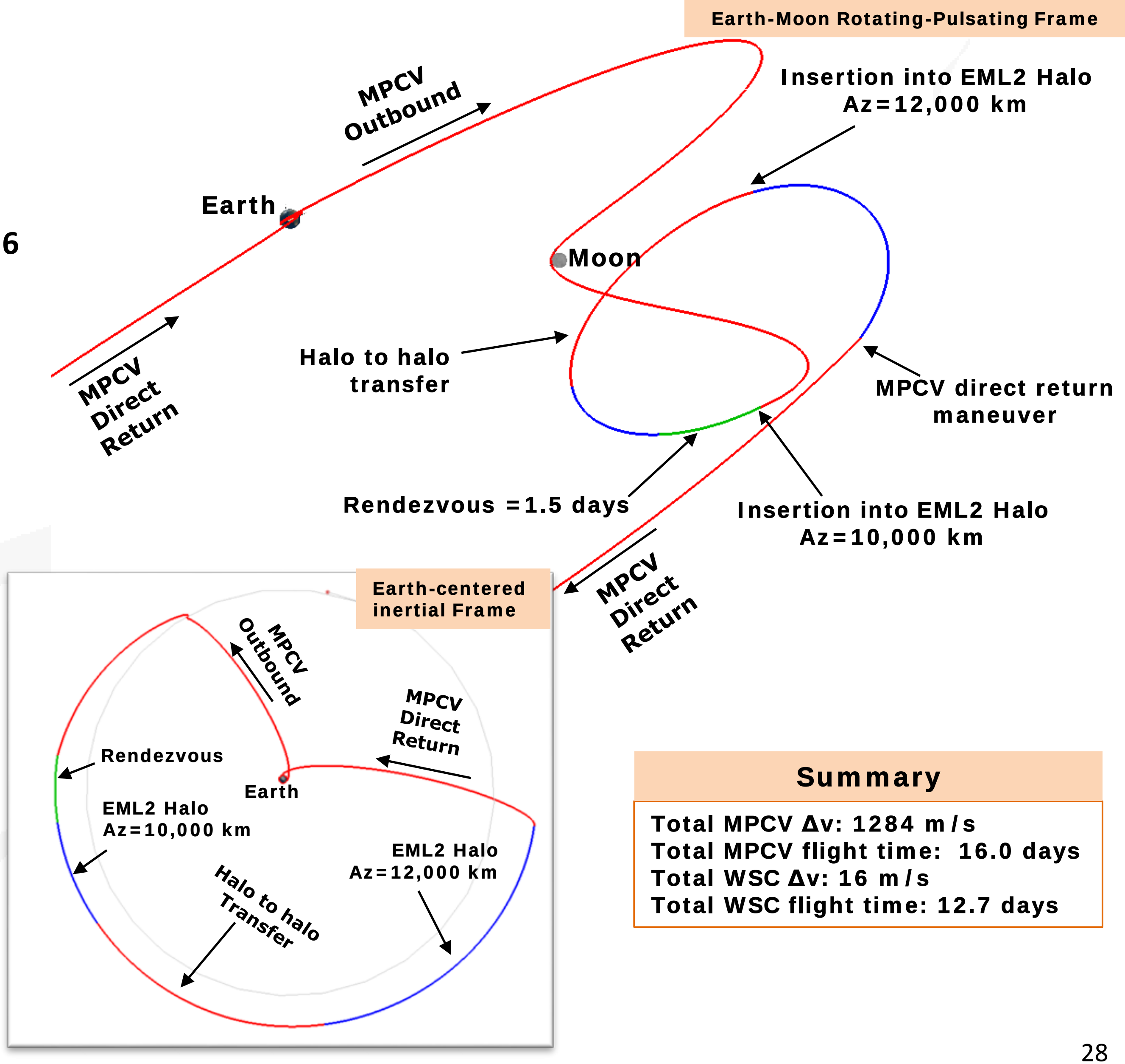
- LEO Orbit: 185x1806 km, Incl. = 28.5 deg
- Earth Departure Δv: 2741 m/s
- Earth to L2 Halo transfer time: 8.6 days
- Flyby Δv: 228 m/s
- L2 Halo insertion Δv: 112 m/s
- L2 halo orbit A_z: 10,000 km
- Rendezvous = 1.5 days

WSC+MPCV

- Stay in L2 Halo orbit: 2.3 days
- Transfer from L2 halo to L2 halo 4.7 days
- Transfer Δv: 16 m/s
- L2 halo orbit A_z: 12,000 km
- Stay in L2 Halo orbit: 5.7 days

MPCV

- Earth return Δv: 944 m/s
- Return time: 5.9 days (direct)



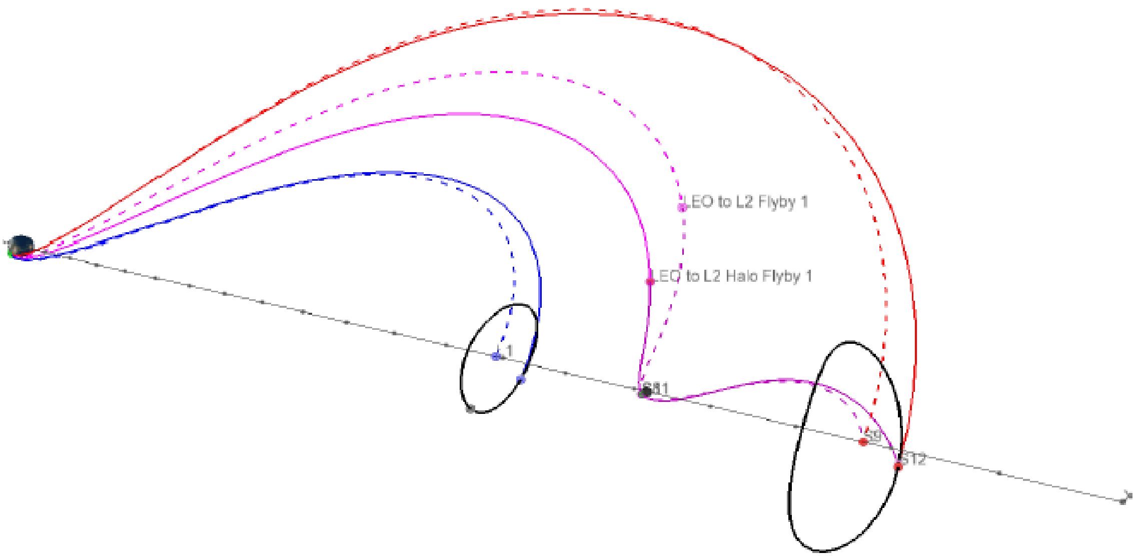
Summary

Total MPCV Δv: 1284 m/s
Total MPCV flight time: 16.0 days
Total WSC Δv: 16 m/s
Total WSC flight time: 12.7 days

Libration Point Orbit–MPCV Performance: L1, L2

Halo orbit test cases with Copernicus
Jacob Williams, ESCG, 11/4/2011

	LEO to L2 Halo		LEO to L2		LEO to L1 Halo	LEO to L1
Maneuver	**direct**	**flyby**	**direct**	**flyby**	**direct**	**direct**
LEO DV	3.172	3.155	3.171	3.156	3.140	3.137
Lunar Periapsis DV	0.000	0.175	0.000	0.186	0.000	0.000
Capture DV	0.958	0.109	1.108	0.148	0.606	0.717
Total DV (km/s)	4.130	3.438	4.278	3.490	3.746	3.855
Transfer Time (days)	6.29	8.35	6.13	8.53	3.94	3.76



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Assumptions

Circularized Moon Orbit at 2011-Jan-1 00:00:00
Objective function is sum of Delta-V's

LEO Departure Orbit

SMA	6478 km
ECC	0
INC	28.5 deg

L1 Halo orbit

Max. Amplitude in x-axis	6537 km
Max. Amplitude in y-axis	23445 km
Max. Amplitude in z-axis	10000 km
Orbit Period	12 day

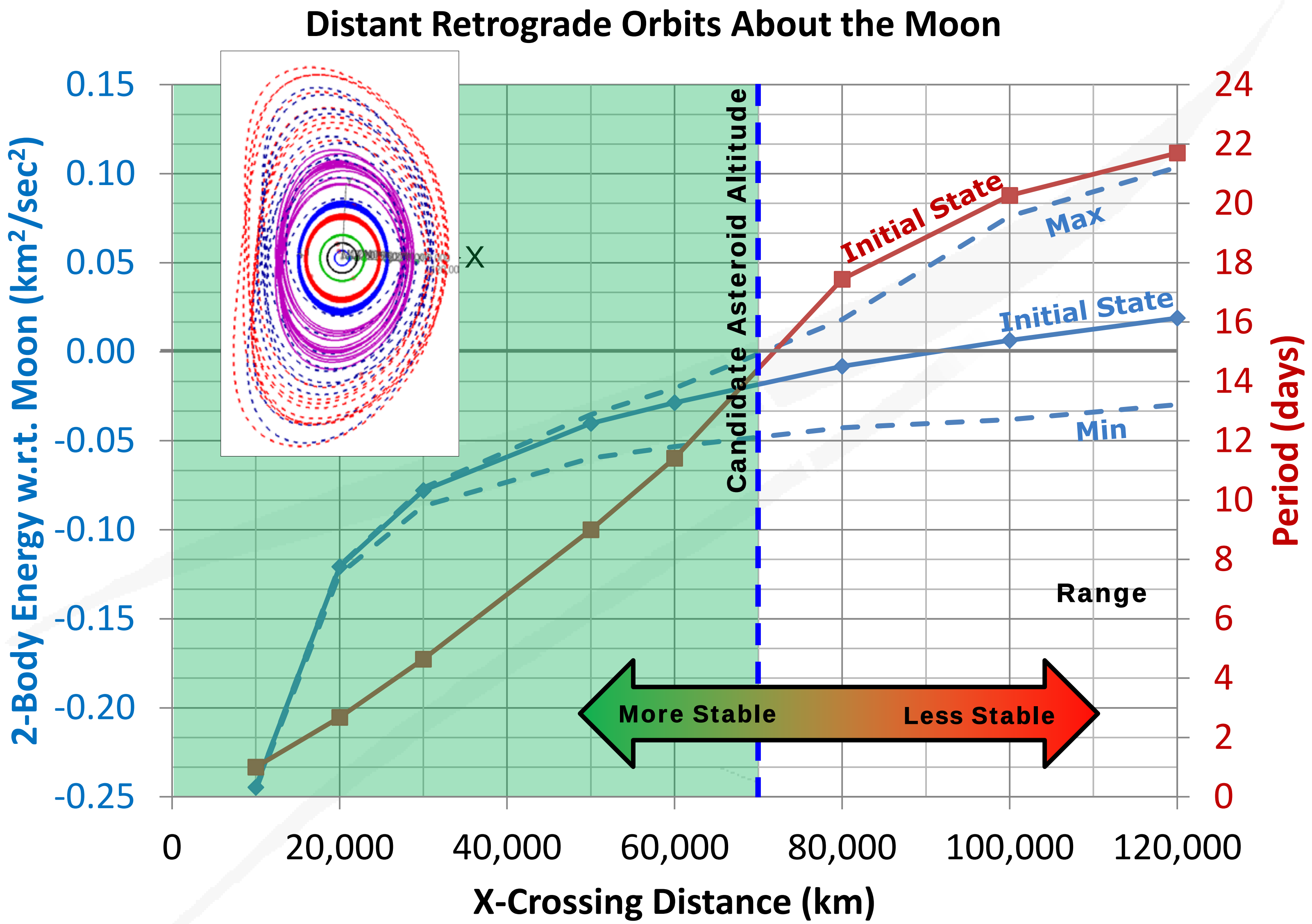
L2 Halo orbit

Max. Amplitude in x-axis	11904 km
Max. Amplitude in y-axis	34672 km
Max. Amplitude in z-axis	10000 km
Orbit Period	14 day

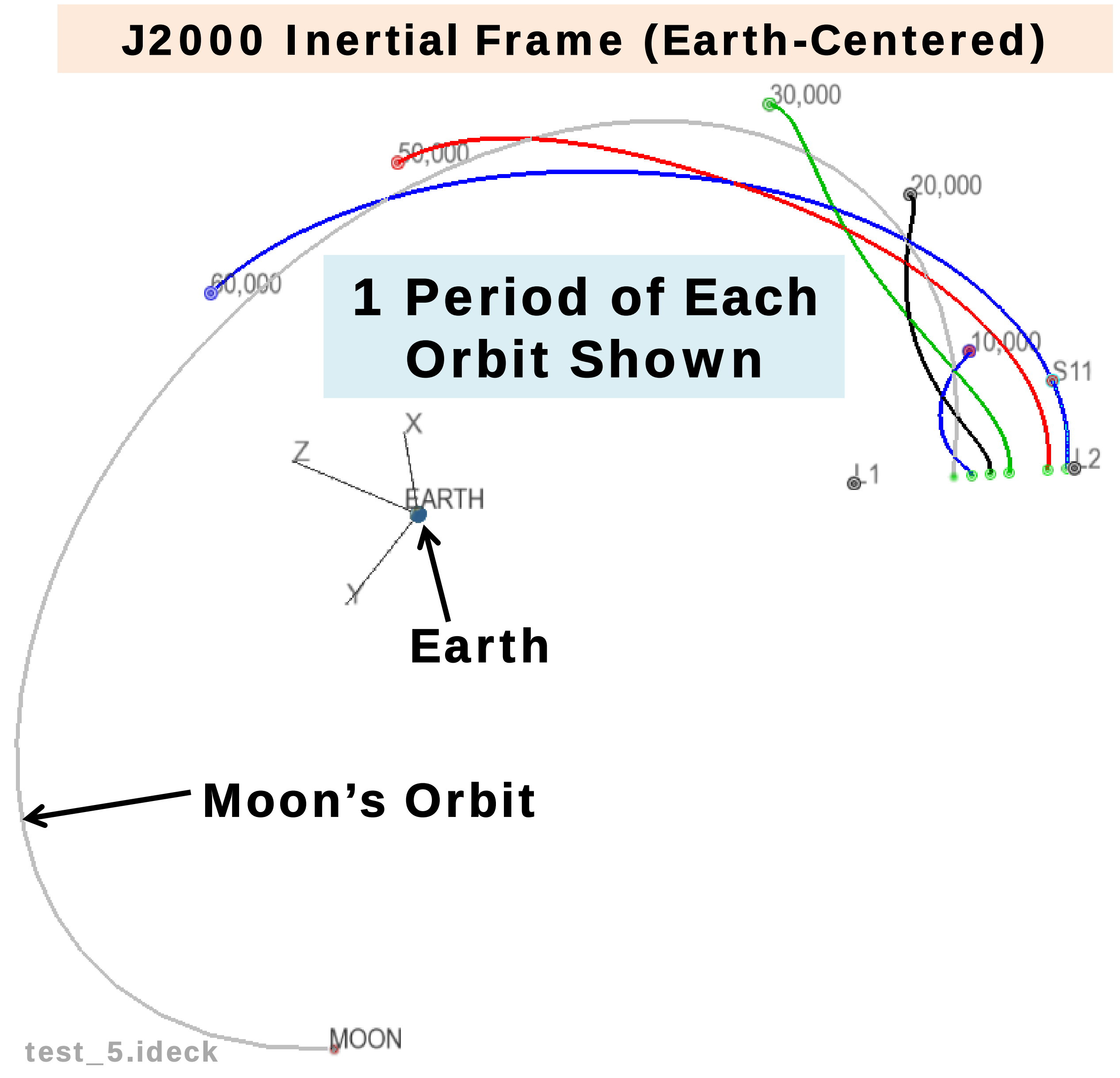
General Vehicle Assumptions

- **MPCV**
  - Mass = 24092.6 kg
  - Usable propellant (after removal of FPR, TCM, ACS, OMs, Sep. mnvrs) = 8086 kg
  - Isp = 315.1 sec
  - Thrust = 6,000 lb (26,689.3 newton)
  - Delta-V capability = 1340 m/s (usable, translational)
  - $T/W_{initial}$ = 0.113
- **iCPS (Current Configuration)**
  - $Mass_{MECO}$ = 55,773 kg (includes 24,092.6 kg MPCV)
  - $Mass_{Earth\_Departure}$ = 54,649.4 kg (includes 24,092.6 kg MPCV)
  - Usable $propellant_{Earth\_Departure+PRM}$ = 25,902.6 kg
  - Isp = 460.296 sec
  - $Thrust_{Earth\_Departure}$ = 110,897.4 N (24,930.7 lb.)
  - Earth Departure delta-V = 2859 m/s (2900 iCPS DV – 41 m/s PRM)
  - $T/W_{initial}$ = 0.207
- **iCPS (18” Extension - Stretched Configuration)**
  - $Mass_{MECO}$ = 58,313.3 kg (includes 24,092.6 kg MPCV)
  - $Mass_{Earth\_Departure+PRM}$ = 57,170.3 kg (includes 24,092.6 kg MPCV)
  - Isp = 462.746 sec
  - $Thrust_{Earth\_Departure}$ = 110,173.6 N (24,768 lb.)
  - Earth Departure delta-V =2890 m/s
  - $T/W_{initial}$ = 0.198

Two-Body Energy vs. DRO Size



Example DRO's: Inertial Frame (Earth-Centered)



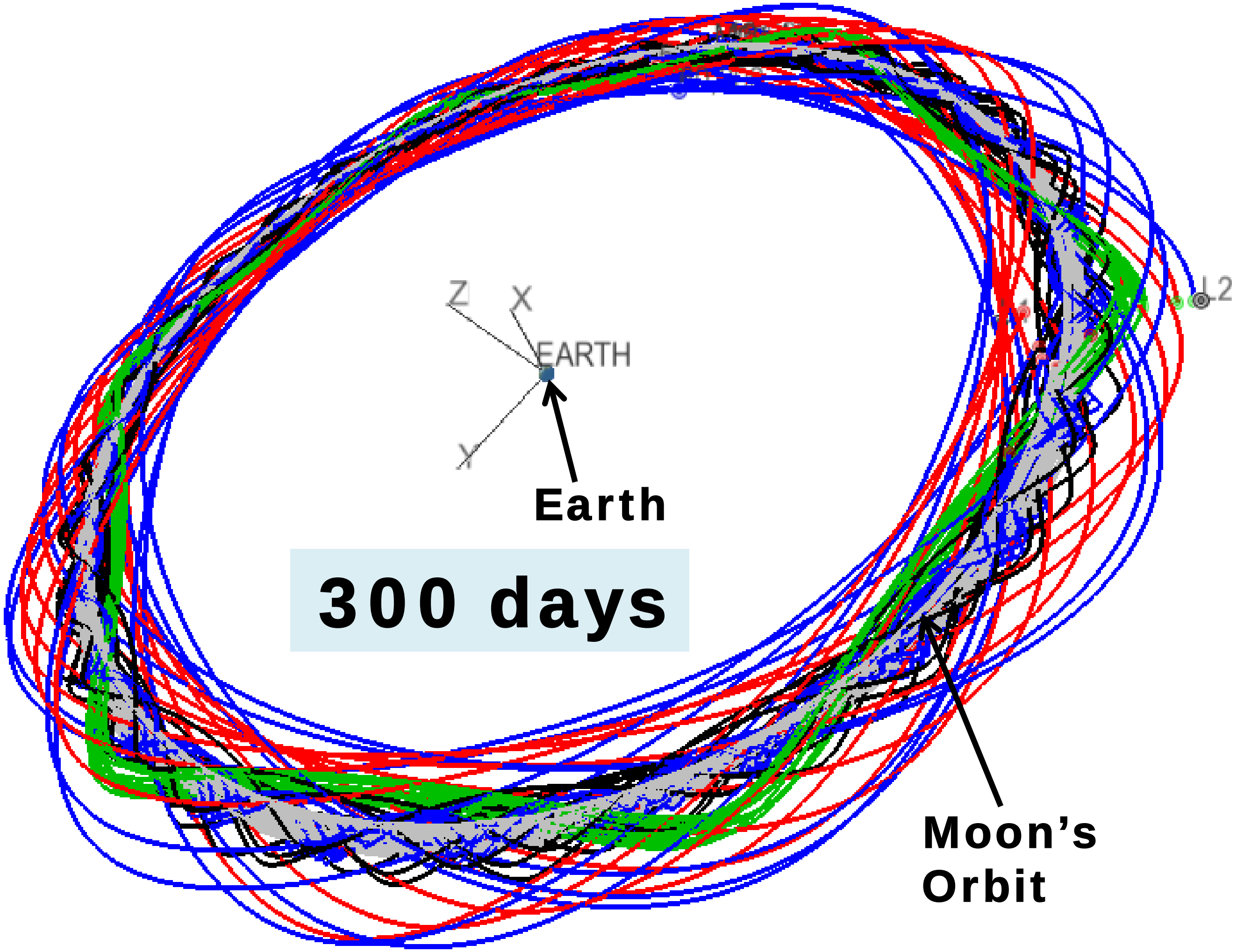
1 Period of Each Orbit Shown

Earth

Moon's Orbit

Example DRO's: Inertial Frame (Earth-Centered)

J2000 Inertial Frame (Earth-Centered)



Earth

300 days

Moon's Orbit

Example Mission – Performance Summary

Example cases only – not for vehicle sizing

- **Departure epoch: August 15, 2021**
- **DRO altitude = 60,000 km**
- **Halo orbit amplitudes are optimized**
- **Flight times are optimized**
- **LEO altitude = 185 x 185 km**
- **All impulsive solutions**

Mission Type	Approximate Departure Epoch	LEO Departure DV (m/s)	EML2 Departure DV (m/s)	Lunar Flyby DV (m/s)	DRO Arrival DV (m/s)	DRO Departure DV (m/s)	LLO Arrival DV (m/s)	LLO Departure DV (m/s)	EML2 Arrival DV (m/s)	Flight Time (days)	Comments
LEO to DRO Transfer (Direct)	August 15, 2021	3152			702					9.3	RAAN solution 1
LEO to DRO Transfer (Direct)	August 15, 2021	3161			801					11.8	RAAN solution 2
LEO to DRO Transfer (Flyby)	August 15, 2021	3134		182	167					8.1	
LEO to DRO Transfer (Flyby)	August 15, 2021	3134		188	171					8.0	
DRO to LLO Transfer (Direct)	August 15, 2021					108	653			5.5	
LLO to DRO Transfer (Direct)	August 15, 2021				107			653		5.5	
DRO to EML2 Halo Transfer (Direct)	August 15, 2021					19			358	5.3	Halo Az = 2,000 km. Opt. Halo alt., Opt. Flt. Time.
DRO to EML2 Halo Transfer (Flyby)	August 15, 2021			146		90			6.9	41.6	Halo Az = 2,000 km. Opt. Halo alt., Opt. Flt. Time.
EML2 Halo to DRO Transfer (Direct)	August 15, 2021		326		28					6.0	Halo Az = 2,000 km. Opt. Halo alt., Opt. Flt. Time.
EML2 Halo to DRO Transfer (Flyby)	August 15, 2021		14	41	112					21.5	Halo Az = 2,000 km. Opt. Halo alt., Opt. Flt. Time.
DRO to Earth EI (Direct)	August 15, 2021					631				5.7	Free Az EI
DRO to Earth EI (Flyby)	August 15, 2021			171		94				32.0	Free Az EI

DRO Mission Design and Performance

- **DROs are stable; No orbit maintenance required**
- **"Lunar" DROs cycle in the vicinity of the moon with a range of altitudes**
- **Visibility from Earth can be designed such that it doesn't cross disc of moon**
- **Possible short solar eclipsing**
- **DROs, being stable, do require a delta-V for insertion and departure (19 – 801 m/s in the examples provided). No manifold for insertion/departure.**
- **Current orbit maintenance delta-V budget for Gateway mission is 20 m/s/year**
  - Note: Artemis robotic mission in a Lissajous orbit used ~7 m/s/year
- **Performance (note: based on single cases, single epochs)**
  - Cost from LEO to DRO appears similar to LEO to EML2H w/ flyby ~350 m/s range
  - Cost from LEO to DRO appears a bit cheaper than LEO to EML2H direct
    - DRO: 801 m/s (direct); EML2H: 957 m/s (direct)
- **Higher cost to go to moon's orbit (parabolic approach vs. EM-L2)**
  - ~761 m/s DRO (~60,000 km) vs ~640 m/s EML2
    - DRO DVs: 108 m/s departure, 653 m/s LLO arrival
    - Note: DROs are stable, so will always have departure Δv, unlike EML2H

Discussion / Recommendations

- **Possible use as a "holding pen"**
  - For example: They could serve as a long term stable holding area for a returning Mars sample return (to address back contamination issues)
- **Recommendation: With a reasonably small orbit maintenance delta-V, there appears to be no significant benefit to DRO for Gateway type missions**
  - Further, the stability of the DROs can result in additional mission delta-V cost.

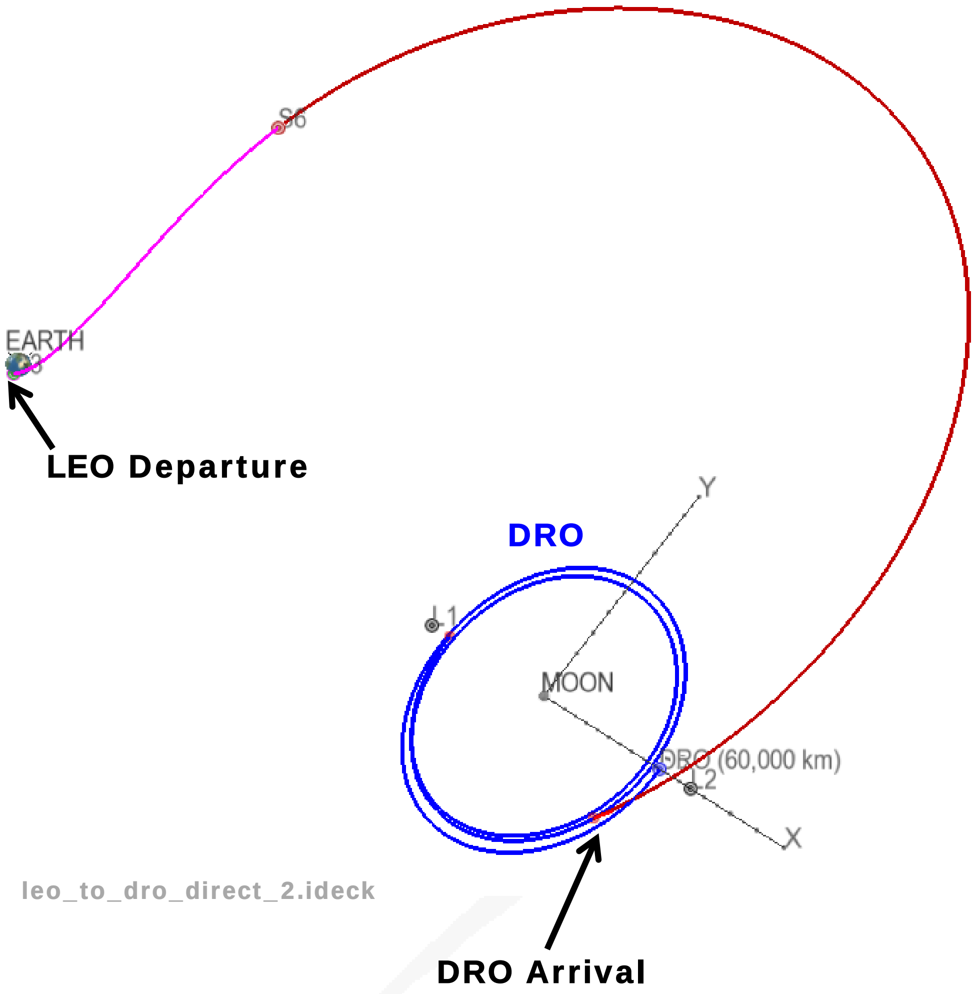
Example Trajectories

Example Transfers

- **A few example transfers to and from a DRO are shown here.**
- **Selected a 60,000 km DRO (which passes near both the Earth-Moon L1 and L2 libration points).**
- **Epoch is in the vicinity of August 15, 2021.**
- **Not meant to be a comprehensive performance study.**

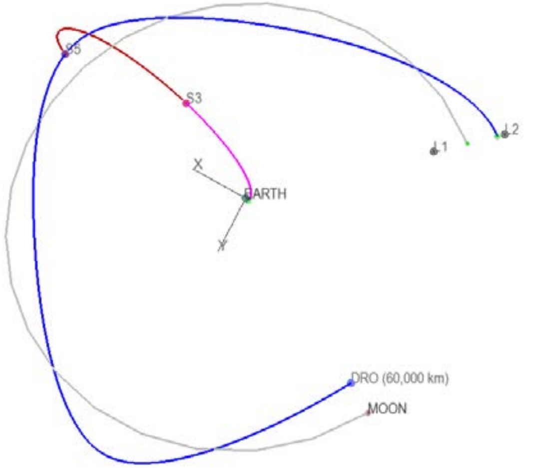
Example LEO to DRO Transfer (Direct)

Earth-Moon Rotating-Pulsating Frame



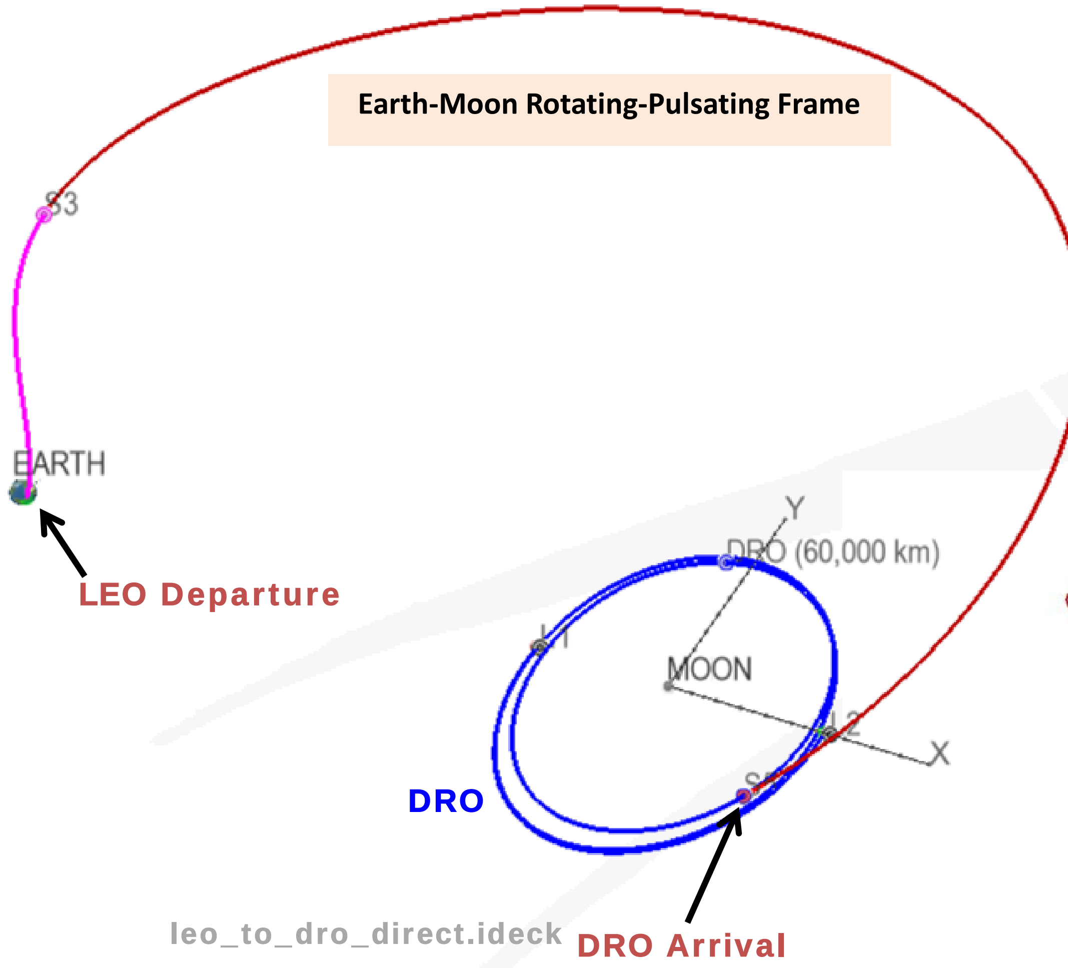
- b_{DRO} = 60,000 km
- LEO Departure epoch: Aug-15-2021
- LEO Departure orbit: 185 km, circular, 28.5 deg inclination (optimized RAAN and TA)
- Optimized flight times and maneuvers

- LEO Departure: 3,152 m/s
- DRO Arrival: 702 m/s
- Flight Time: 9.25 days



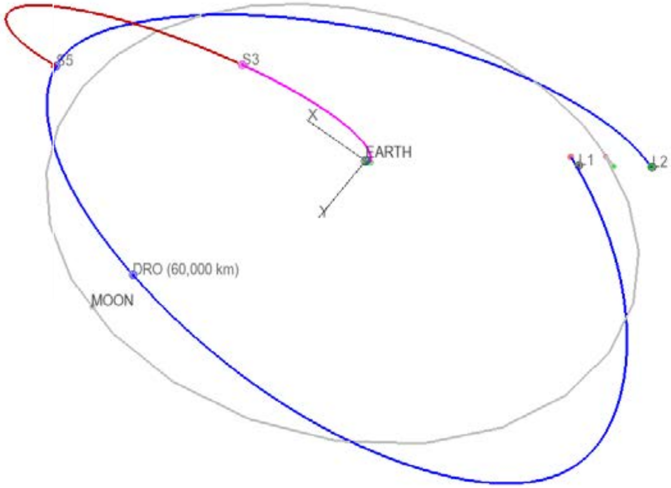
J2000 Inertial Frame

Example LEO to DRO Transfer (Direct)



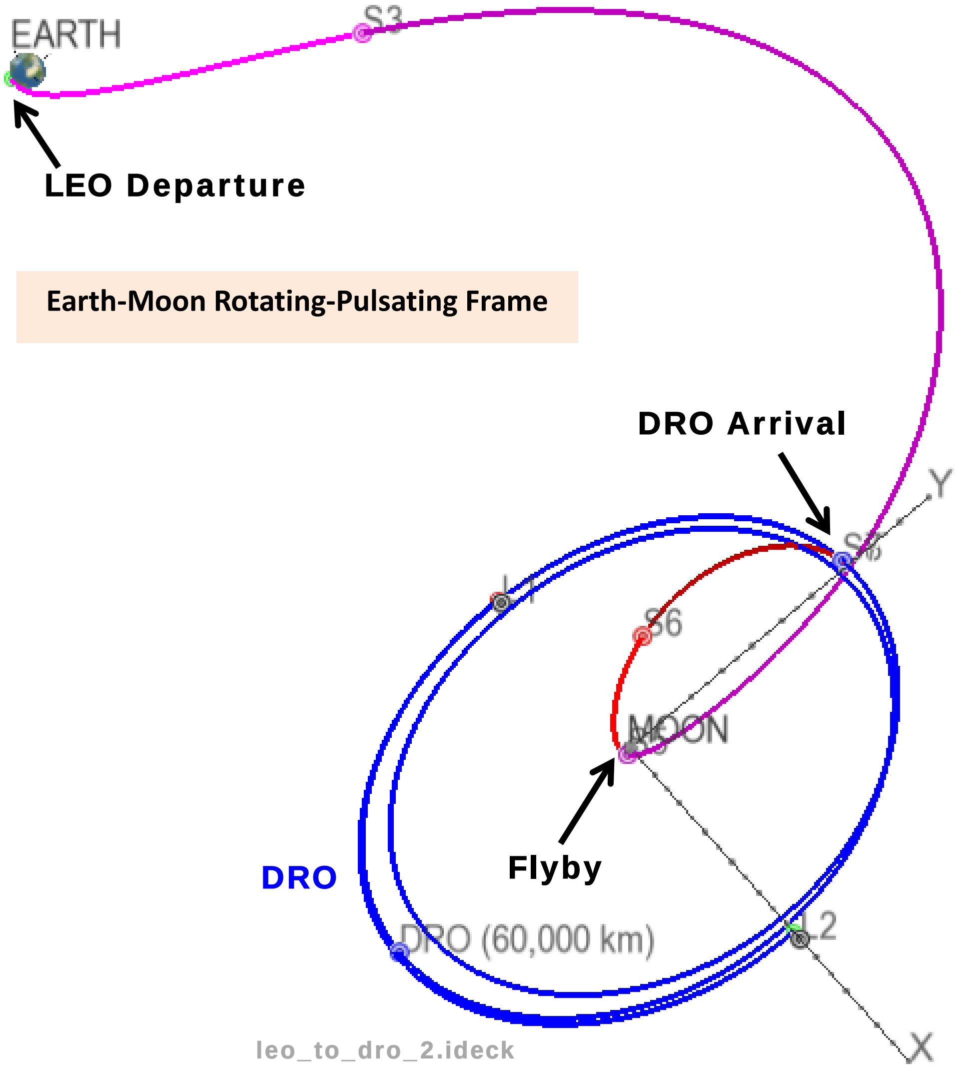
- b_{DRO} = 60,000 km
- LEO Departure epoch: Aug-15-2021
- LEO Departure orbit: 185 km, circular, 28.5 deg inclination (optimized RAAN and TA)
- Optimized flight times and maneuvers

- LEO Departure: 3,161 m/s
- DRO Arrival: 801 m/s
- Flight Time: 11.8 days



J2000 Inertial Frame

Example LEO to DRO Transfer (Flyby)



LEO Departure

DRO Arrival

Flyby

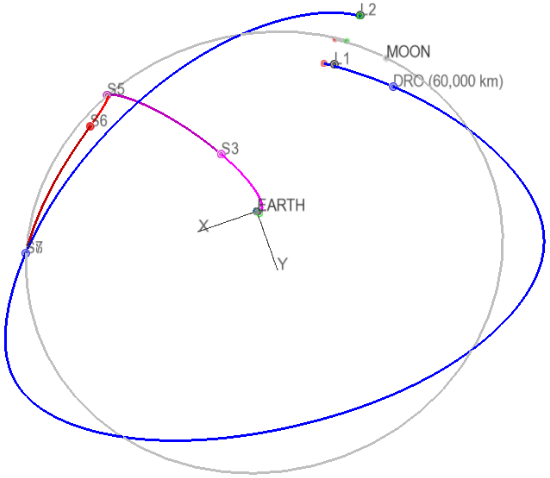
DRO

Earth-Moon Rotating-Pulsating Frame

- b_{DRO} = 60,000 km
- LEO Departure epoch: Aug-15-2021
- LEO Departure orbit: 185 km, circular, 28.5 deg inclination (optimized RAAN and TA)
- Powered lunar flyby (minimum 100 km altitude)
- Optimized flight times and maneuvers

- LEO Departure: 3,134 m/s
- Flyby : 182 m/s
- DRO Arrival: 167 m/s
- Flight Time: 8.05 days

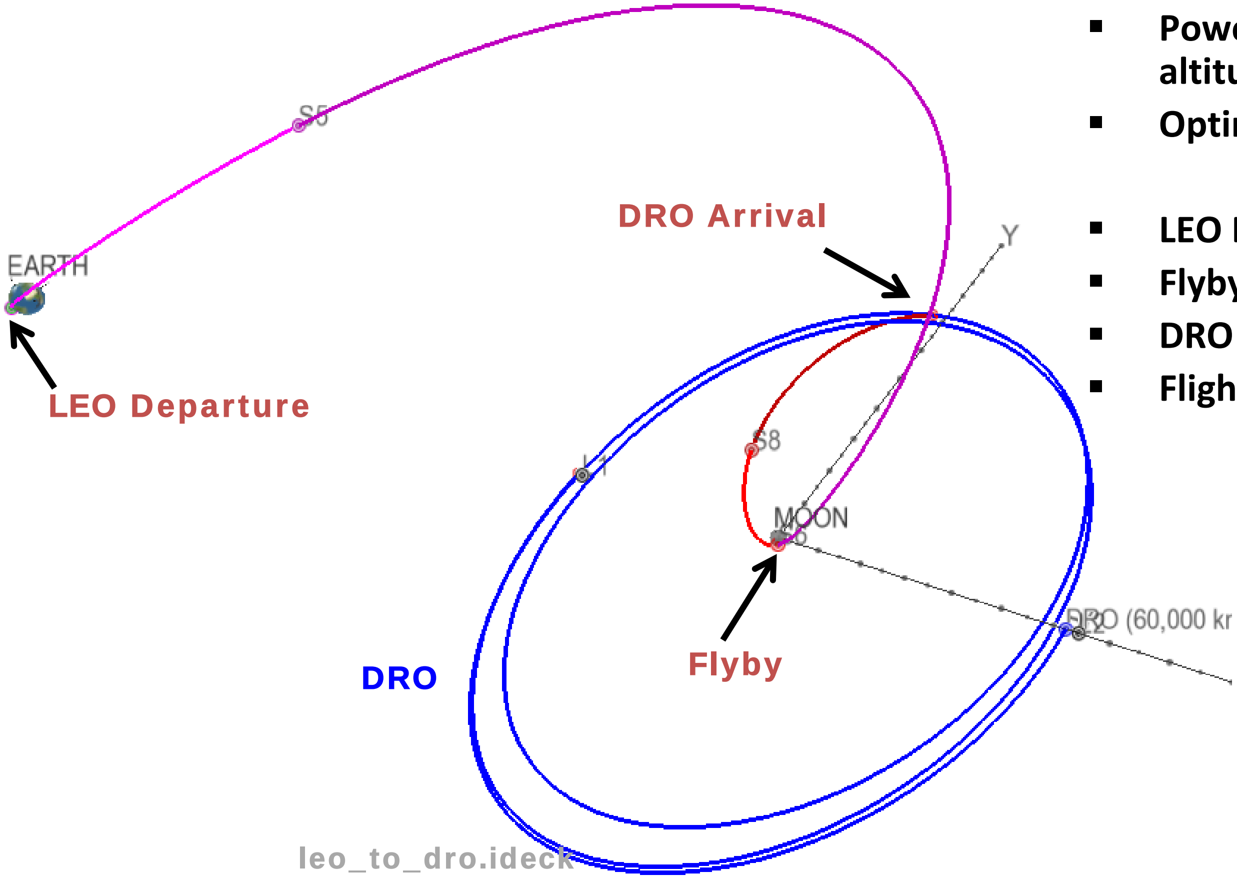
Total: 349 m/s

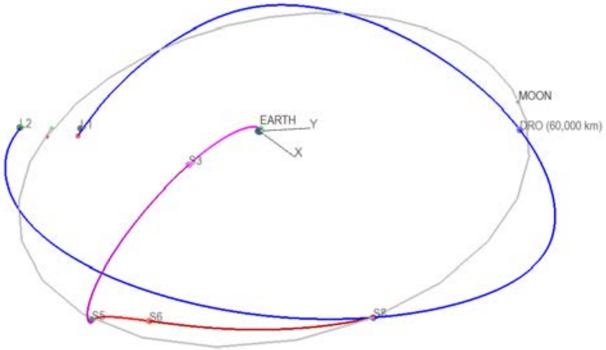


J2000 Inertial Frame

Example LEO to DRO Transfer (Flyby)

Earth-Moon Rotating-Pulsating Frame





J2000 Inertial Frame

- b_{DRO} = 60,000 km
- LEO Departure epoch: Aug-15-2021
- LEO Departure orbit: 185 km, circular, 28.5 deg inclination (optimized RAAN and TA)
- Powered lunar flyby (minimum 100 km altitude)
- Optimized flight times and maneuvers

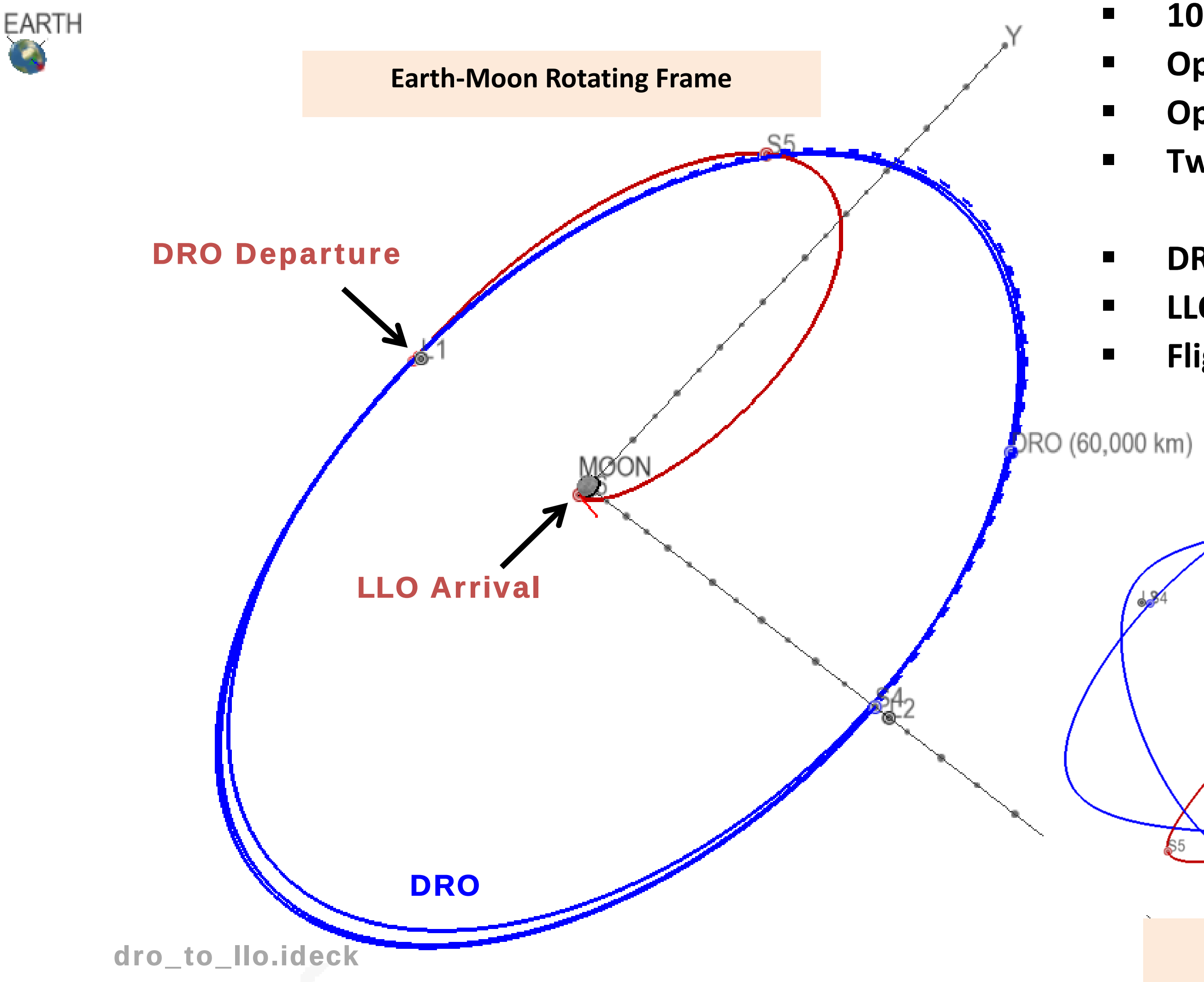
- LEO Departure: 3,134 m/s
- Flyby : 188 m/s
- DRO Arrival: 171 m/s
- Flight Time: 8.0 days

Total: 359 m/s

Example DRO to LLO Transfer

- b_{DRO} = 60,000 km
- LLO Arrival epoch: Aug-15-2021
- 100 km altitude circular LLO
- Optimized LLO Inc, RAAN, Ta
- Optimized flight time
- Two-impulse transfer

- DRO Departure: 108 m/s
- LLO Arrival: 653 m/s
- Flight Time: 5.52 days



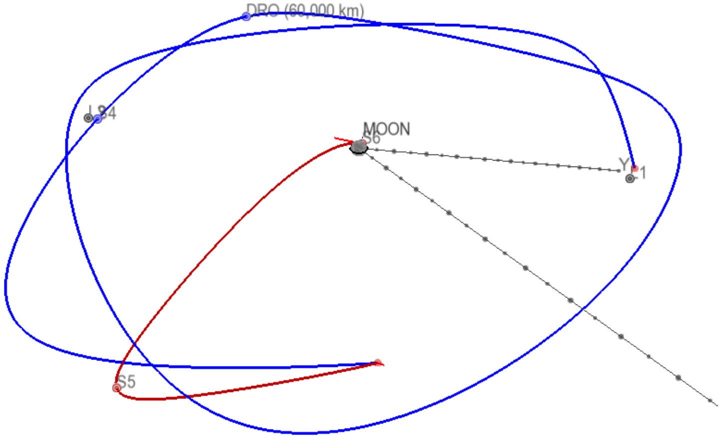
Earth-Moon Rotating Frame

DRO Departure

LLO Arrival

DRO

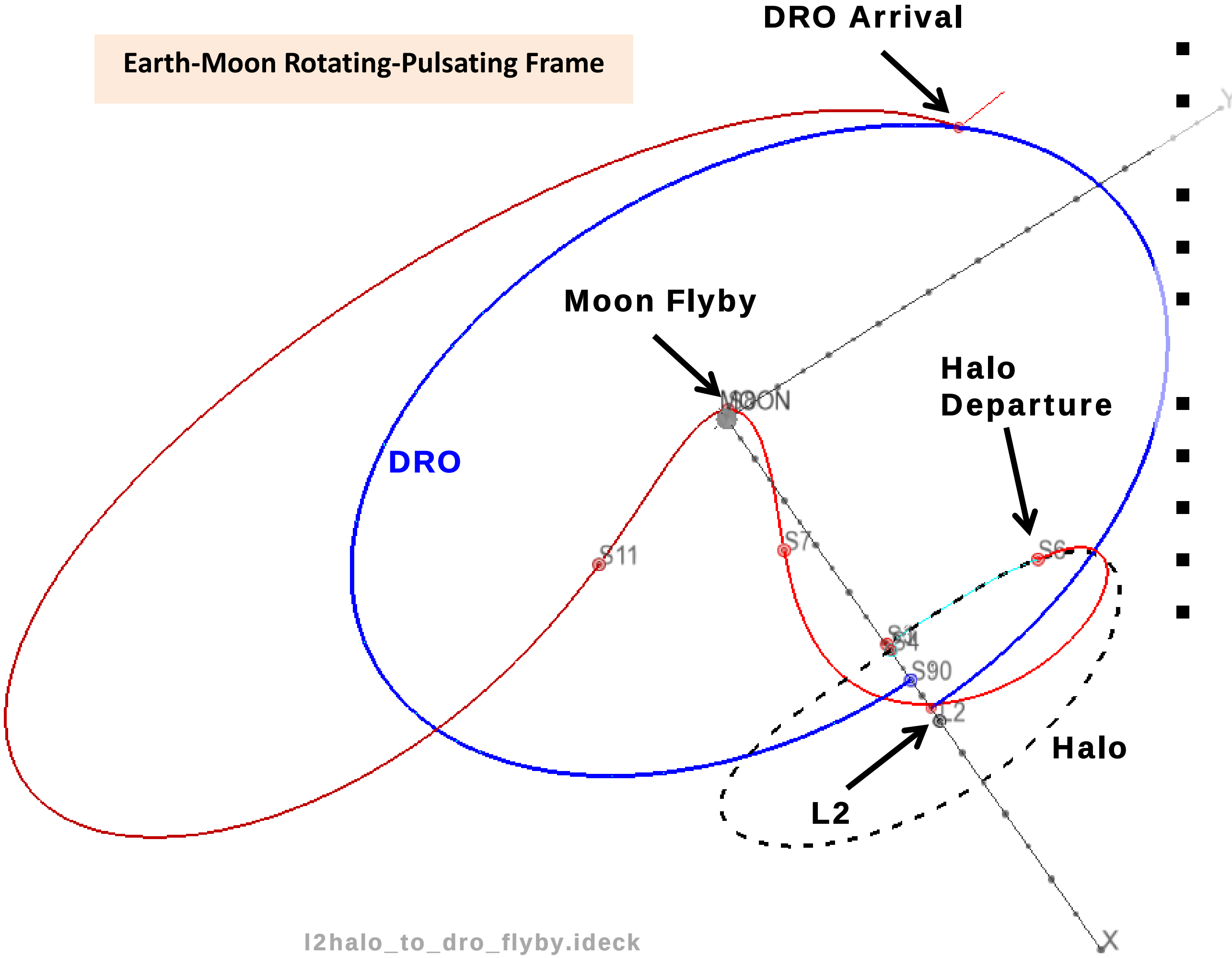
dro_to_llo.ideck



J2000 Inertial Frame

Example L2 Halo to DRO Transfer (Flyby)

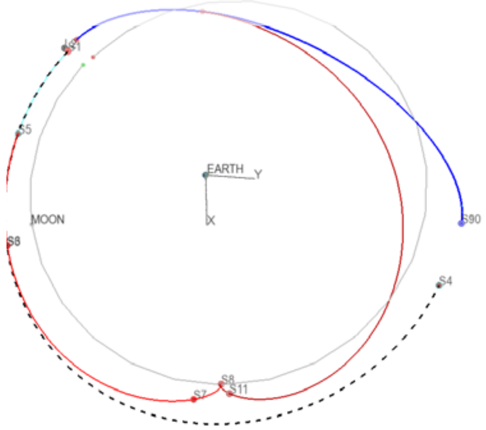
Earth-Moon Rotating-Pulsating Frame



- b_{DRO} = 60,000 km
- Halo departure epoch: Aug-15-2021
- Optimized flight time
- Optimized halo amplitude
- Three-impulse transfer

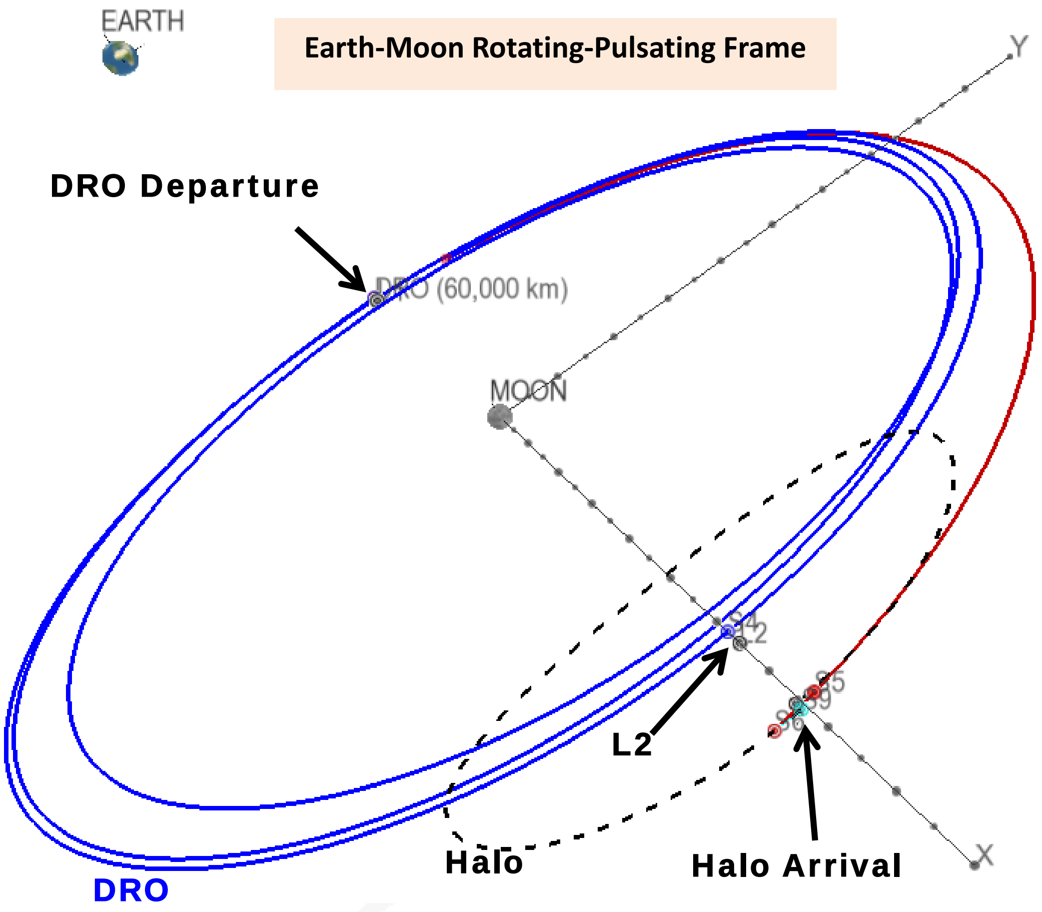
- Halo Departure: 14 m/s
- Flyby: 41 m/s
- DRO Arrival: 112 m/s
- Flight Time: 21.5 days
- Halo A_z = 2,000 km

Total: 168 m/s



J2000 Inertial Frame

Example DRO to L2 Halo Transfer (Direct)



Earth-Moon Rotating-Pulsating Frame

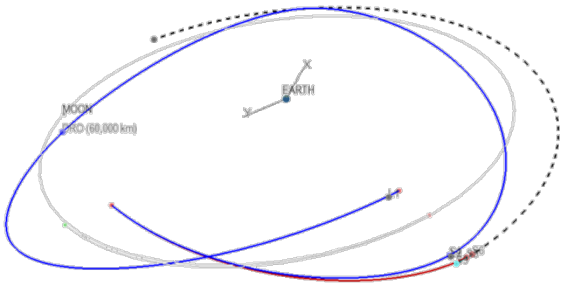
dro_to_l2halo.ideck

- b_{DRO} = 60,000 km
- Halo arrival epoch: Aug-15-2021
- Two impulse transfer
- Optimized halo amplitude
- Optimized flight times and maneuvers

- DRO Departure: 19 m/s
- Halo Arrival: 358 m/s

Total: 377 m/s

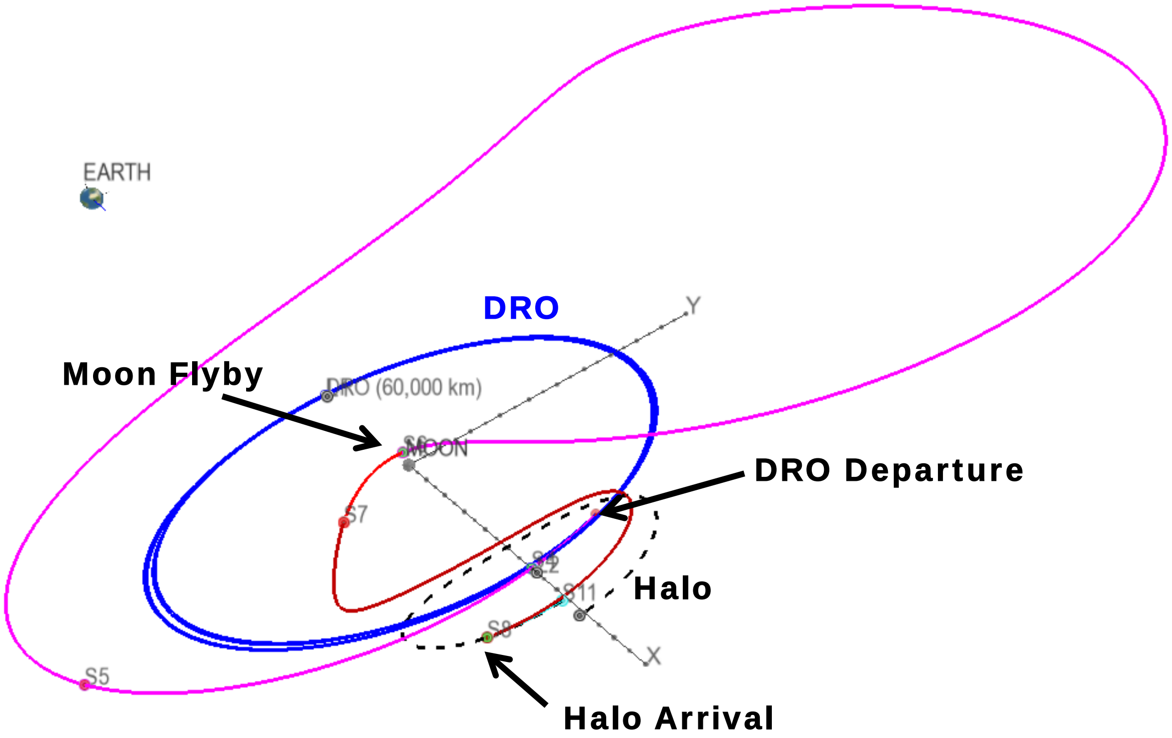
- Flight Time: 5.3 days
- Halo A_z = 2,000 km



J2000 Inertial Frame

Example DRO to L2 Halo Transfer (Flyby)

Earth-Moon Rotating Frame

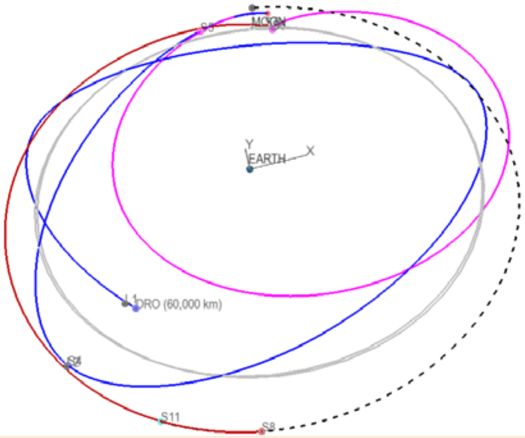


dro_to_l2halo_flyby_2.ideck

- b_{DRO} = 60,000 km
- DRO departure epoch: ~ Aug-15-2021
- Optimized flight time
- Optimized halo amplitude
- Three-impulse transfer

- DRO Departure: 90 m/s
- Flyby: 146 m/s
- Halo Arrival: 6.9 m/s
- Flight Time: 41.6 days
- Halo A_z = 2,000 km

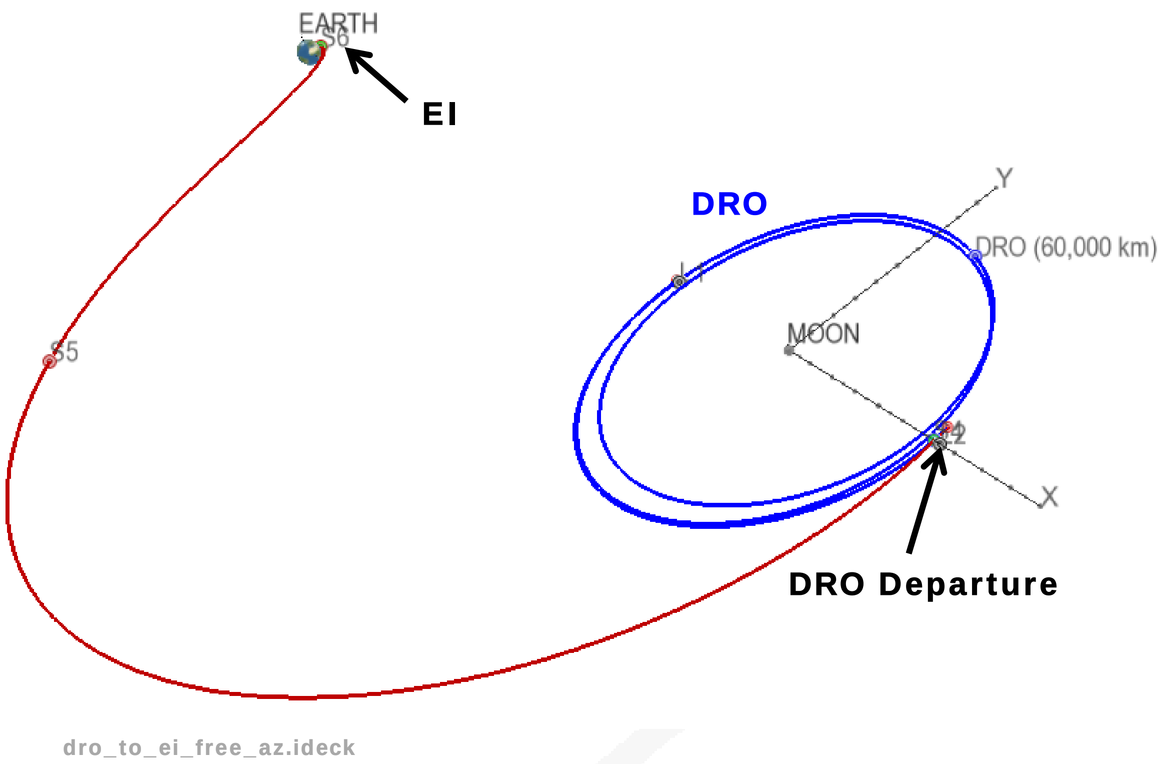
Total: 244 m/s



J2000 Inertial Frame

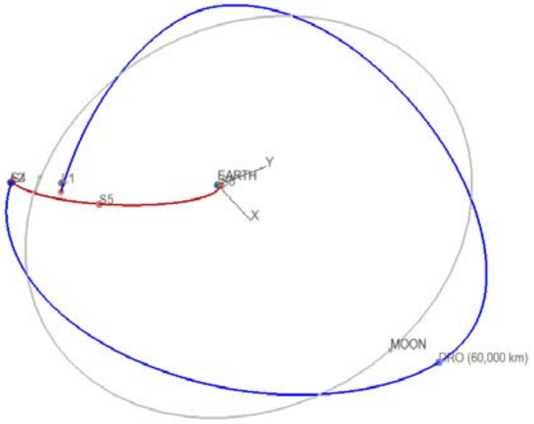
Example DRO to EI (Direct)

Earth-Moon Rotating-Pulsating Frame



- b_{DRO} = 60,000 km
- DRO departure epoch: ~ Aug-15-2021
- EI Altitude: 121.9 km
- EI FPA: -5.86 deg

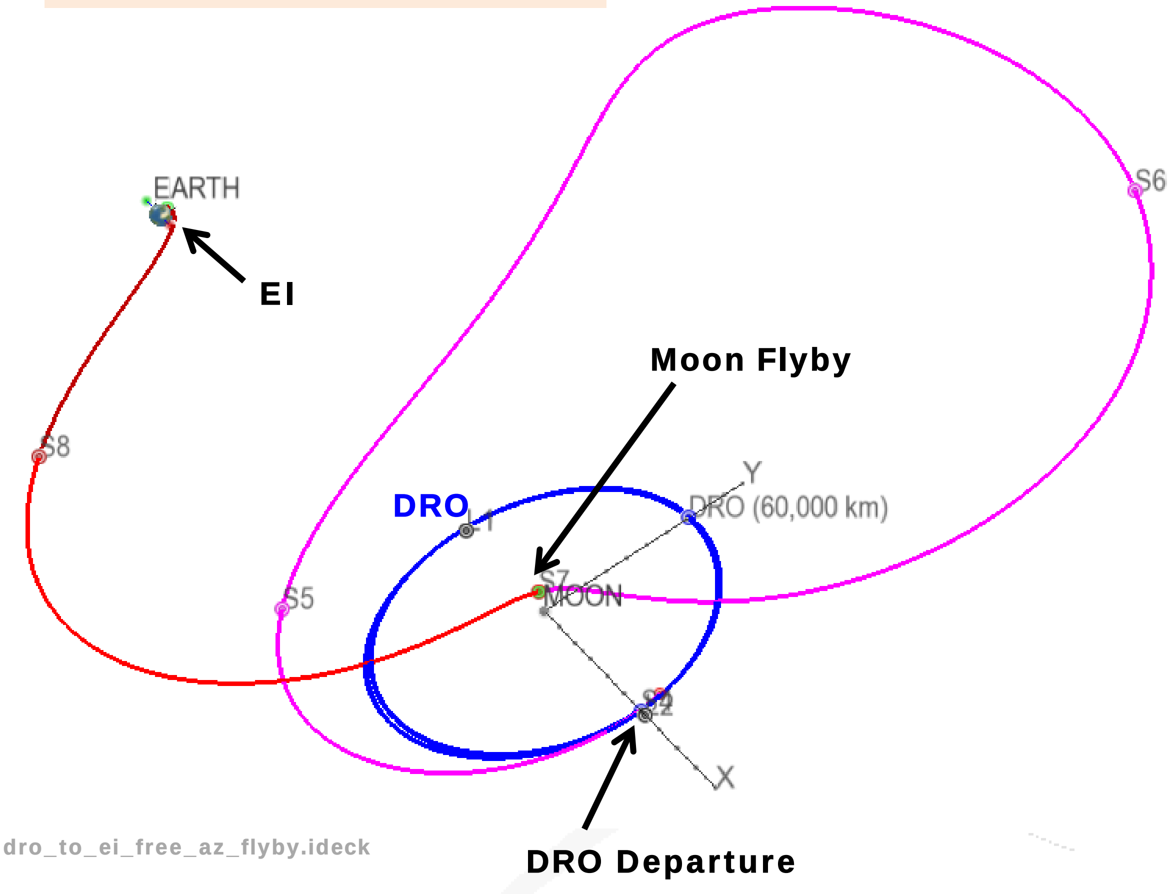
- DRO Departure: 631 m/s
- Flight Time: 5.7 days
- Entry velocity: 10.99 km/s



J2000 Inertial Frame

Example DRO to EI (Flyby)

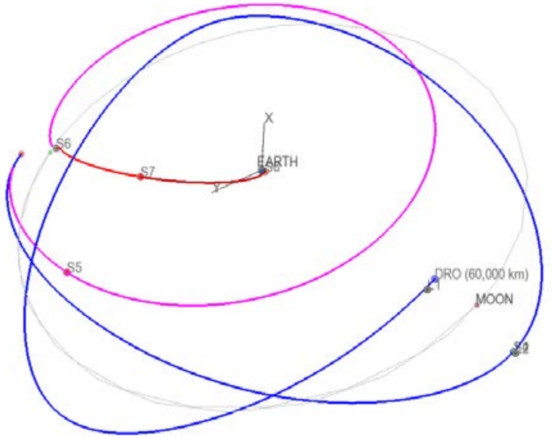
Earth-Moon Rotating Frame



- b_{DRO} = 60,000 km
- DRO departure epoch: ~ Aug-15-2021
- EI Altitude: 121.9 km
- EI FPA: -5.86 deg

- DRO Departure: 94 m/s
- Moon Flyby: 171 m/s
- Flight Time: 32 days
- Entry velocity: 10.98 km/s

Total: 265 m/s



J2000 Inertial Frame

Families of Periodic Orbits

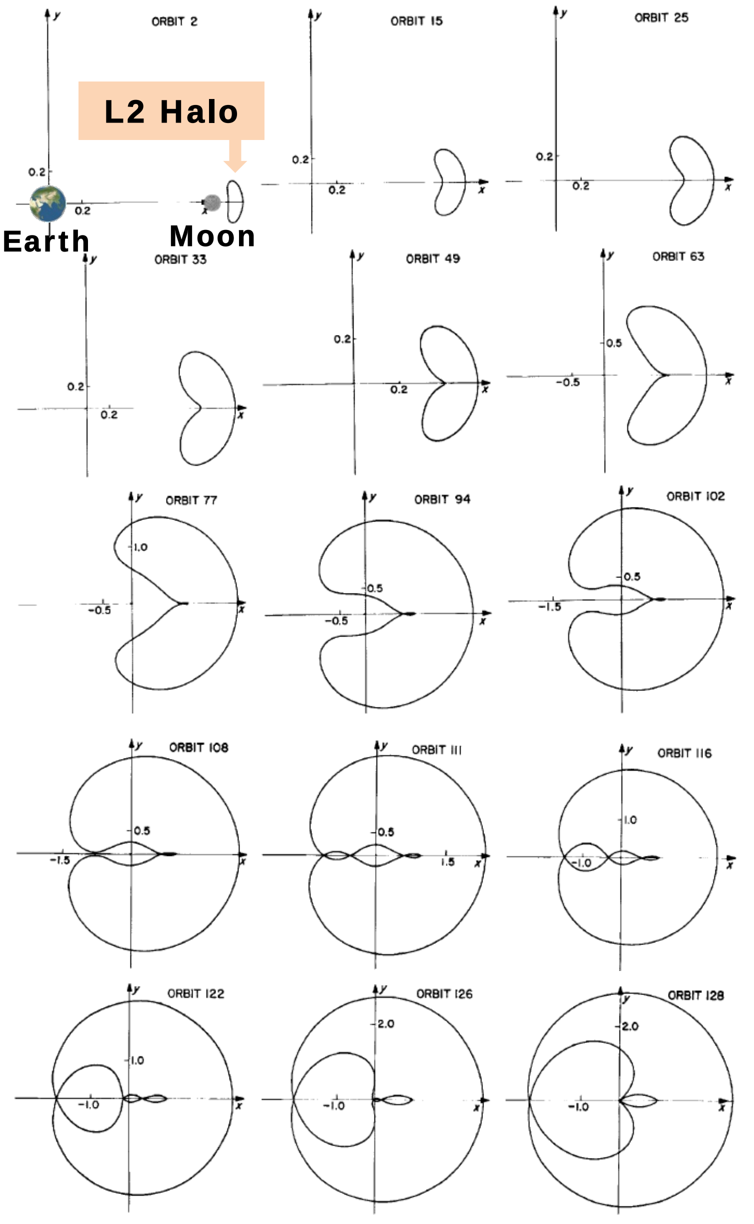


Fig. 11 (contd)

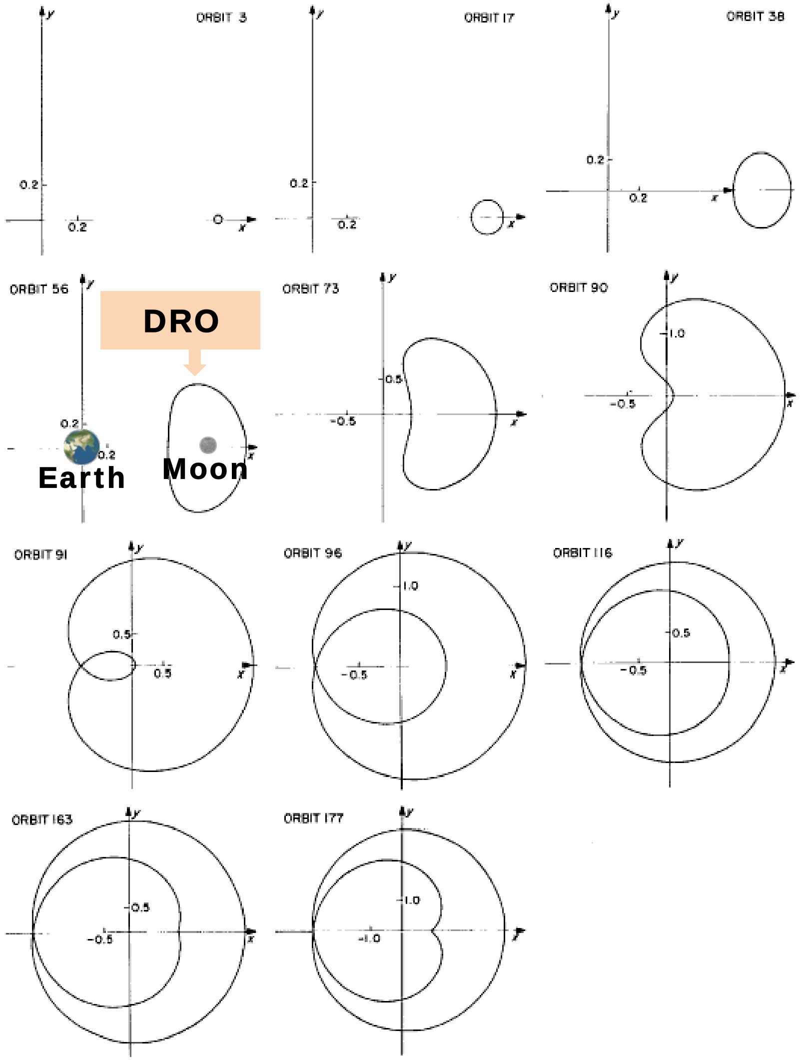
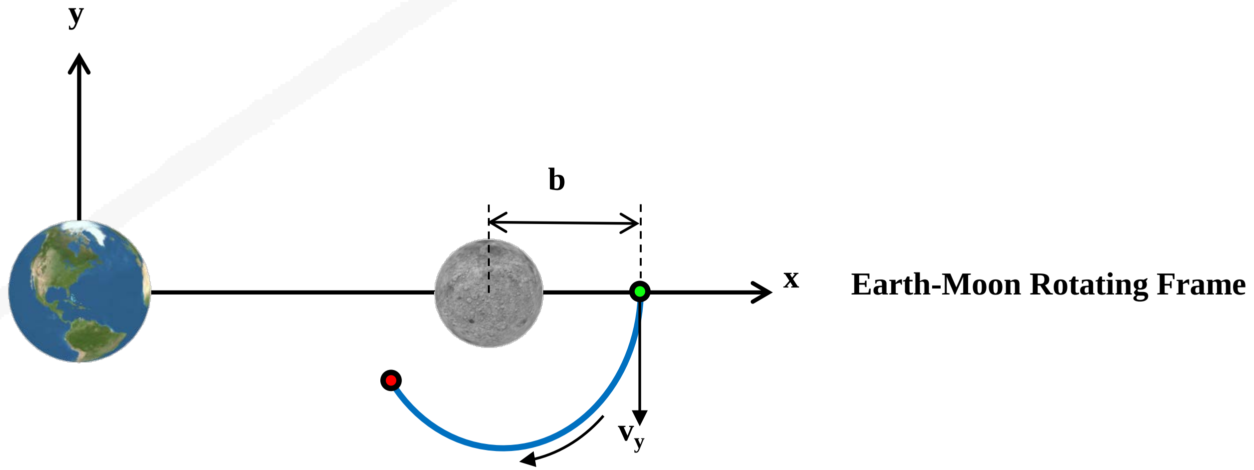


Fig. 30. Typical trajectories in family C of retrograde periodic orbits around m_2

From: **R.A. Broucke, "Periodic Orbits in the Restricted Three-Body Problem with Earth-Moon Masses", JPL Technical Report 32-1168, 1968.**

Computing DRO’s

- **Only considering planar-DRO’s here (in the Earth-Moon plane)**
- **Using Copernicus (latest development build)**
  - Force Model: Earth, Moon, Sun
  - SNOPT optimizer
  - DDEABM (Adams) integration method (1e-11 tolerance)
- **Optimization Problem**
  - Optimization variables: Δt and v_y
  - r_x coordinate specified (“semiminor axis” b)
  - Target v_x=0 at next x-axis crossing (r_y=0) [repeat for a couple periods]



References

- R.A. Broucke, “Periodic Orbits in the Restricted Three-Body Problem with Earth-Moon Masses”, JPL Technical Report 32-1168, 1968.
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LUNAR MISSION TUTORIAL

Part 1 – Lunar Orbit Mechanics

Developed by EG/Aeroscience and Flight Mechanics Division
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Introduction

- On January 14, 2004, President Bush announced a new vision for NASA
  - *Extend human presence across the solar system, starting with a human return to the Moon by the year 2020, in preparation for human exploration of Mars and other destinations;*
- Key Elements of New Space Policy
  - *Begin robotic missions to the Moon by 2008, followed by a period of evaluating lunar resources and technologies for exploration.*
  - *Begin human expeditions to the Moon in the 2015 – 2020 timeframe.*

Approach

This presentation provides a tutorial of lunar astrodynamic characteristics

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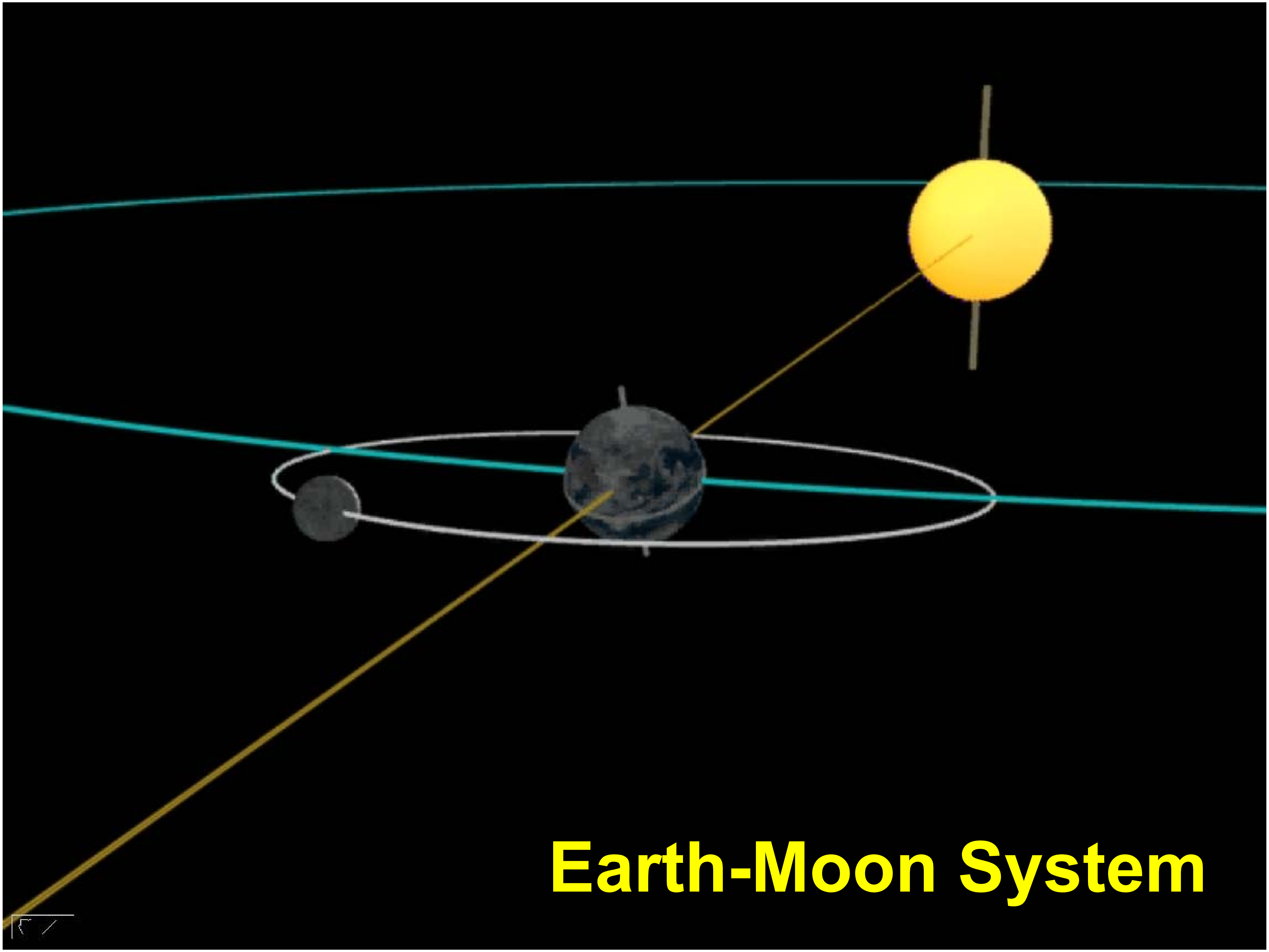
It addresses orbital mechanics as it applies to a human lunar mission design

Outline: Part I - Lunar Orbit Mechanics

❑ Earth-Moon System

- Lunar Inclination
- Lunar Libration
- Earth to Moon (Outbound)
  - Geocentric Characteristics
  - Selenocentric Characteristics
- Lunar Orbit
- Moon to Earth (Inbound)
- Libration Points
- Environment

Earth-Moon System

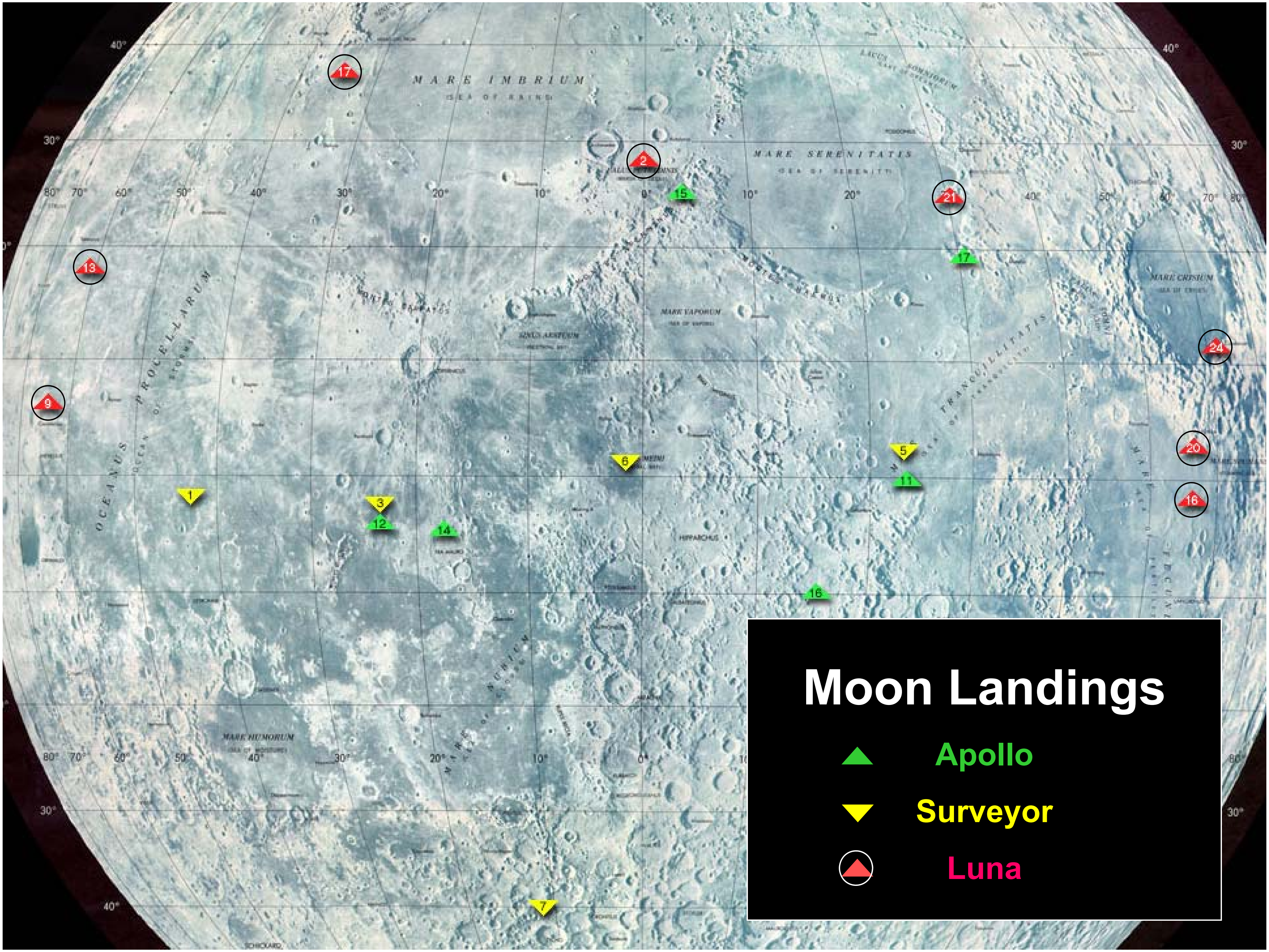


Moon and Earth Facts

Comparison	Moon	Earth	% of Earth
Mass (kg)	7.3483×10^{22}	5.9742×10^{24}	1.23
Volume (km^3)	2.1958×10^{10}	1.0832×10^{12}	2.03
Equatorial radius (km)	1738.1	6378.1	27.25
Polar radius (km)	1736.0	6356.8	27.31
Ellipticity (Flattening)	0.0012	0.00335	36.0
Mean density (kg/m^3)	3350	5515	60.7
Surface gravity (m/s^2)	1.62	9.80	16.5
Escape velocity (km/s)	2.38	11.2	21.3
Gravitational Parameter (km^3/s^2)	4.902×10^3	3.986×10^5	1.23
J2 (effects of nonspherical/homogenous body)	202.7×10^{-6}	1082.63×10^{-6}	18.7

Moon Facts

Parameter	Moon
Semimajor axis (km)	384,400
Perigee (km)	363,300
Apogee (km)	405,500
Revolution Period (days)	27.3217
Synodic Period (days)	29.53
Mean Orbital Velocity (km/s)	1.023
Max. Orbital Velocity (km/s)	1.076
Min. Orbital Velocity (km/s)	0.964
Inclination to Ecliptic (deg)	5.145
Inclination to Equator (deg)	18.28-28.58
Orbit Eccentricity	0.0549
Sidereal Rotation Period (days)	27.32
Obliquity to orbit (deg)	6.68
Recession rate from Earth (cm/yr)	3.8
Mean Values	
Distance from Earth (km)	384,467
Apparent diameter (seconds of arc)	1,864.2
Apparent visual magnitude	-12.74



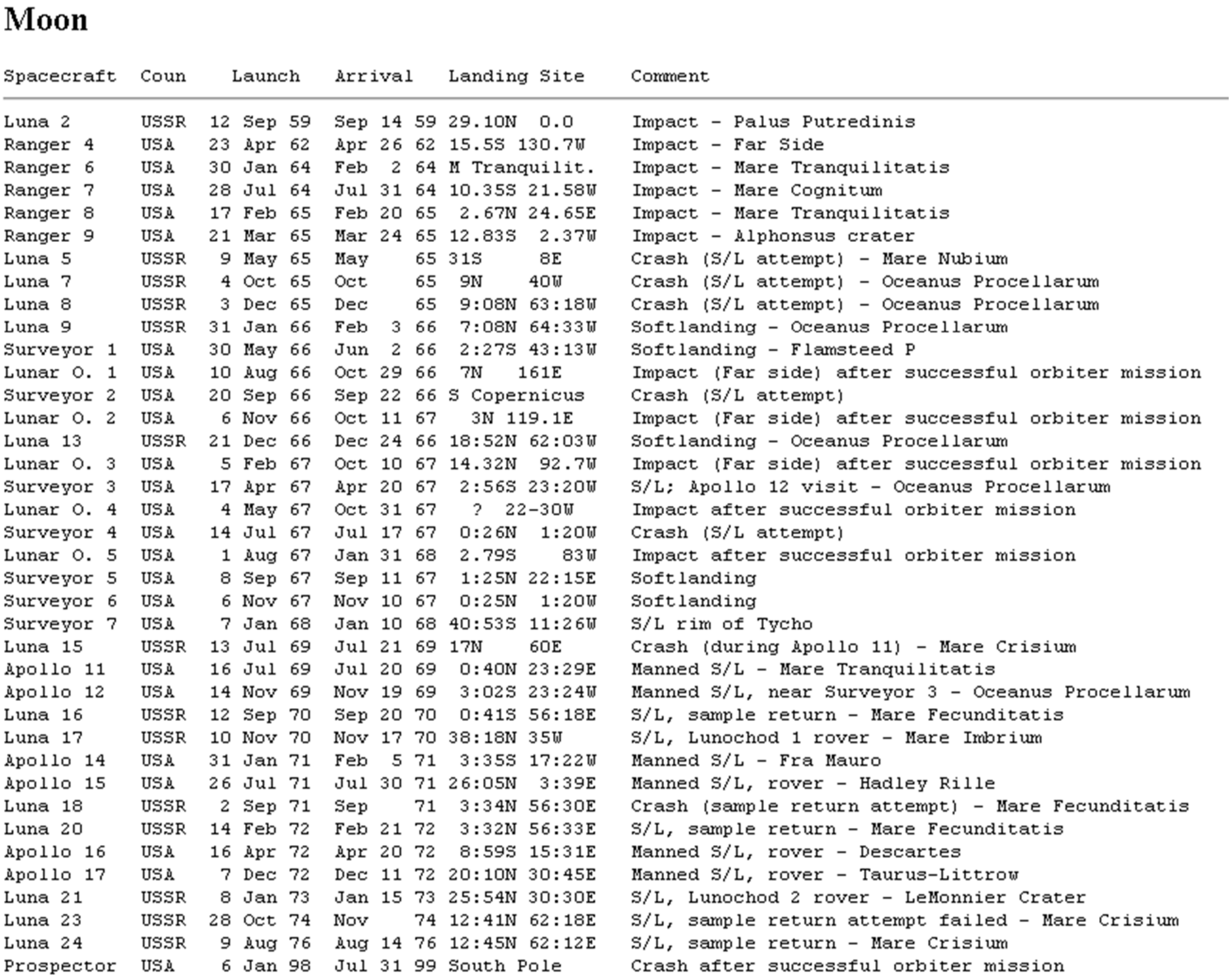
Moon Landings

▲ Apollo

▼ Surveyor

🔺 Luna

Lunar Mission History



Moon

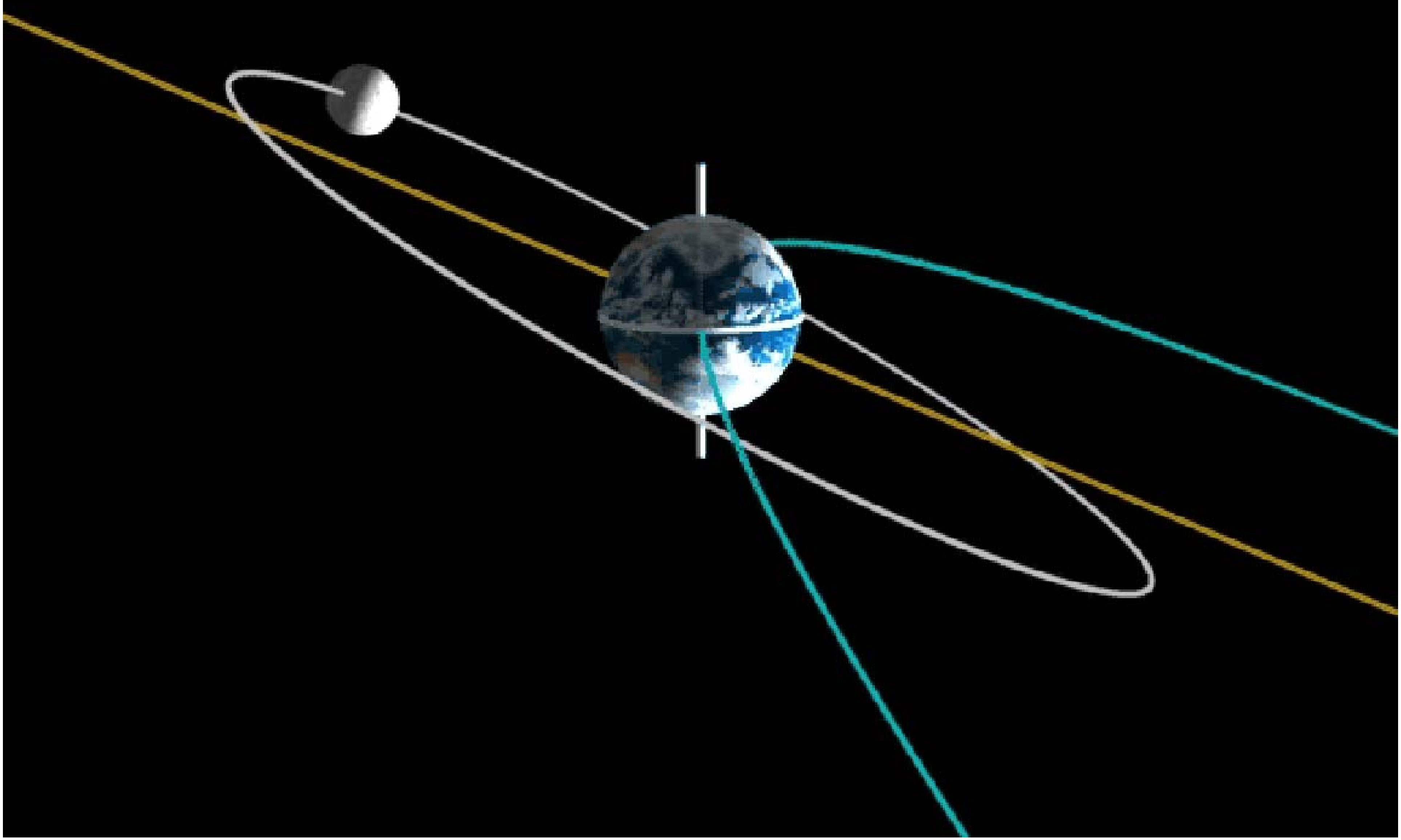
Spacecraft	Coun	Launch	Arrival	Landing Site	Comment
Luna 2	USSR	12 Sep 59	Sep 14 59	29.10N 0.0	Impact - Palus Putredinis
Ranger 4	USA	23 Apr 62	Apr 26 62	15.5S 130.7W	Impact - Far Side
Ranger 6	USA	30 Jan 64	Feb 2 64	M Tranquilit.	Impact - Mare Tranquilitatis
Ranger 7	USA	28 Jul 64	Jul 31 64	10.35S 21.58W	Impact - Mare Cognitum
Ranger 8	USA	17 Feb 65	Feb 20 65	2.67N 24.65E	Impact - Mare Tranquilitatis
Ranger 9	USA	21 Mar 65	Mar 24 65	12.83S 2.37W	Impact - Alphonsus crater
Luna 5	USSR	9 May 65	May 65	31S 8E	Crash (S/L attempt) - Mare Nubium
Luna 7	USSR	4 Oct 65	Oct 65	9N 40W	Crash (S/L attempt) - Oceanus Procellarum
Luna 8	USSR	3 Dec 65	Dec 65	9:08N 63:18W	Crash (S/L attempt) - Oceanus Procellarum
Luna 9	USSR	31 Jan 66	Feb 3 66	7:08N 64:33W	Softlanding - Oceanus Procellarum
Surveyor 1	USA	30 May 66	Jun 2 66	2:27S 43:13W	Softlanding - Flamsteed P
Lunar O. 1	USA	10 Aug 66	Oct 29 66	7N 161E	Impact (Far side) after successful orbiter mission
Surveyor 2	USA	20 Sep 66	Sep 22 66	S Copernicus	Crash (S/L attempt)
Lunar O. 2	USA	6 Nov 66	Oct 11 67	3N 119.1E	Impact (Far side) after successful orbiter mission
Luna 13	USSR	21 Dec 66	Dec 24 66	18:52N 62:03W	Softlanding - Oceanus Procellarum
Lunar O. 3	USA	5 Feb 67	Oct 10 67	14.32N 92.7W	Impact (Far side) after successful orbiter mission
Surveyor 3	USA	17 Apr 67	Apr 20 67	2:56S 23:20W	S/L; Apollo 12 visit - Oceanus Procellarum
Lunar O. 4	USA	4 May 67	Oct 31 67	? 22-30W	Impact after successful orbiter mission
Surveyor 4	USA	14 Jul 67	Jul 17 67	0:26N 1:20W	Crash (S/L attempt)
Lunar O. 5	USA	1 Aug 67	Jan 31 68	2.79S 83W	Impact after successful orbiter mission
Surveyor 5	USA	8 Sep 67	Sep 11 67	1:25N 22:15E	Softlanding
Surveyor 6	USA	6 Nov 67	Nov 10 67	0:25N 1:20W	Softlanding
Surveyor 7	USA	7 Jan 68	Jan 10 68	40:53S 11:26W	S/L rim of Tycho
Luna 15	USSR	13 Jul 69	Jul 21 69	17N 60E	Crash (during Apollo 11) - Mare Crisium
Apollo 11	USA	16 Jul 69	Jul 20 69	0:40N 23:29E	Manned S/L - Mare Tranquilitatis
Apollo 12	USA	14 Nov 69	Nov 19 69	3:02S 23:24W	Manned S/L, near Surveyor 3 - Oceanus Procellarum
Luna 16	USSR	12 Sep 70	Sep 20 70	0:41S 56:18E	S/L, sample return - Mare Fecunditatis
Luna 17	USSR	10 Nov 70	Nov 17 70	38:18N 35W	S/L, Lunochod 1 rover - Mare Imbrium
Apollo 14	USA	31 Jan 71	Feb 5 71	3:35S 17:22W	Manned S/L - Fra Mauro
Apollo 15	USA	26 Jul 71	Jul 30 71	26:05N 3:39E	Manned S/L, rover - Hadley Rille
Luna 18	USSR	2 Sep 71	Sep 71	3:34N 56:30E	Crash (sample return attempt) - Mare Fecunditatis
Luna 20	USSR	14 Feb 72	Feb 21 72	3:32N 56:33E	S/L, sample return - Mare Fecunditatis
Apollo 16	USA	16 Apr 72	Apr 20 72	8:59S 15:31E	Manned S/L, rover - Descartes
Apollo 17	USA	7 Dec 72	Dec 11 72	20:10N 30:45E	Manned S/L, rover - Taurus-Littrow
Luna 21	USSR	8 Jan 73	Jan 15 73	25:54N 30:30E	S/L, Lunochod 2 rover - LeMonnier Crater
Luna 23	USSR	28 Oct 74	Nov 74	12:41N 62:18E	S/L, sample return attempt failed - Mare Crisium
Luna 24	USSR	9 Aug 76	Aug 14 76	12:45N 62:12E	S/L, sample return - Mare Crisium
Prospector	USA	6 Jan 98	Jul 31 99	South Pole	Crash after successful orbiter mission

Tidal Locking

- The Moon pulls on Earth with a force that varies from point to point, thereby causing tidal bulges that follow the terrestrial sublunar point and its antipode. In the very long term, the friction caused by the attendant flow of ocean water slows the Earth’s rotation rate.

- Similarly, Earth pulling on the Moon over many millennia caused a tidal bulge that has become frozen in place while concurrently putting the Moon into a state of gravity-gradient stabilization. This keeps it “tidally locked” so that it rotates at the same rate at which it revolves, showing the Earth only one face.

Lunar Inclination



18 Year Lunar Inclination Cycle

- ☐ Lunar inclination (with respect to the Earth equator) varies from minimum of 18.3° to a maximum 28.6° over a period of about 18.6 years

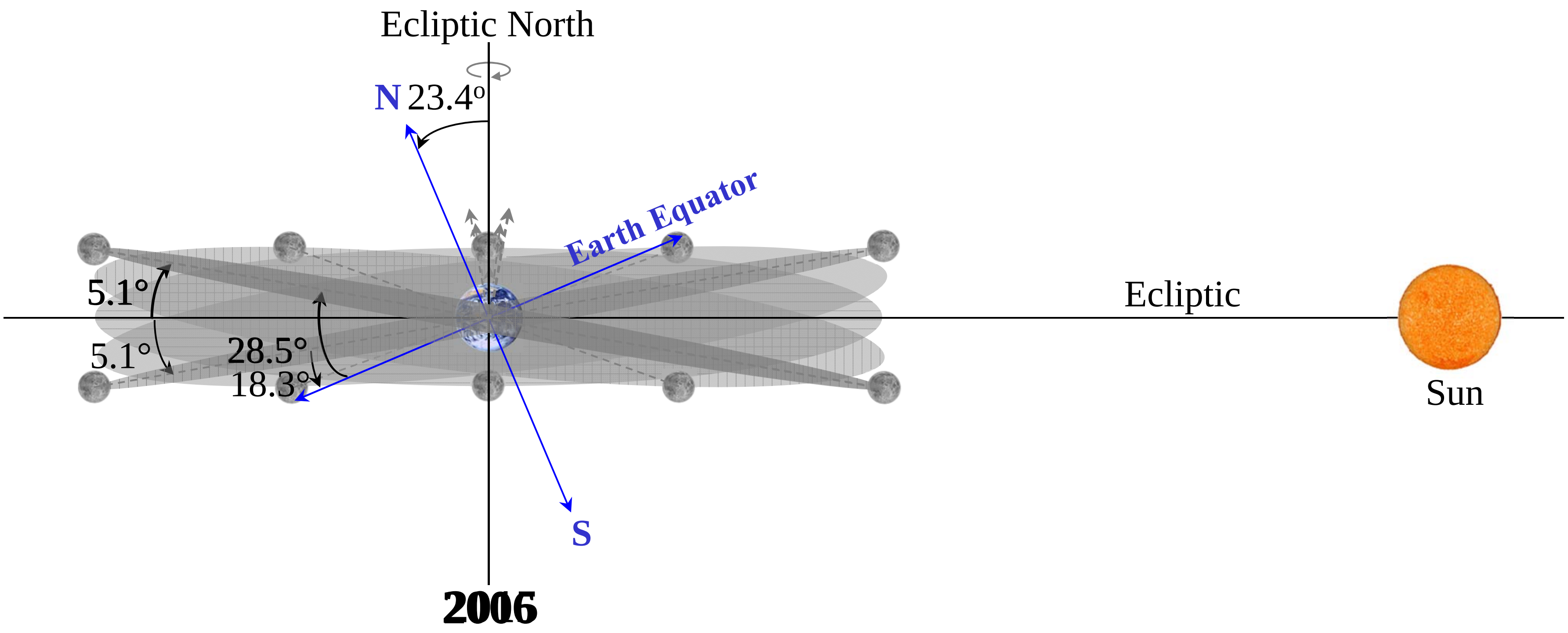
- ☐ The next maximum inclination: June 2006

- ☐ The next minimum inclination: October 2015

- ☐ The lunar inclination affects the geocentric lunar transfer orbit inclination, hence propulsion costs
  - Dependent upon launch scenario
    - Ground launched to immediate departure phasing orbit
    - Fixed Earth orbit departure

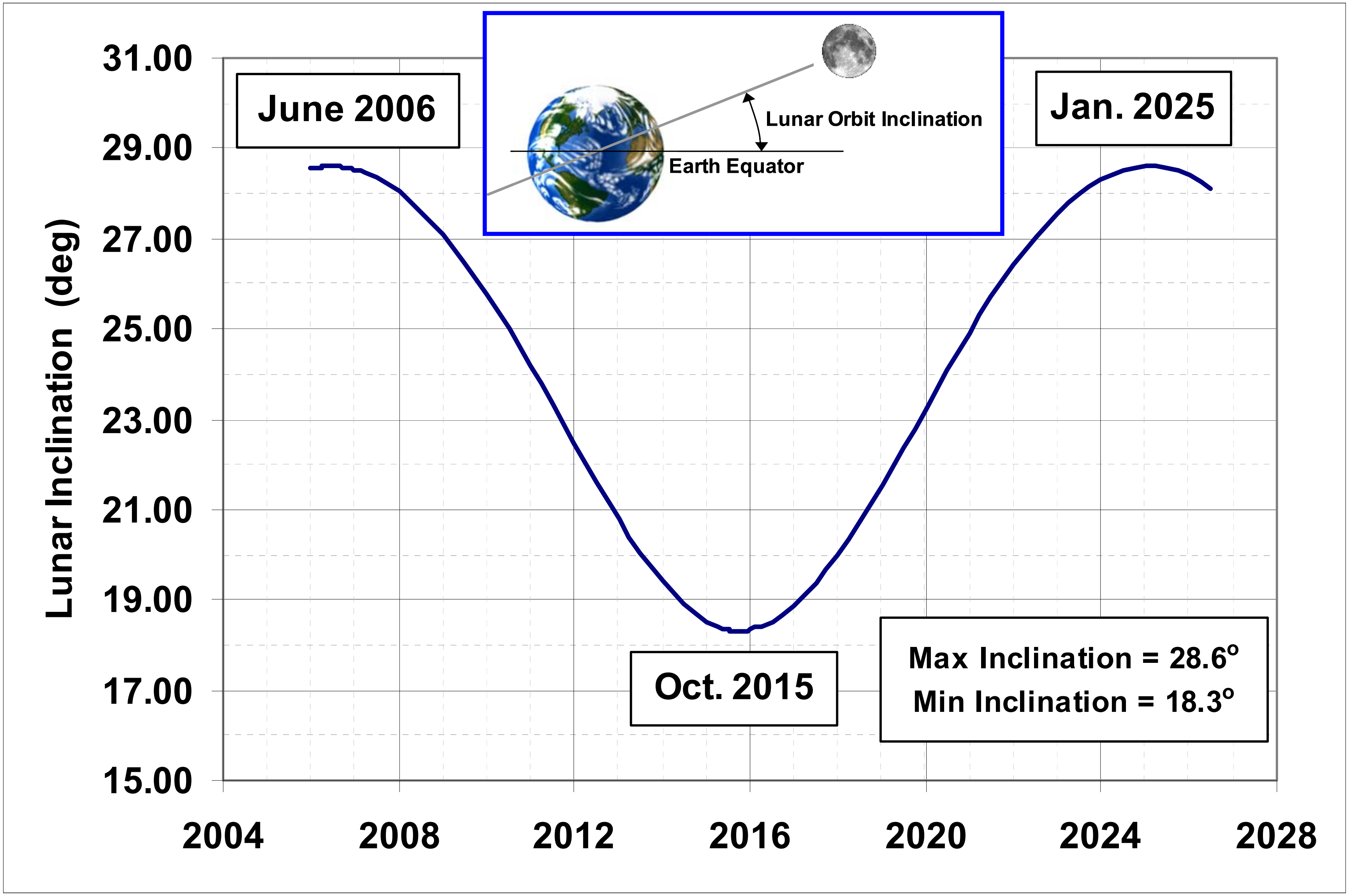
Lunar Inclination

- The Earth’s equator is tilted 23.4° from the ecliptic plane
- The Moon’s orbit is tilted 5.1° from the ecliptic plane
- The Moon’s orbit rotates about the ecliptic north 360° about every 18.6 years
  - In 2006, this results in a 28.5° lunar inclination to the Earth equator
  - In 2015, this results in an 18.3° lunar inclination to the Earth equator



18 Year Lunar Inclination Cycle

Lunar Inclination wrt Earth Equator vs Date



Lunar Libration

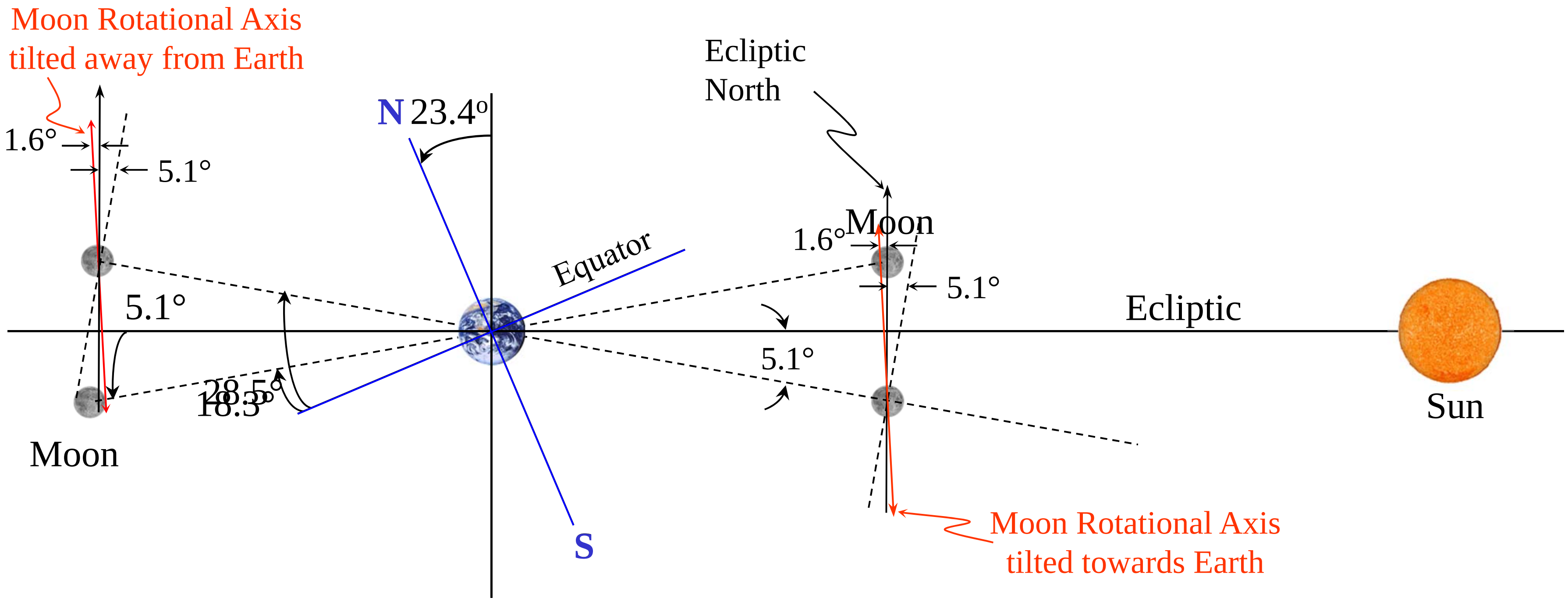
Lunar Libration

- Lunar libration causes a variation in the lunar surface that faces Earth
- Up to 59% of the lunar surface is visible from Earth (about 50% without libration)
- Libration occurs in both longitude (±8°) and latitude (±6.7°)
- Note: Lunar libration can cause points on the lunar surface to rotate in and out of view from Earth

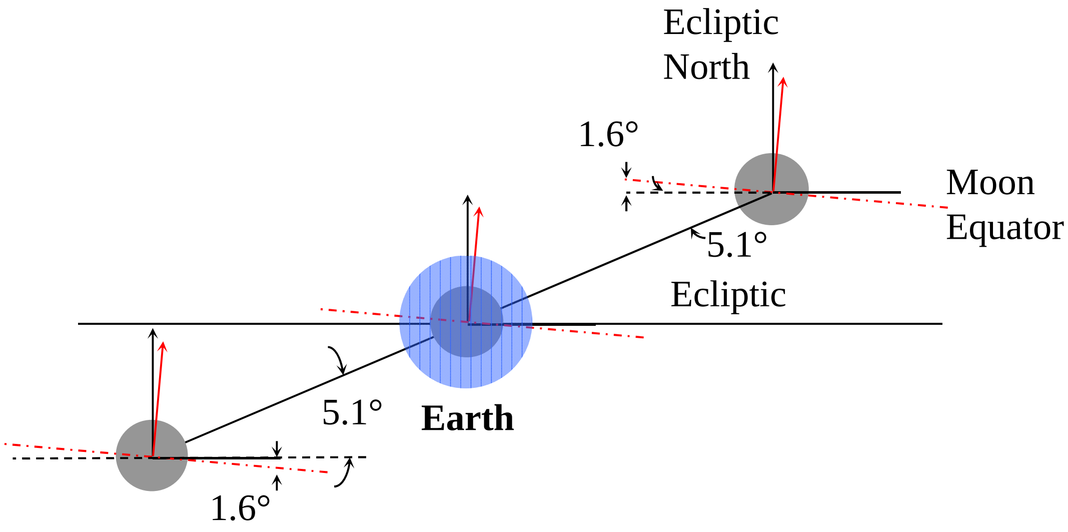
Lunar Libration and Phases

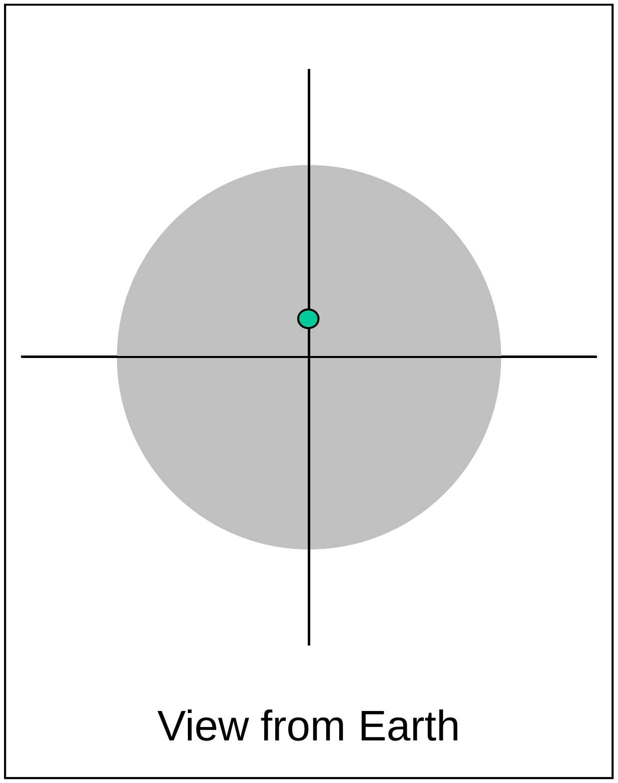
Lunar Libration - Latitude

- ❑ The Moon ‘faces the Earth’ as it rotates about the Earth
- ❑ The Moon maintains a 5.1° inclination to the ecliptic
- ❑ The Moon’s rotational axis is inclined:
  - ▪ 1.6° from the ecliptic north
  - ▪ 6.7° from the angular momentum vector of the lunar orbit plane
- ❑ This results in an apparent latitude movement of about 6.7° (up and down) as viewed from Earth



Lunar Libration -- Latitude



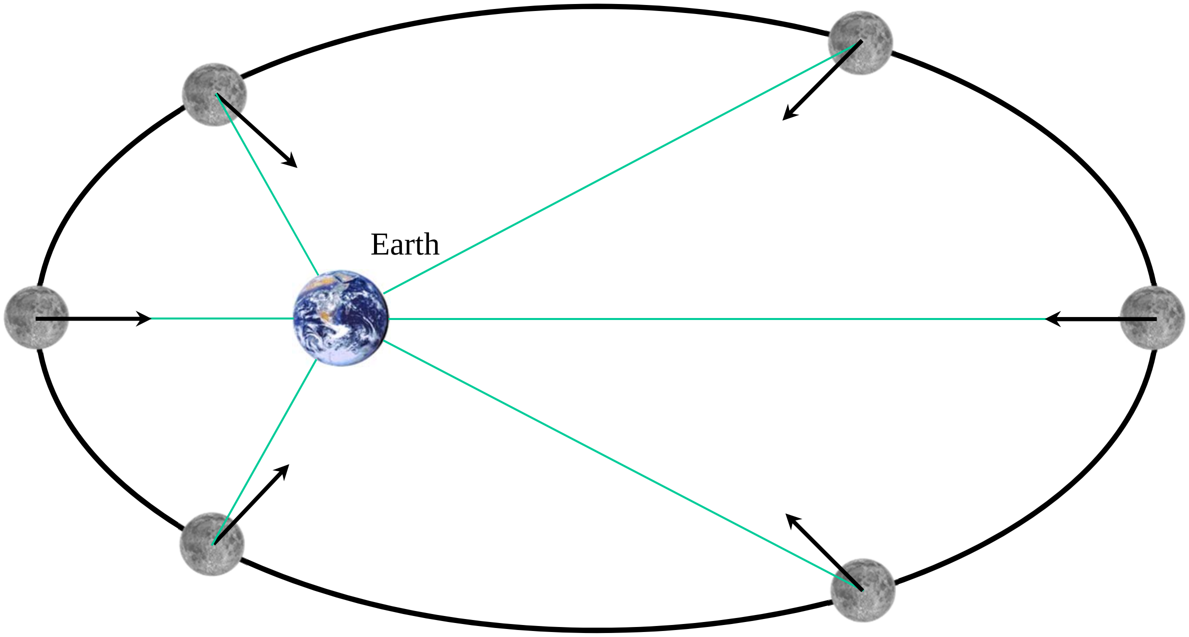


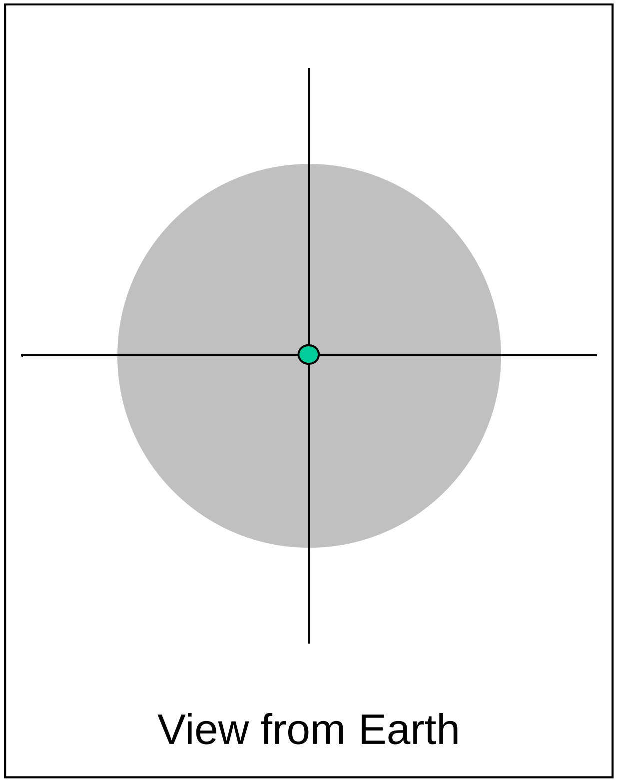
Lunar Libration - Longitude

- ☐ The Moon 'turns the same face to the Earth', so that its rotation about its axis is equal in period to the time for one orbit around the Earth
- ☐ The Moon's speed of rotation about its axis remains essentially constant as a consequence of the conservation of angular momentum
- ☐ The Moon has an elliptical orbit about the Earth, so the Moon speeds up near perigee and slows down near apogee in accordance with Kepler's laws
- ☐ The differences between the lunar rotation rate and the rotation rate of the Moon’s velocity vector create an apparent back and forth (east-west) nodding of the Moon.

Lunar Libration - Longitude

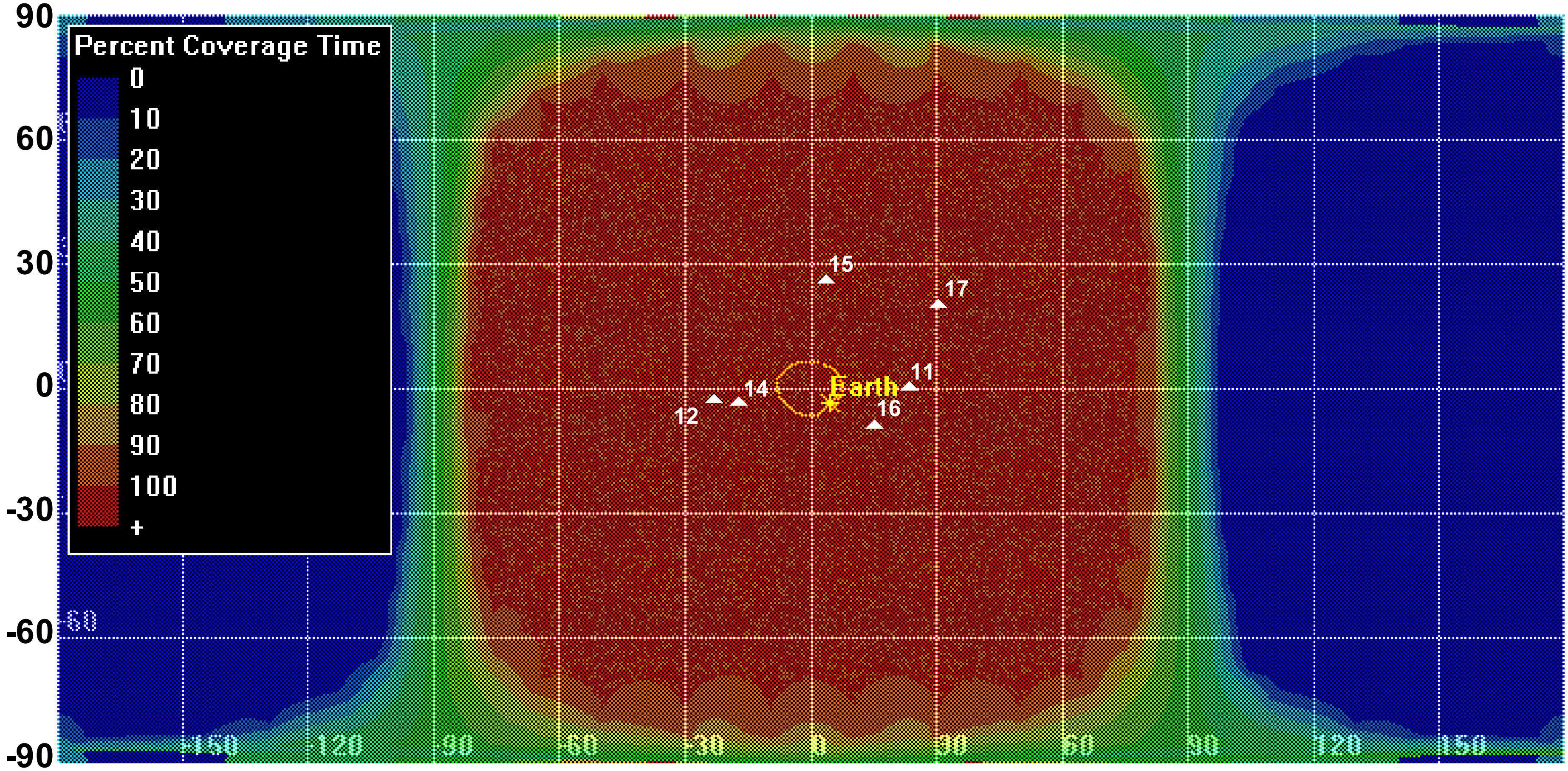
- The differences between the lunar rotation rate and the rotation rate of the Moon’s velocity vector create an apparent back and forth (east-west) nodding of the Moon.





Moon View From Earth

Percentage Viewing Over One Lunar Rotation

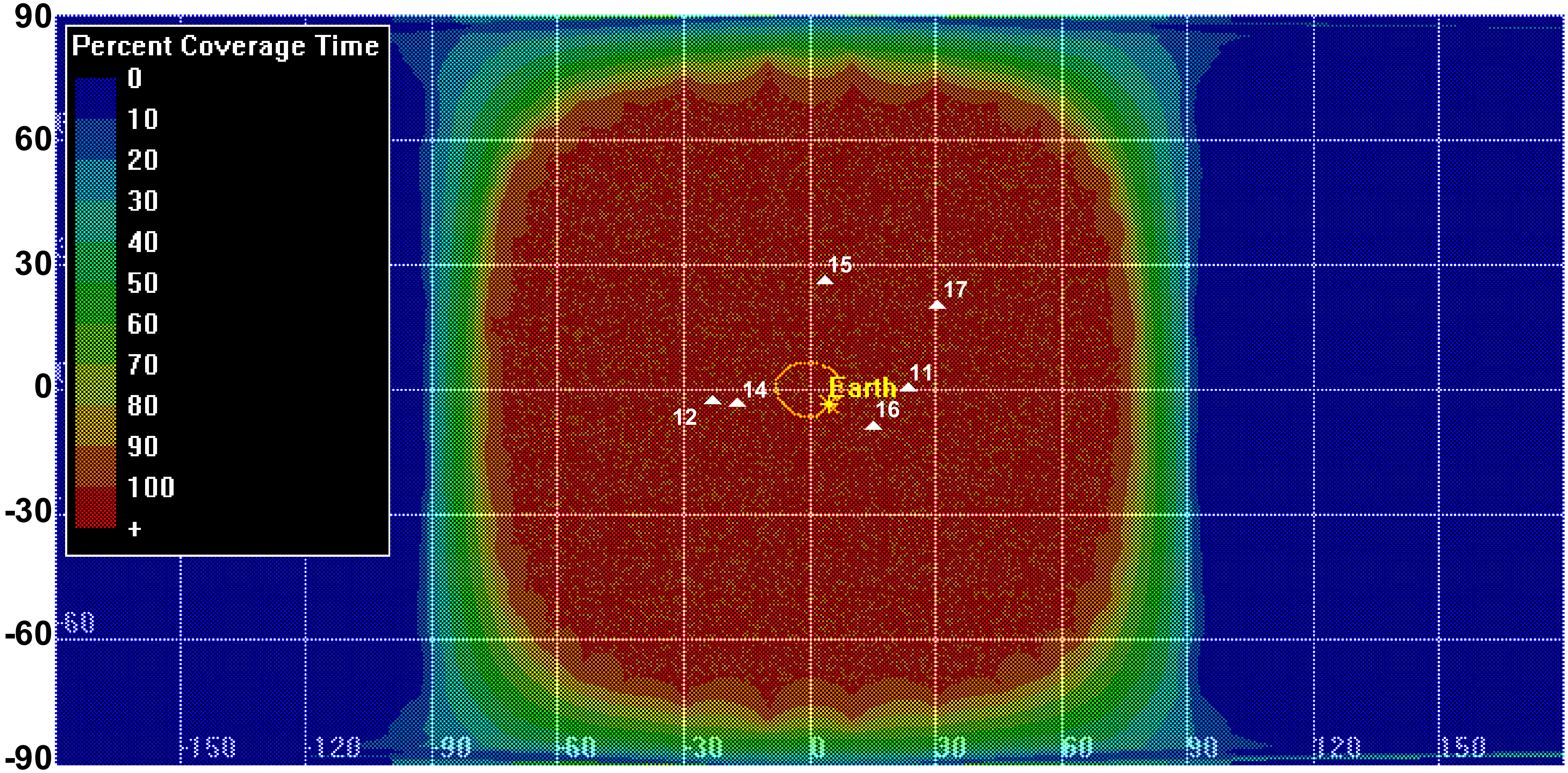


Minimum 0° Mask Angle at Moon

Mar-2011

Moon View From Earth

Percentage Viewing Over One Lunar Rotation

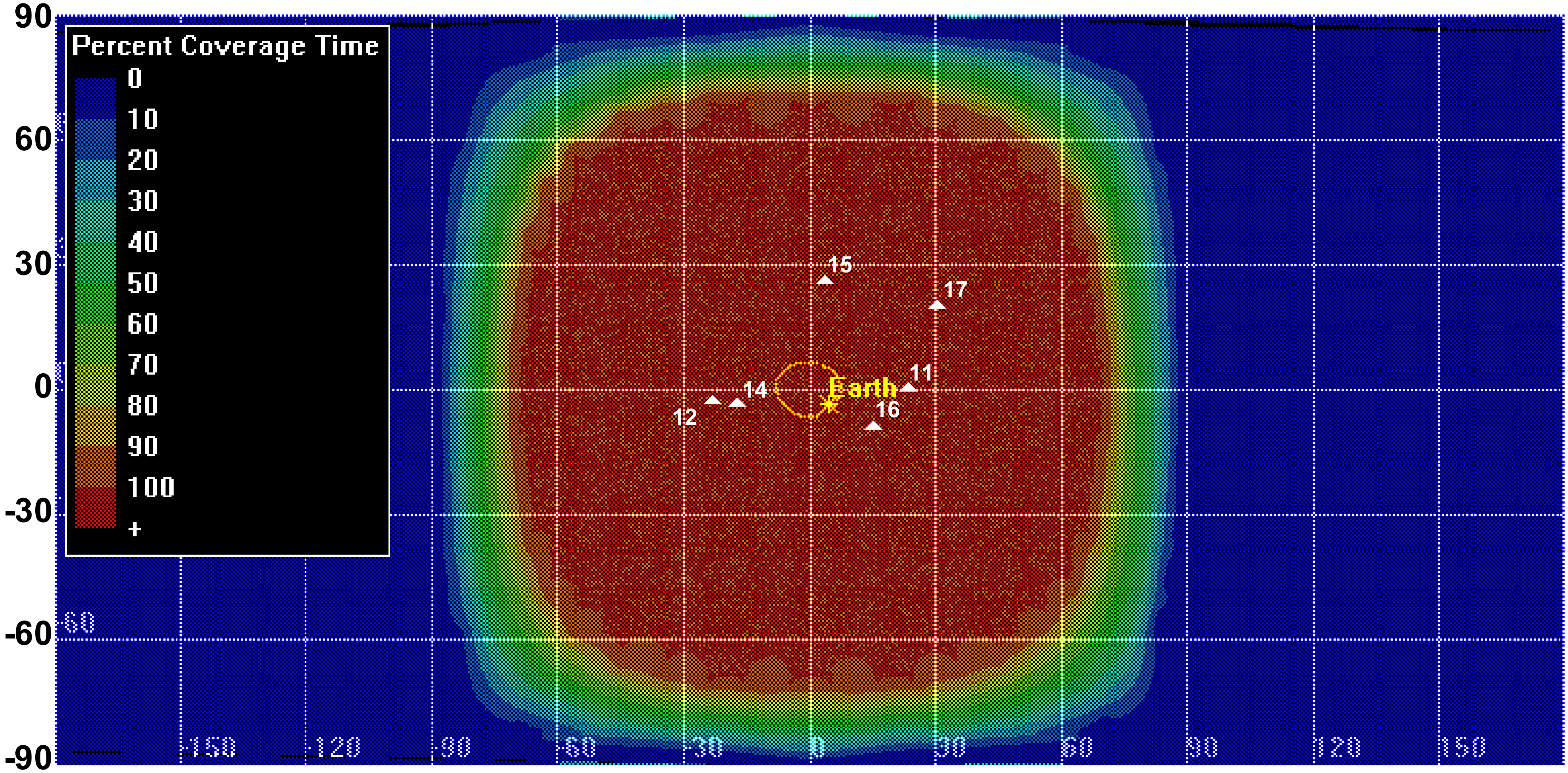


Minimum 5° Elevation Mask Angle at Moon

Mar-2011

Moon View From Earth

Percentage Viewing Over One Lunar Rotation

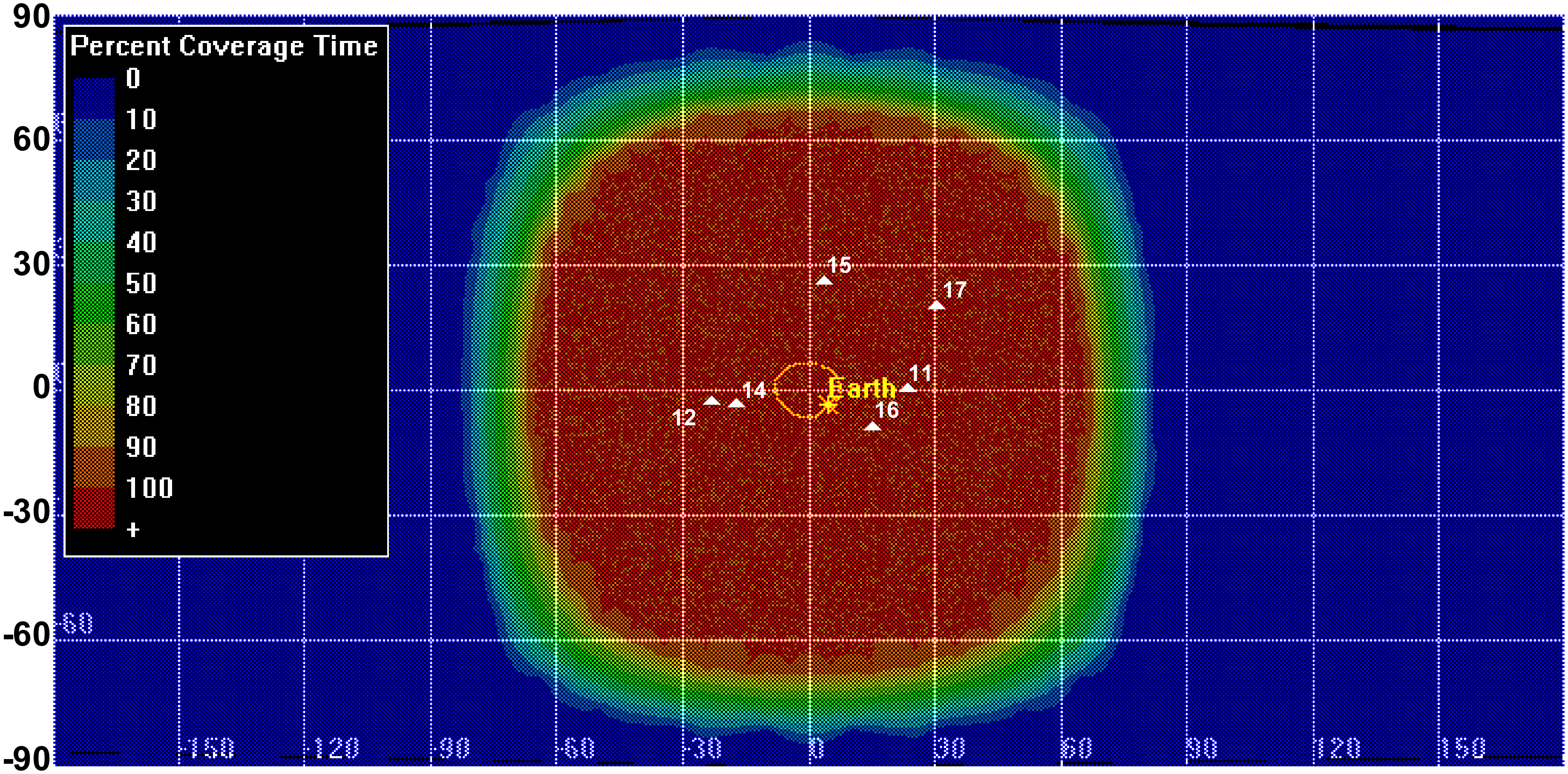


Minimum 10° Elevation Mask Angle at Moon

Mar-2011

Moon View From Earth

Percentage Viewing Over One Lunar Rotation

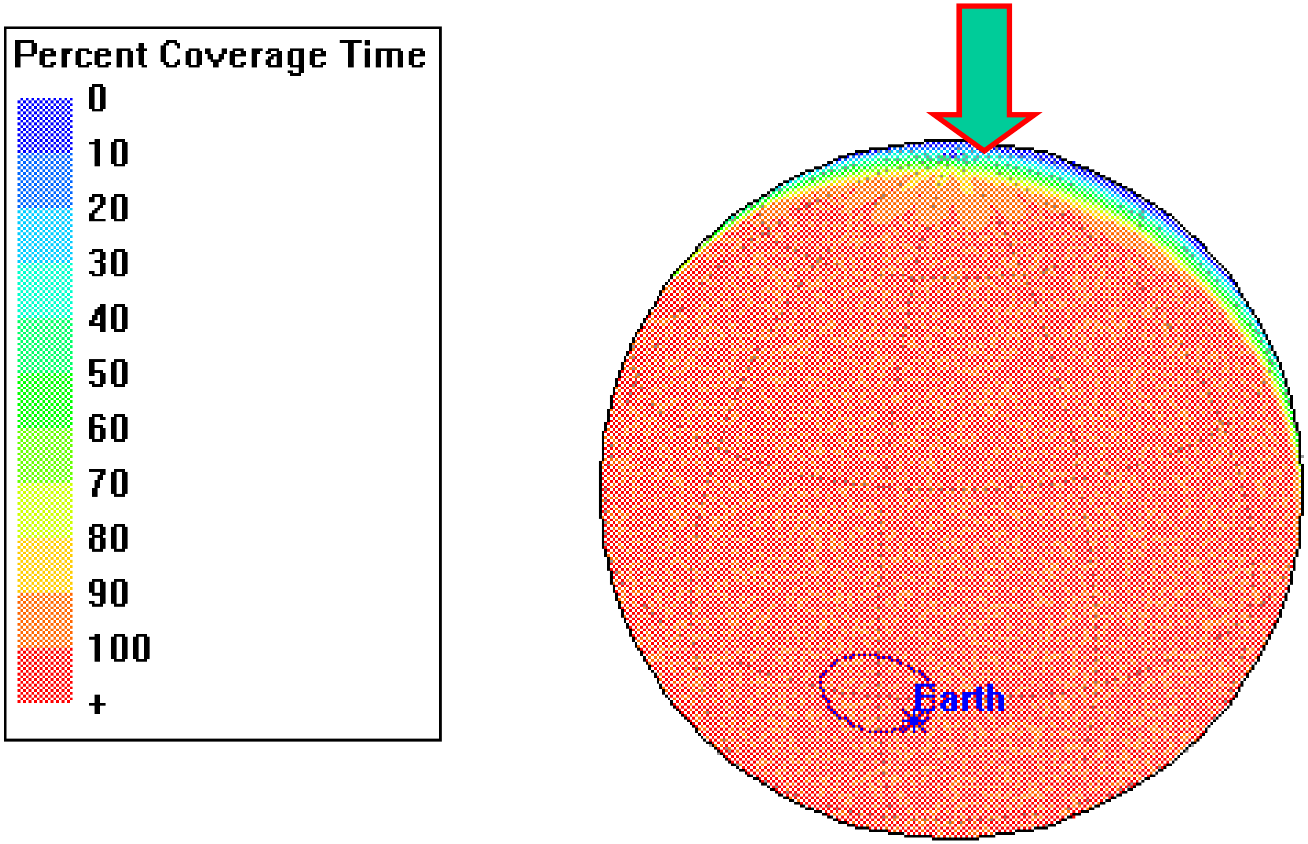


Minimum 15° Elevation Mask Angle at Moon

Mar-2011

Moon View From Earth

Percentage Viewing Over One Lunar Rotation

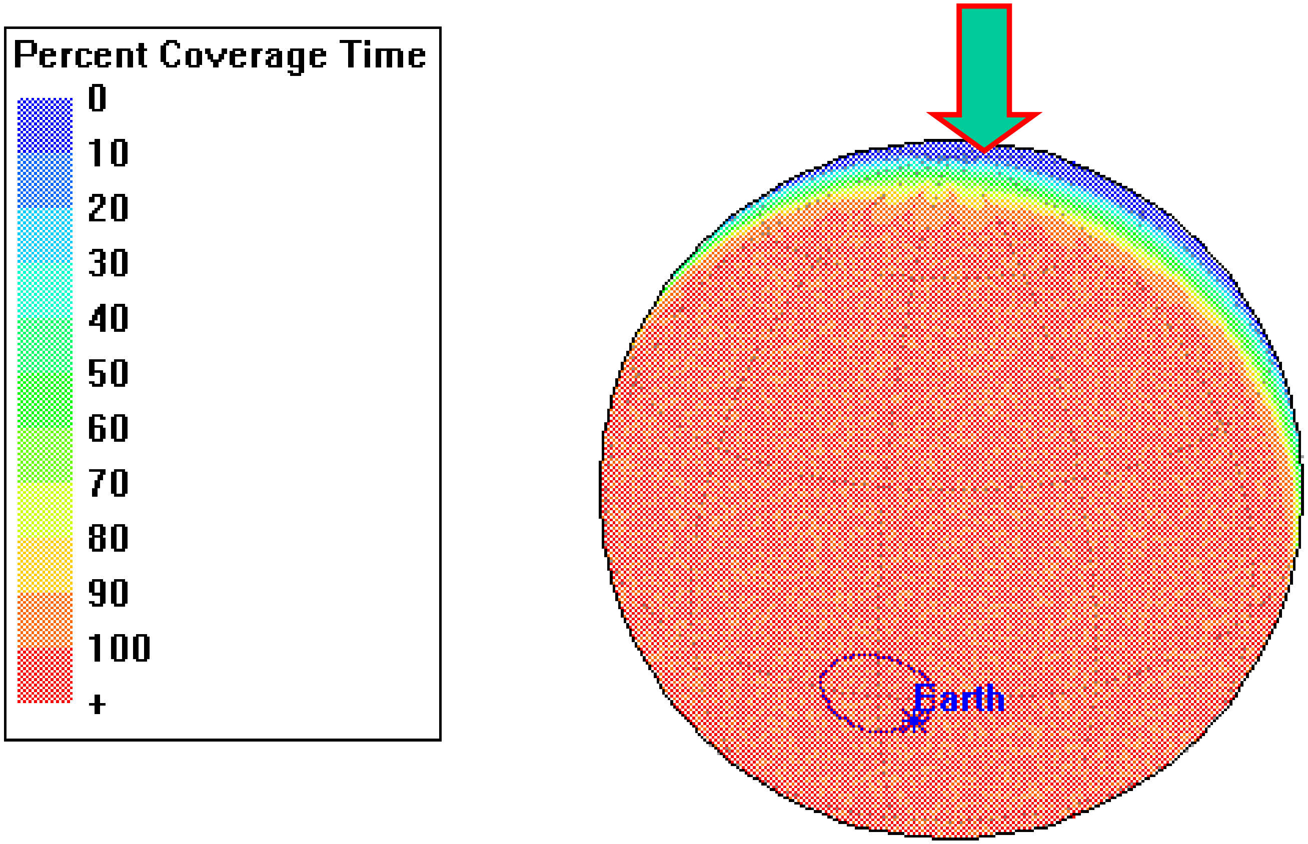


Minimum 0° Elevation Mask Angle at Moon

Mar-2011

Moon View From Earth

Percentage Viewing Over One Lunar Rotation

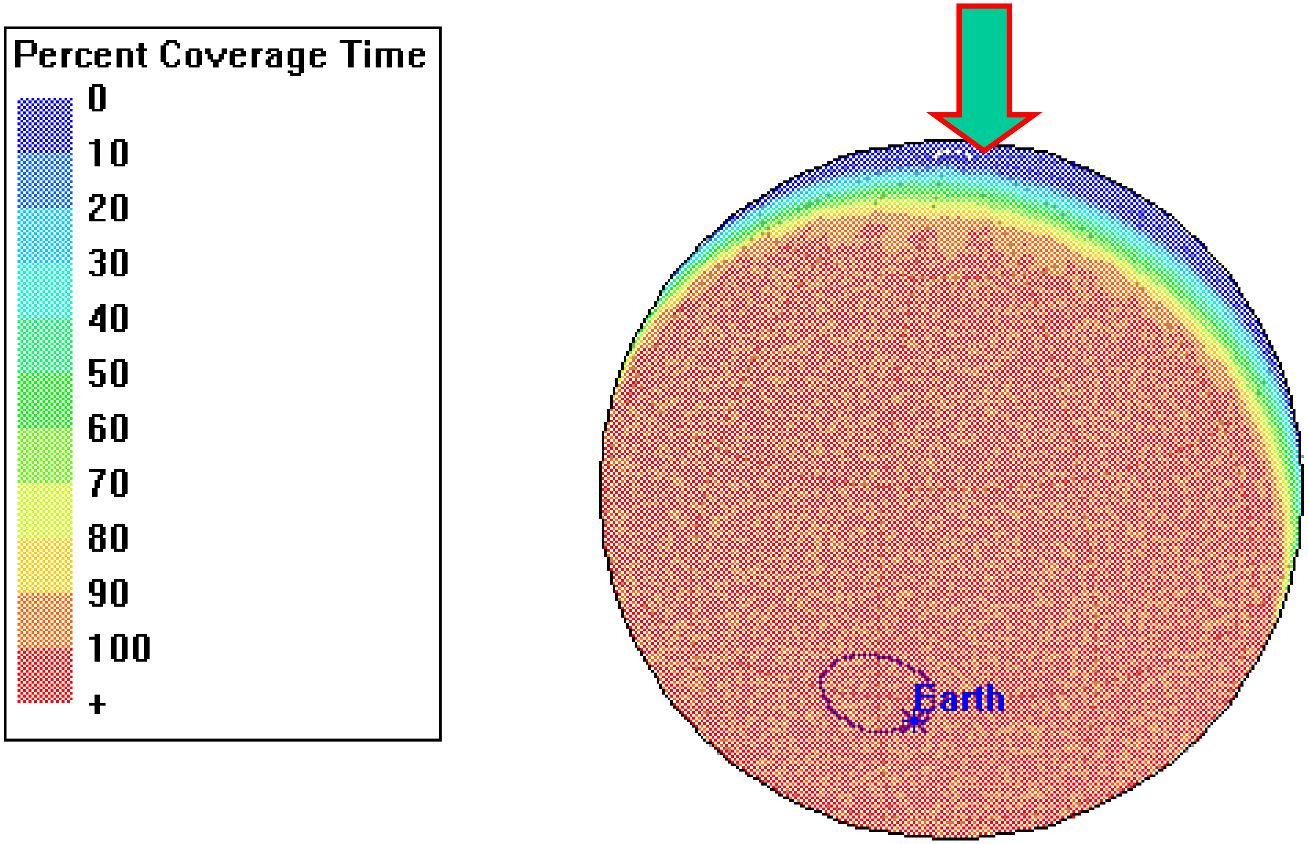


Minimum 5° Elevation Mask Angle at Moon

Mar-2011

Moon View From Earth

Percentage Viewing Over One Lunar Rotation

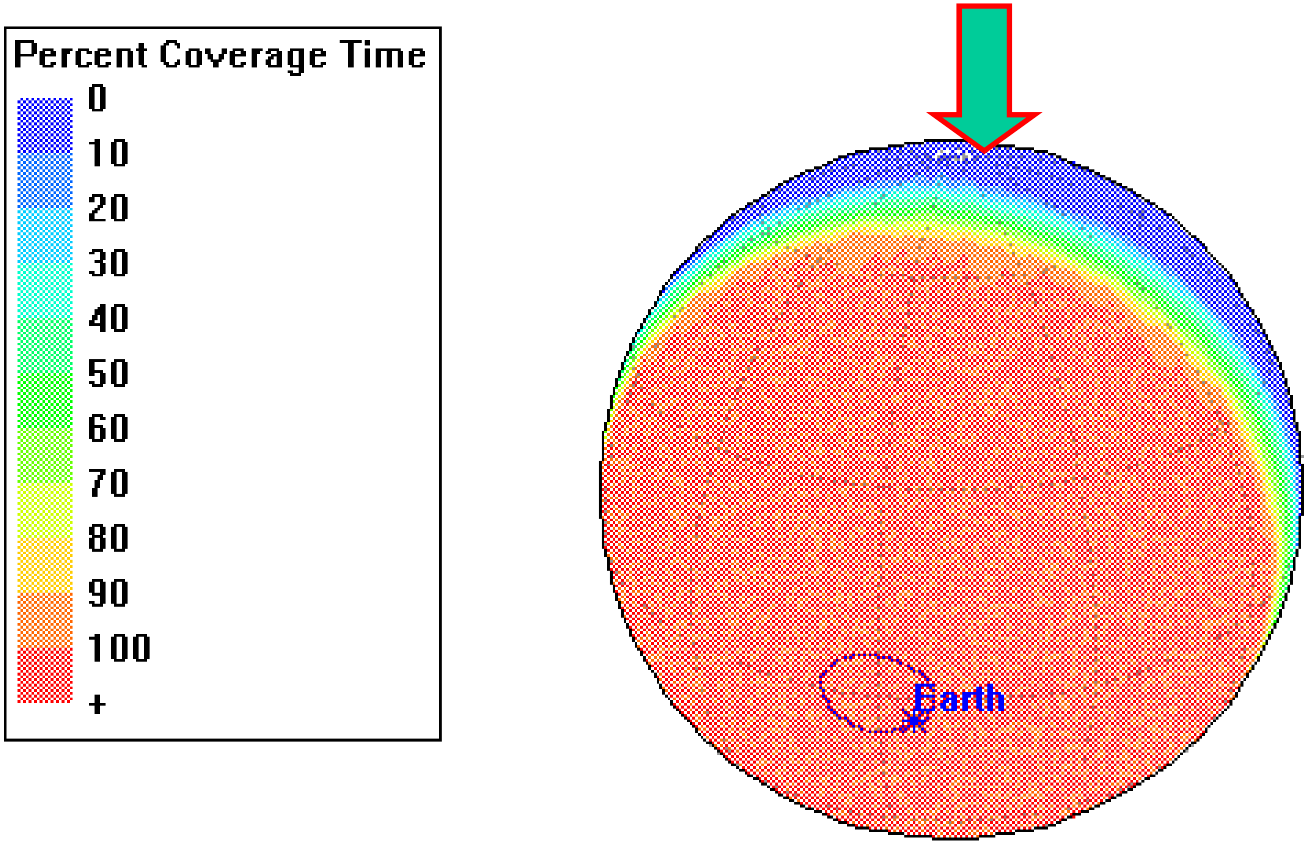


Minimum 10° Elevation Mask Angle at Moon

Mar-2011

Moon View From Earth

Percentage Viewing Over One Lunar Rotation



Minimum 15° Elevation Mask Angle at Moon

Mar-2011

Lunar Libration - Summary

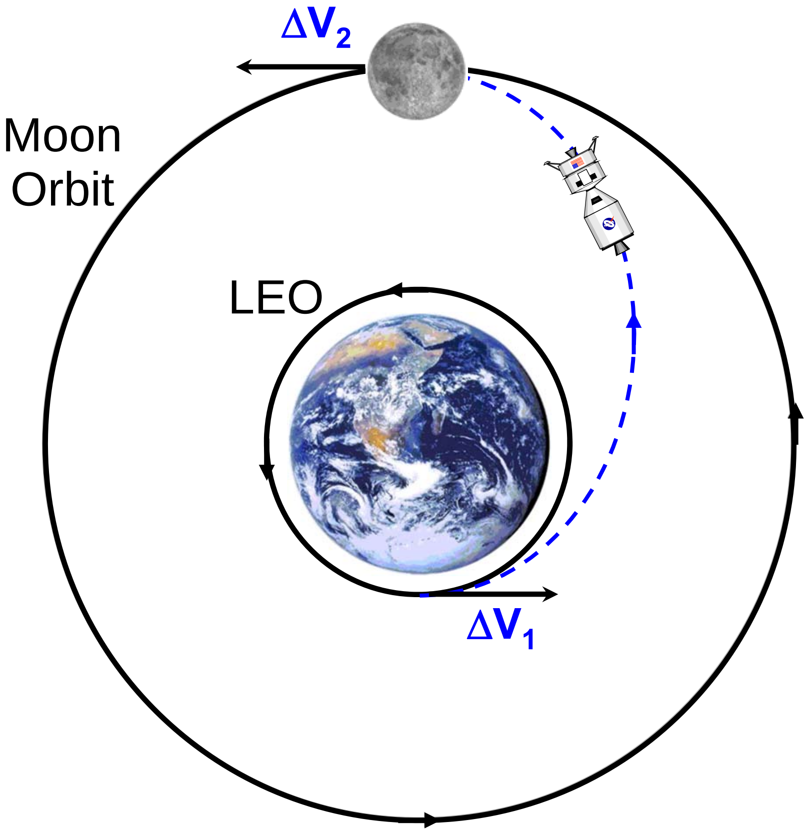
- ☐ Up to 59% of the lunar surface is visible from Earth (about 50% without libration)
- ☐ Libration occurs in both longitude (±8°) and latitude (±6.7°)
- ☐ Landing sites near the limb of the Moon (e.g., north and south poles and east and west limbs) may nod in and out of Earth view periodically with lunar rotation about Earth
  - Surface crew out of Earth communication (without bent-pipe satellite aid)
- ☐ Lunar terrain may exacerbate the Earth-viewing problem
  - A polar landing site in a valley would have Earth viewing further reduced
  - A polar landing site on a high hill may have continuous Earth view
  - Better lunar terrain models are needed*

*** Currently, proposals exist to provide high-resolution lunar gravity mapping and improved lunar terrain models.**

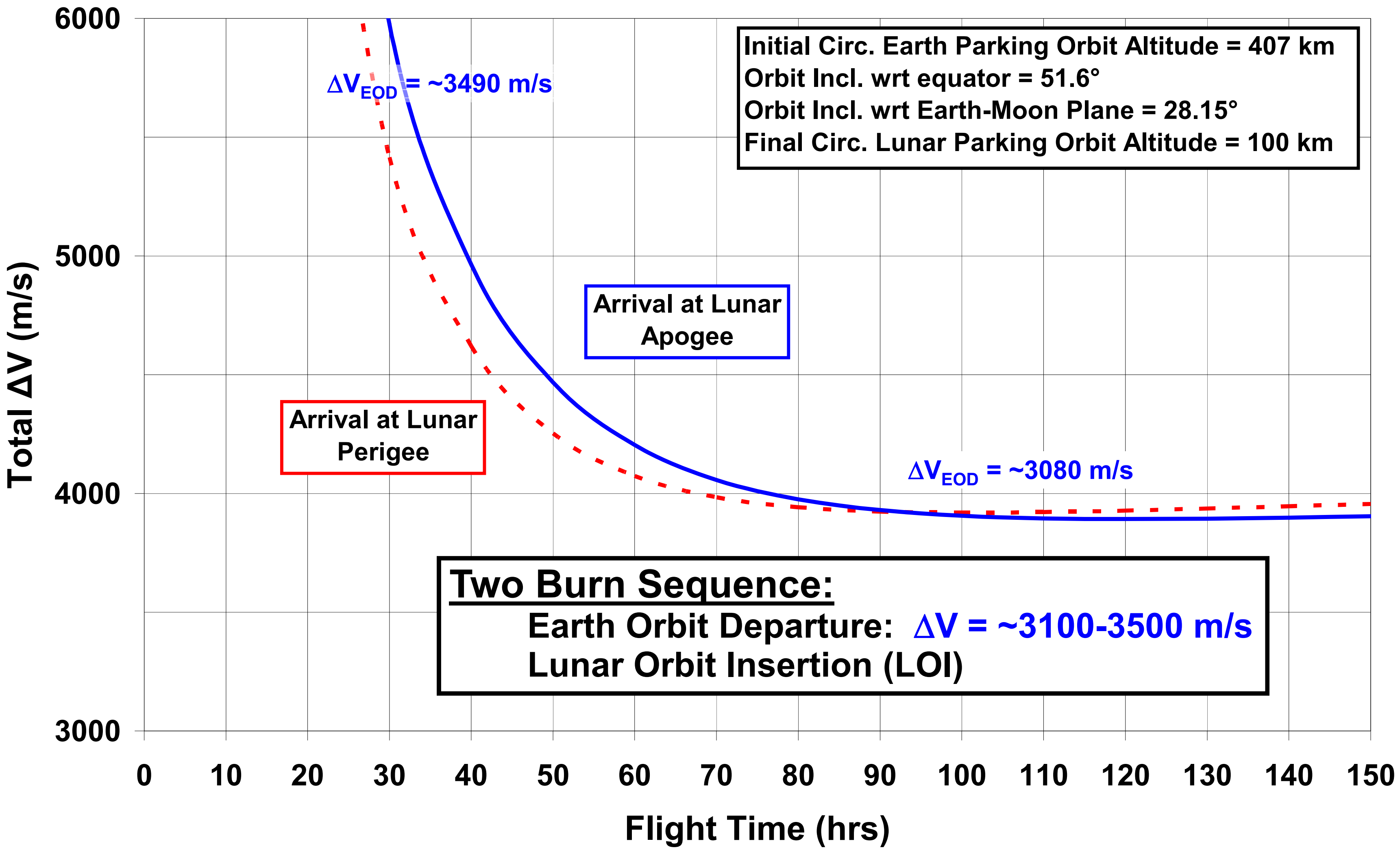
Geocentric Transfer Characteristics

Earth to Moon Transfer

- ❑ High thrust Earth-Moon transfer consists of two primary maneuvers: Earth orbit departure (EOD) and Lunar Orbit Insertion (LOI)
- ❑ The ΔV cost for EOD and LOI is about 3100 m/s and 900 m/s, respectively



Earth to Moon – ΔV Cost



Initial Circ. Earth Parking Orbit Altitude = 407 km
Orbit Incl. wrt equator = 51.6°
Orbit Incl. wrt Earth-Moon Plane = 28.15°
Final Circ. Lunar Parking Orbit Altitude = 100 km

Two Burn Sequence:
- Earth Orbit Departure: ΔV = ~3100-3500 m/s
- Lunar Orbit Insertion (LOI)

Earth-Moon Transfer

- ☐ Earth orbit departure (EOD)
  - ▪ Tangential EOD
    - Non-coplanar, non-tangential thrusting has severe performance penalties

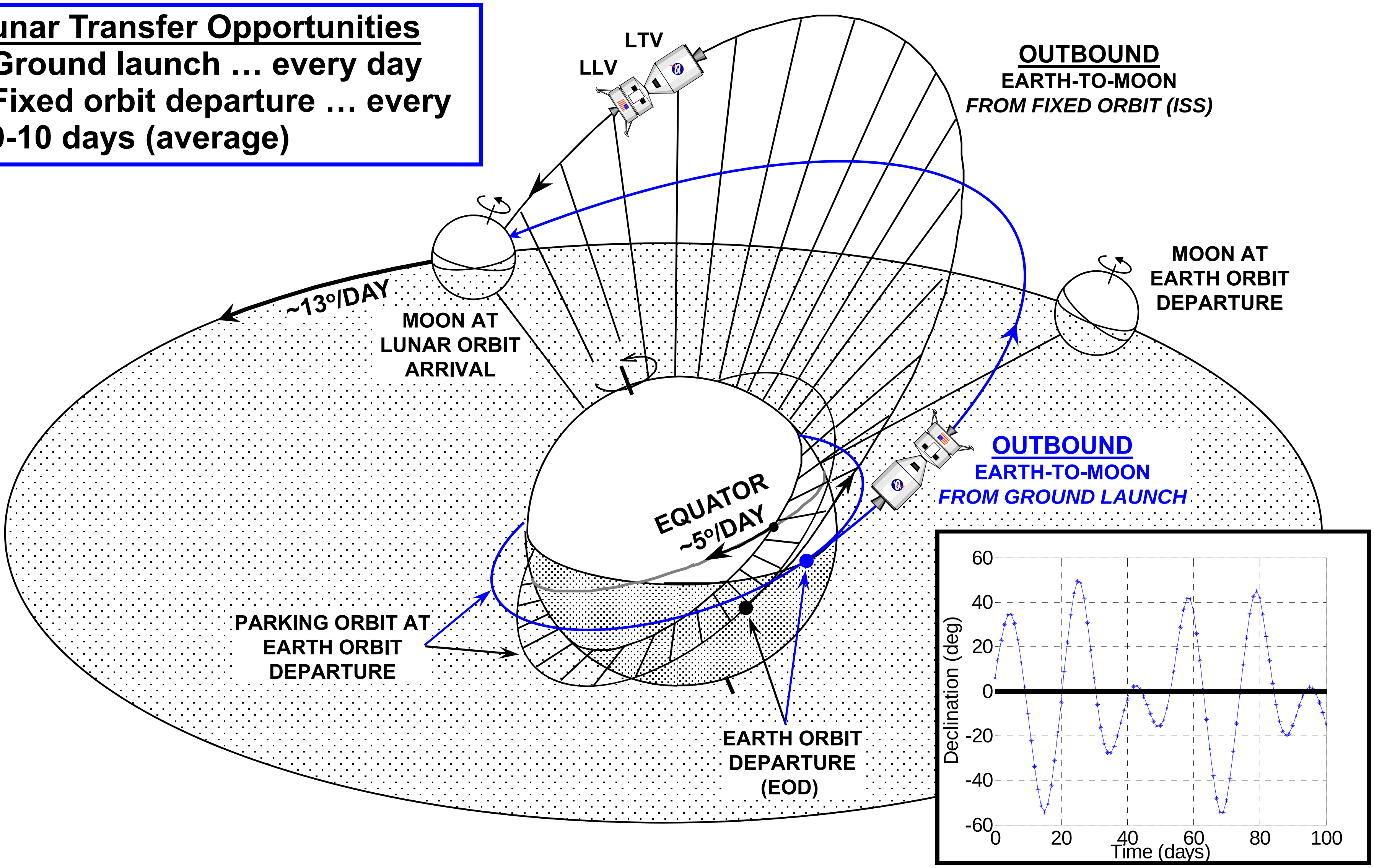
Departure Options:

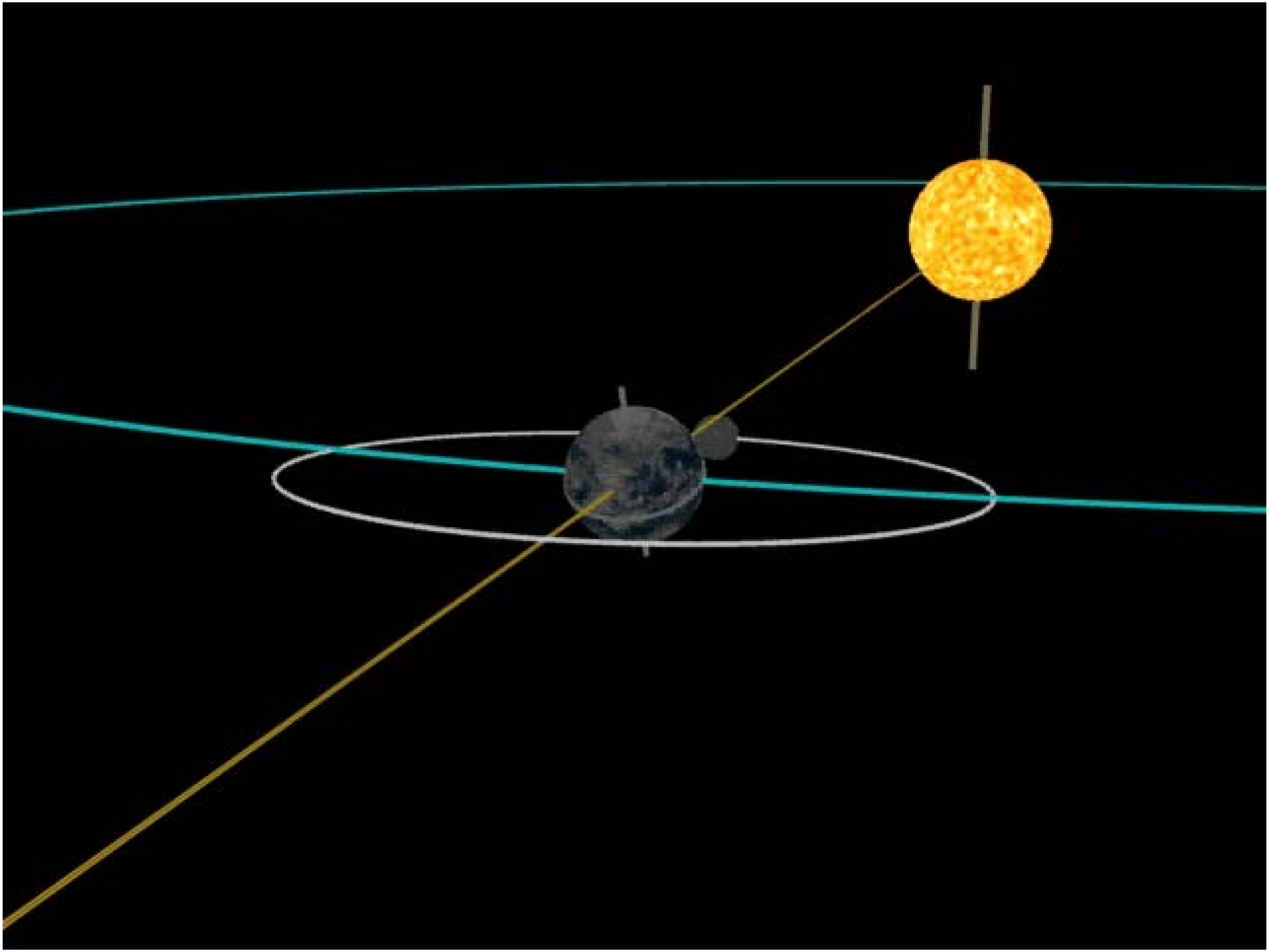
1. EOD after ground launch to a low Earth orbit (LEO) phasing orbit
   - Selectable departure plane
   - Daily launch/EOD opportunities
2. EOD from pre-established LEO parking orbit (e.g., ISS, 28.5° construction orbit)
   - Fixed departure plane
   - EOD opportunities average every 9-10 days*

*The combination of the Moon's orbital motion (~13°/day) plus ISS nodal regression of ~5°/day results in ~18°/day relative movement between orbit plane and Moon or a coplanar EOD opportunity averaging about every 10 days. For a 28.5° orbit, EOD opportunities occur on the average about every 9 days.

Earth-Moon Transfer

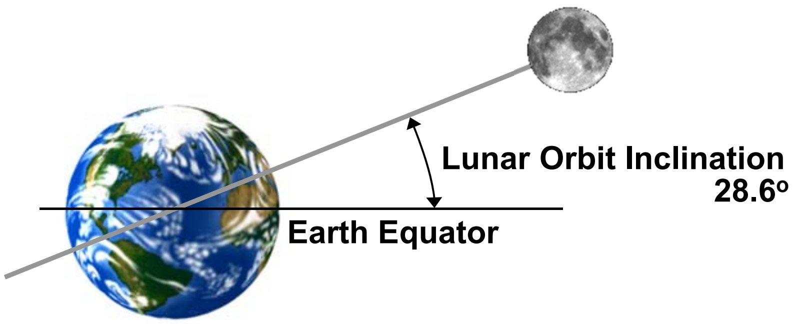
Lunar Transfer Opportunities
- Ground launch … every day
- Fixed orbit departure … every 9-10 days (average)

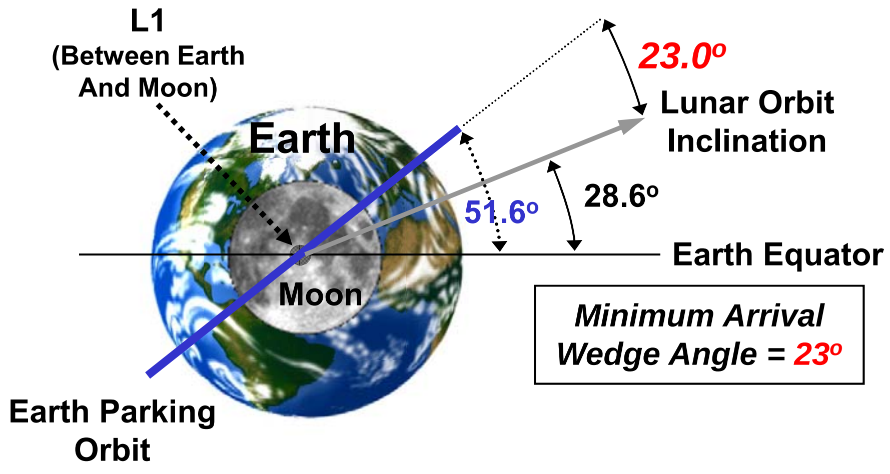




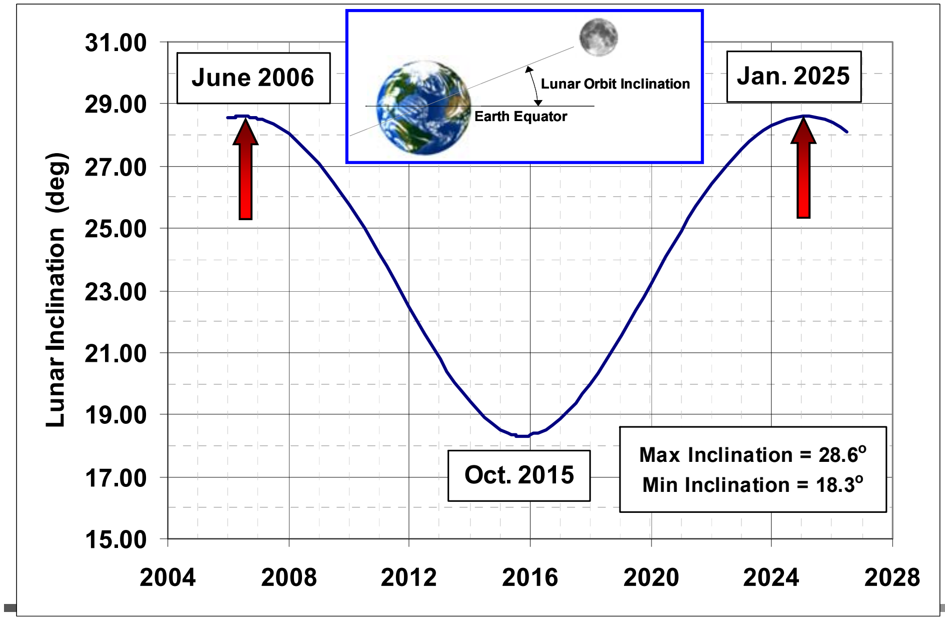
18 Year Lunar Inclination Cycle

Example: Departure from Fixed (ISS) Parking Orbit

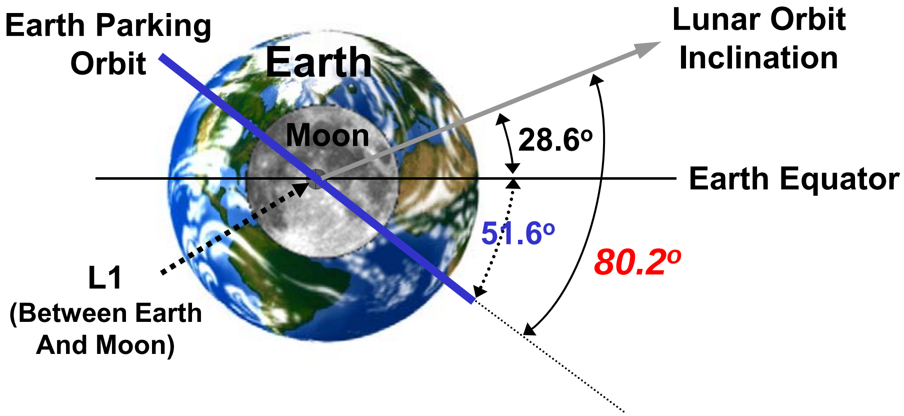




Minimum Arrival Wedge Angle = 23°

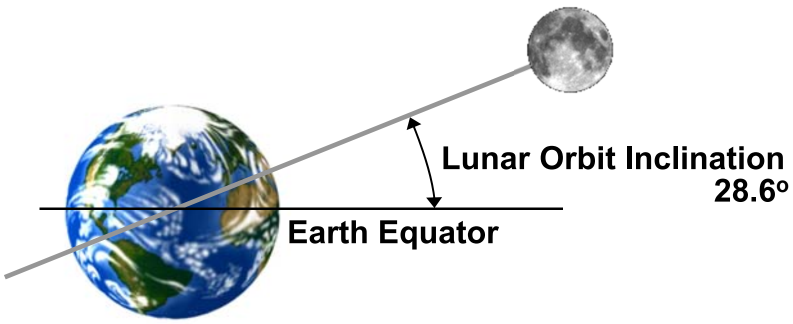


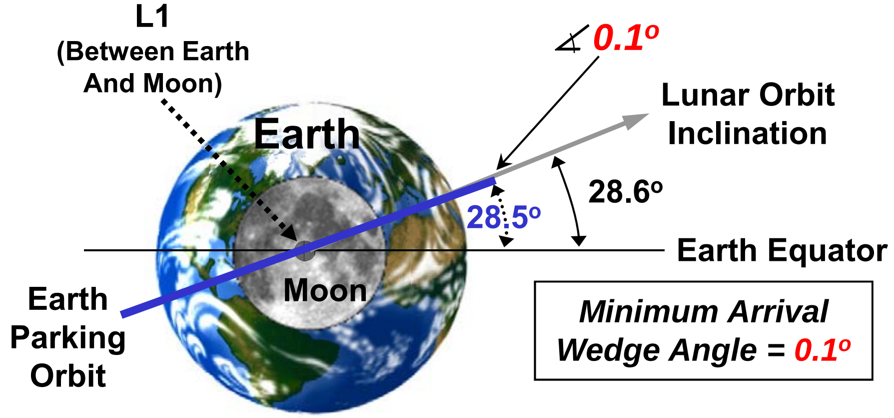
Maximum Arrival Wedge Angle = 80.2°



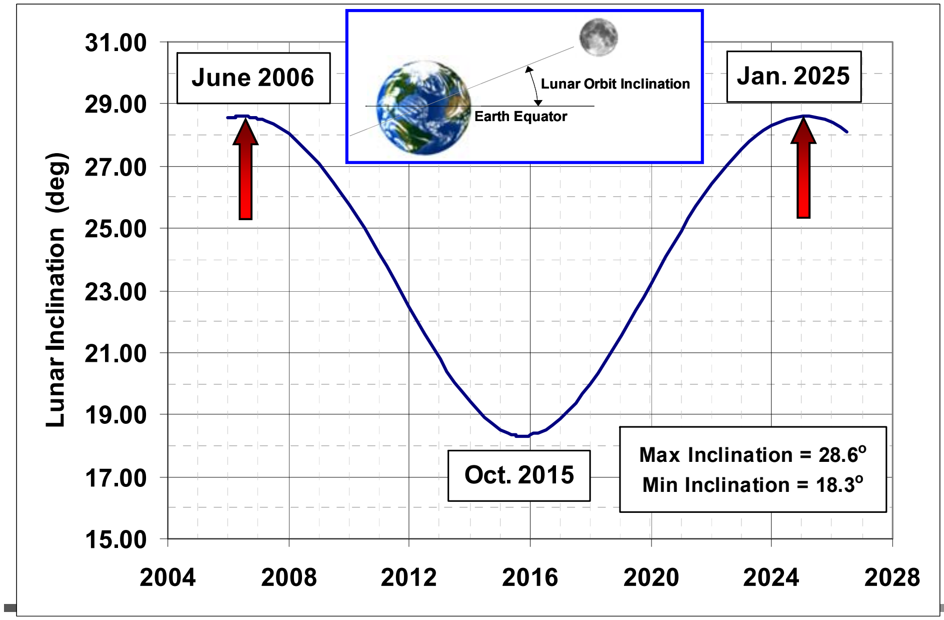
18 Year Lunar Inclination Cycle

Example: Departure from Fixed (28.5°) Parking Orbit



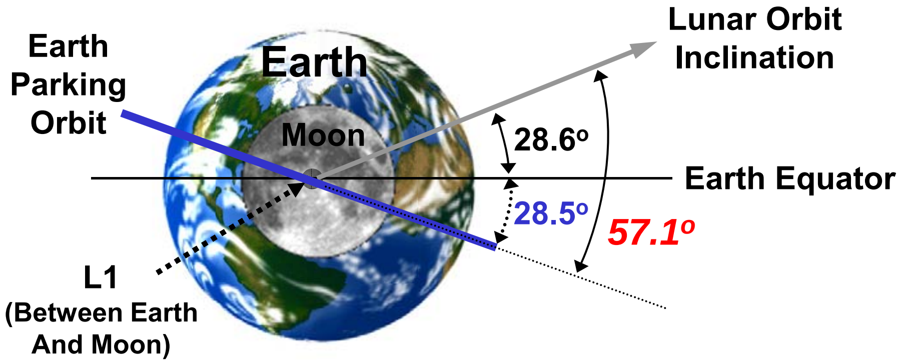


Minimum Arrival Wedge Angle = 0.1°



Max Inclination = 28.6°
Min Inclination = 18.3°

Maximum Arrival Wedge Angle = 57.1°

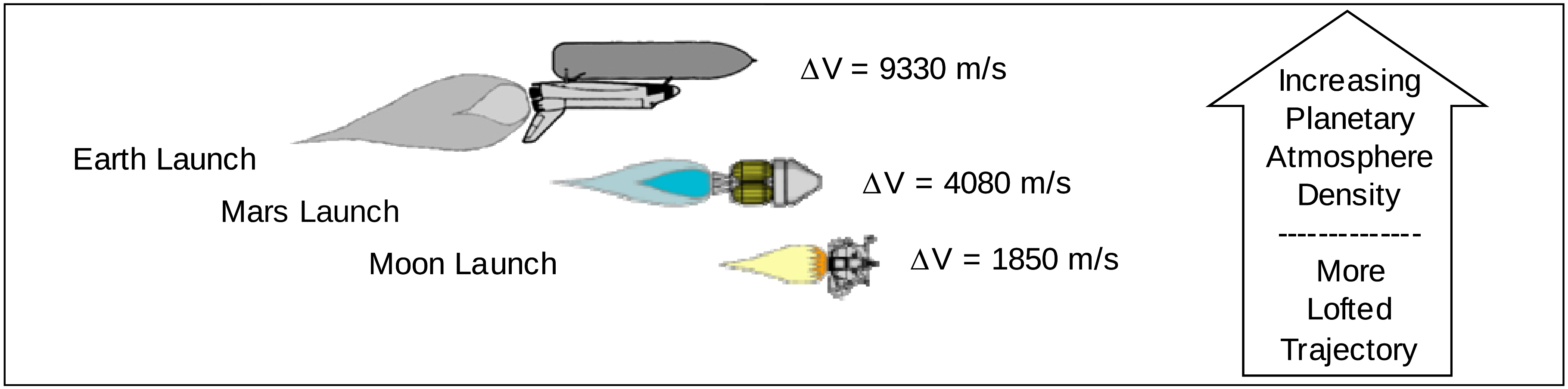


Geocentric Wedge Angle

Best and worst case geocentric wedge angle as a function of maximum and minimum lunar orbit inclination

	28.5° Departure Orbit		51.6° (ISS) Departure Orbit	
Lunar Inclination (w.r.t. Earth Equator)	18.3° (Minimum)	28.6° (Maximum)	18.3° (Minimum)	28.6° (Maximum)
Worst-Case Geocentric Wedge Angle between Earth-Moon Transfer Orbit and Lunar Orbit Plane	46.8°	57.1°	69.9°	80.2°
Best-Case Geocentric Wedge Angle between Earth-Moon Transfer Orbit and Lunar Orbit Plane	10.2°	0.0°	33.3°	23.0°

Ground Launch Delta-V Cost



Earth Launch

ΔV = 9330 m/s

Mars Launch

ΔV = 4080 m/s

Moon Launch

ΔV = 1850 m/s

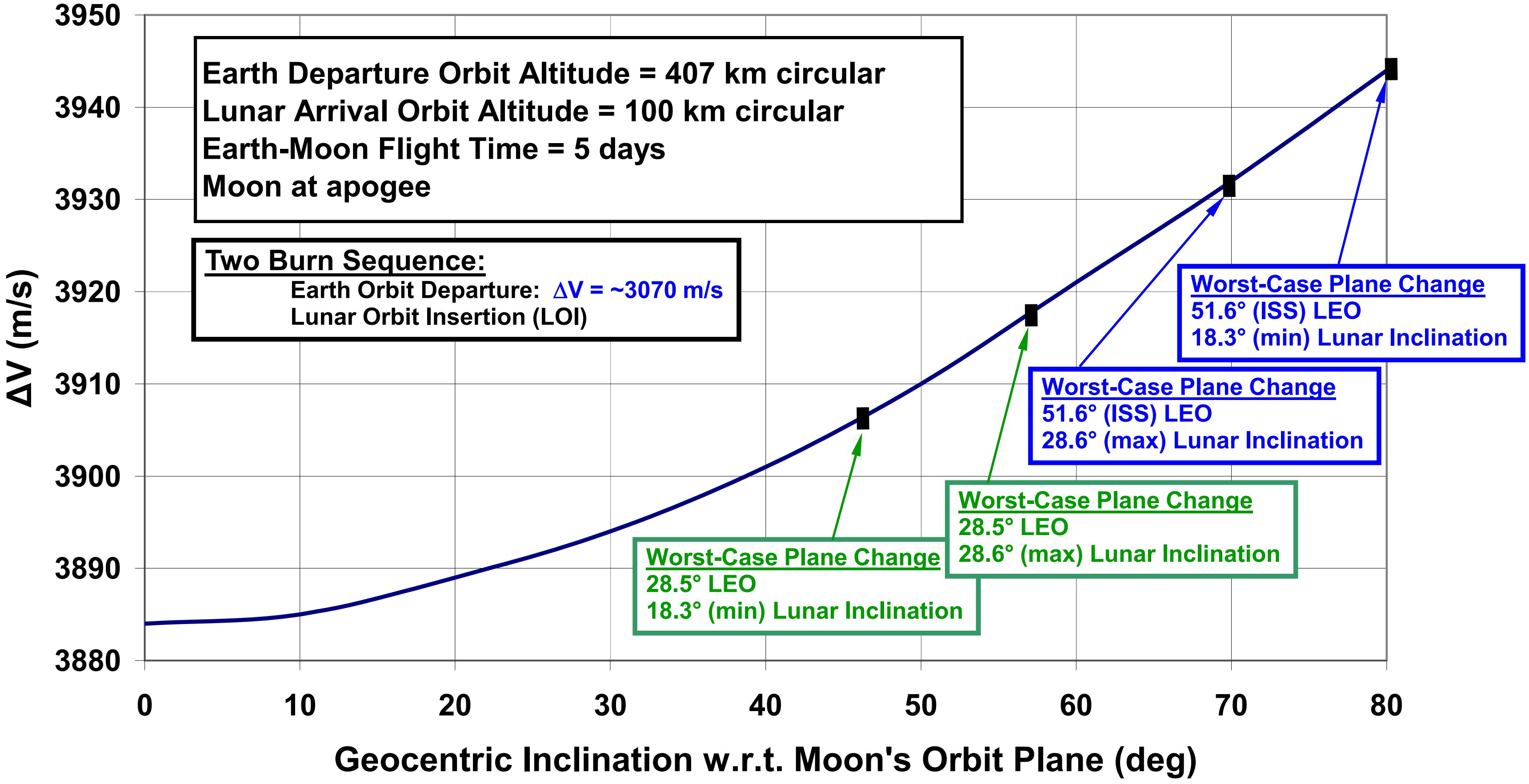
Increasing Planetary Atmosphere Density

More Lofted Trajectory

- Earth Launch
  - 100% Earth gravity; largest drag velocity losses
- Mars Launch
  - 38% Earth gravity; reduced drag velocity losses
- Moon Launch
  - 17% Earth gravity; no drag velocity losses

Delta-V vs Geocentric Inclination for Earth to Moon Transfer

Earth Parking Orbit to Lunar Parking Orbit Transfer ΔV vs. Geocentric Inclination w.r.t. Moon's Orbit Plane



Earth Departure Orbit Altitude = 407 km circular
Lunar Arrival Orbit Altitude = 100 km circular
Earth-Moon Flight Time = 5 days
Moon at apogee

Two Burn Sequence:
Earth Orbit Departure: ΔV = ~3070 m/s
Lunar Orbit Insertion (LOI)

Worst-Case Plane Change
51.6° (ISS) LEO
18.3° (min) Lunar Inclination

Worst-Case Plane Change
51.6° (ISS) LEO
28.6° (max) Lunar Inclination

Worst-Case Plane Change
28.5° LEO
28.6° (max) Lunar Inclination

Worst-Case Plane Change
28.5° LEO
18.3° (min) Lunar Inclination

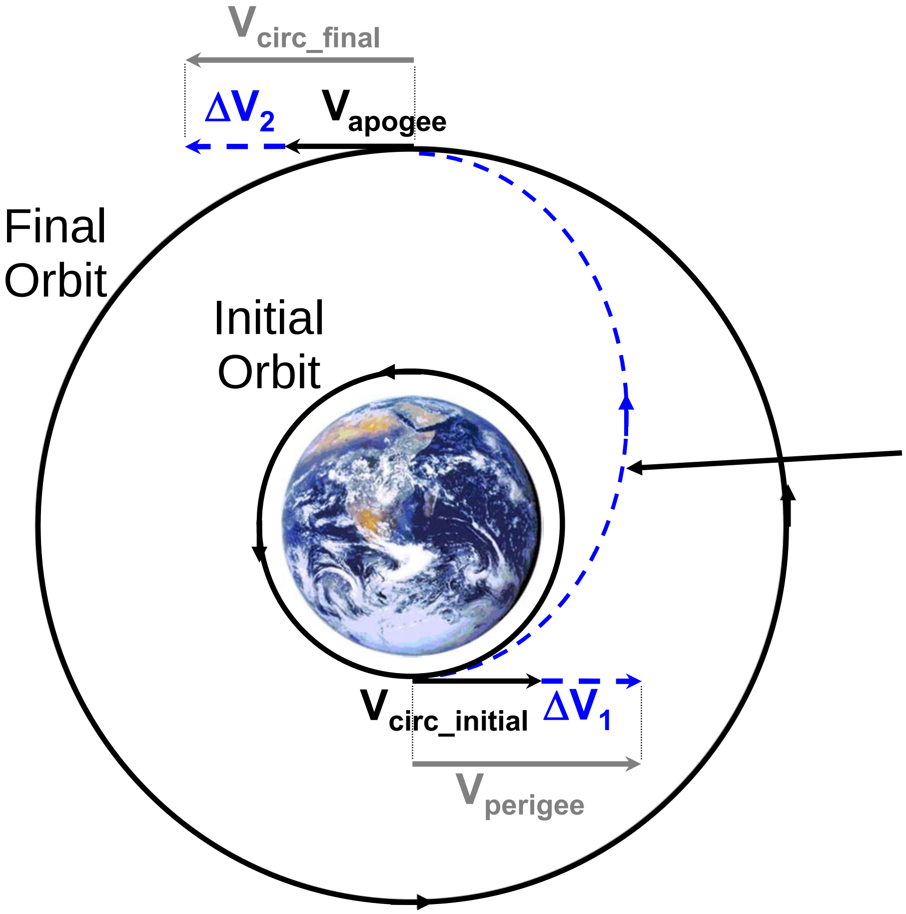
Earth-Moon Transfer Summary

- ❑ Ground launched lunar missions provide daily opportunities
- ❑ Lunar missions departing from an existing fixed orbit provide opportunities only about every 9 days for a 28.5° parking orbit or every 10 days for 51.6°
- ❑ The general ΔV cost for lunar missions is about 3100 m/s for Earth Orbit Departure and about 900 for Lunar Orbit Insertion

Selenocentric Characteristics

Minimum Energy Orbital Transfer

A *high thrust* orbital transfer between the Earth and the Moon with the least fuel requirement traverses a central angle of 180° and has a tangential departure and arrival. This is known as a **Hohmann Transfer**.



OUTBOUND

Initial orbit: Spacecraft (S/C) in circular orbit

$V_{circ\_initial}$

Transfer orbit: S/C at perigee of elliptical orbit

$V_{perigee} = V_{circ\_initial} + \Delta V_1$

Hohmann Transfer

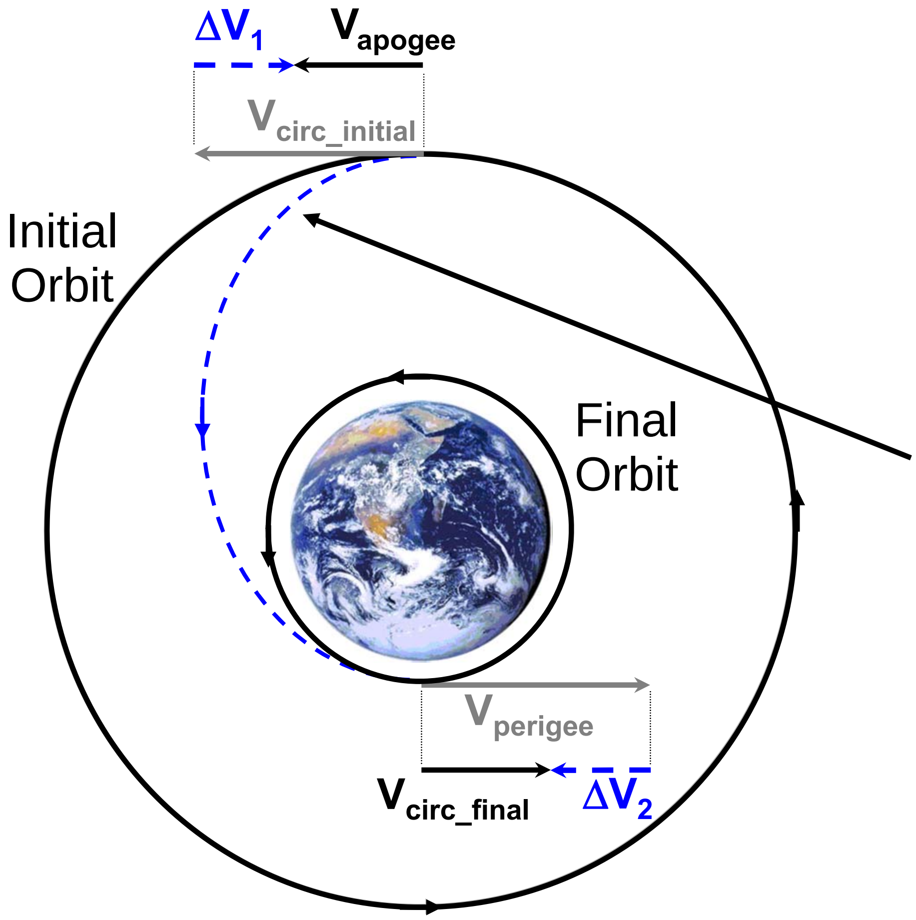
Transfer orbit: S/C at apogee of elliptical orbit

V_{apogee}

Final orbit: Spacecraft in circular orbit

$V_{circ\_final} = V_{apogee} + \Delta V_2$

Minimum Energy Orbital Transfer



INBOUND

Initial orbit: Spacecraft (S/C) in circular orbit

$V_{circ\_initial}$

Transfer orbit: S/C at apogee of elliptical orbit

$V_{apogee} = V_{circ\_initial} - \Delta V_1$

Hohmann Transfer

Transfer orbit: S/C at perigee of elliptical orbit

$V_{perigee}$

Final orbit: Spacecraft in circular orbit

$V_{circ\_final} = V_{perigee} - \Delta V_2$

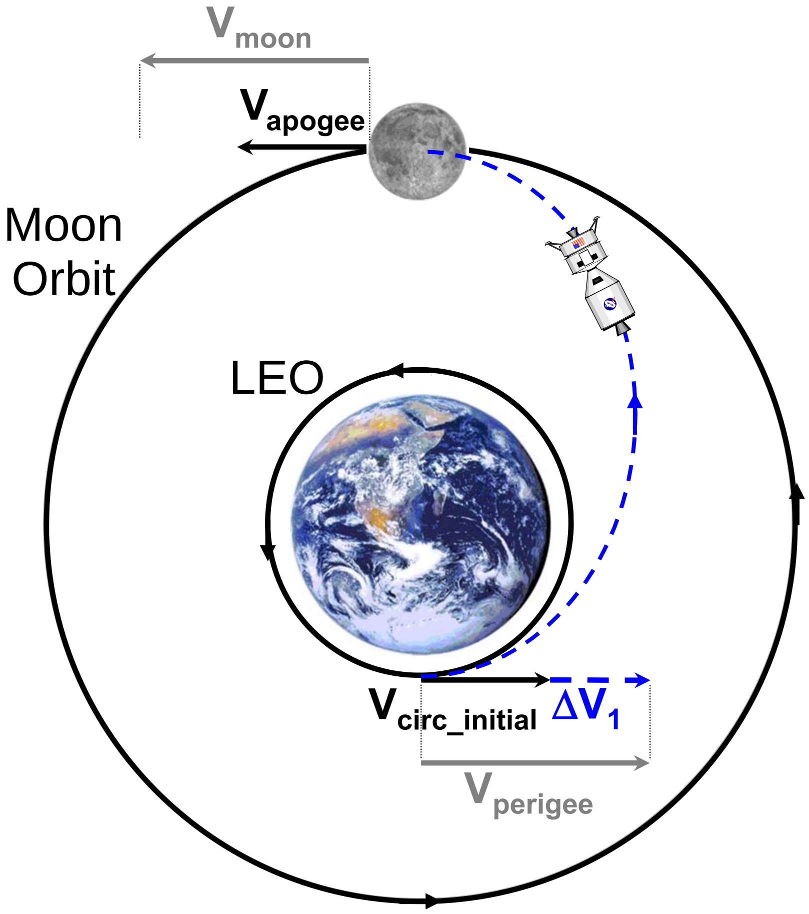
Earth to Moon Transfer

- ☐ In the geocentric reference frame, a delta-velocity maneuver (ΔV_1) in low Earth orbit establishes a Moon intercept transfer ellipse trajectory
- ☐ After coasting from perigee to apogee (at lunar altitude) on the transfer ellipse, the spacecraft (s/c) encounters the Moon (V_{apogee})
- ☐ Since the apogee velocity of the transfer ellipse is slower than the circular lunar orbit velocity (V_{moon}), the Moon overtakes the s/c
- ☐ The difference between V_{moon} and V_{apogee} (of the transfer ellipse) is the lunar approach vector known as V_∞
- ☐ The V_∞ is a measure of the energy per unit mass of a lunar approach hyperbolic trajectory

Earth to Moon Transfer

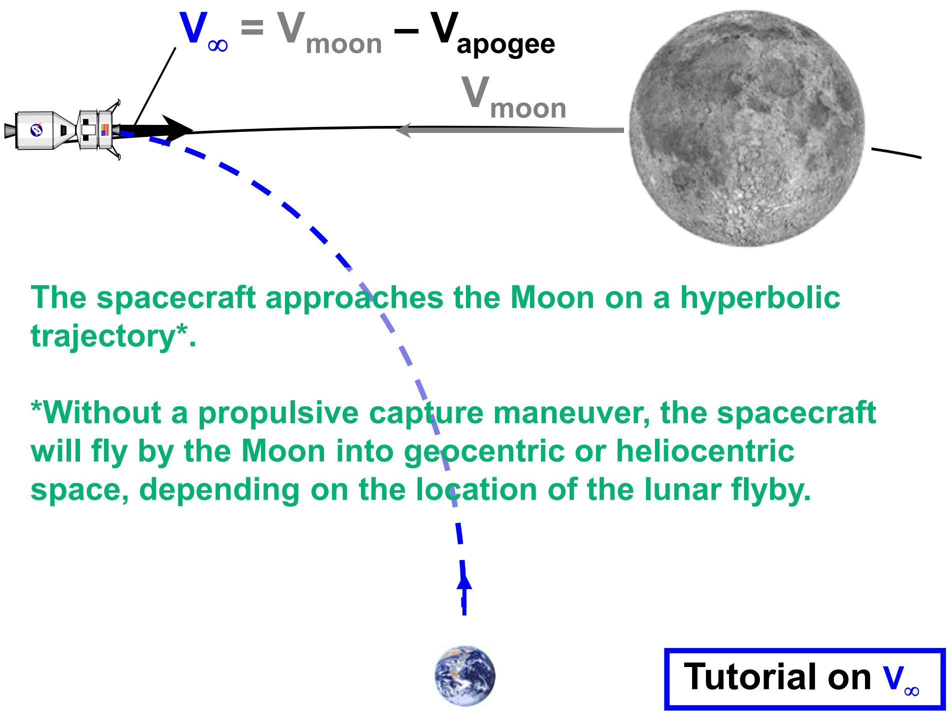
Geocentric Reference Frame

Faster moving Moon (V_{moon}) overtakes slower moving spacecraft (V_{apogee}) at lunar encounter.



Selenocentric Reference Frame

From the perspective of the lunar surface, the spacecraft appears to be approaching from the opposite direction of the Moon's motion at a velocity which is the difference between the Moon's velocity and the spacecraft velocity at the apogee of its transfer orbit.



The spacecraft approaches the Moon on a hyperbolic trajectory*.

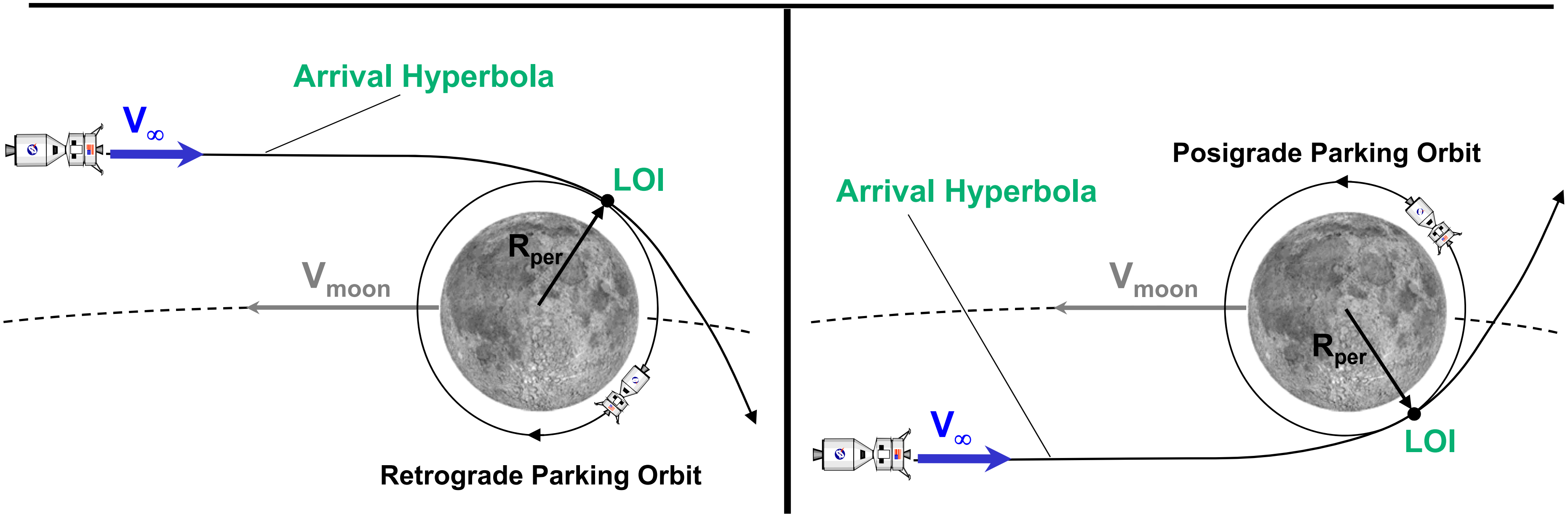
*Without a propulsive capture maneuver, the spacecraft will fly by the Moon into geocentric or heliocentric space, depending on the location of the lunar flyby.

Tutorial on V_∞

LOI: 2-D Seleocentric View

The incoming hyperbola (V_∞) can be adjusted at Earth departure (for a negligible ΔV cost) to poise the arriving spacecraft to perform lunar orbit insertion (LOI) into a posigrade or retrograde lunar parking orbit.

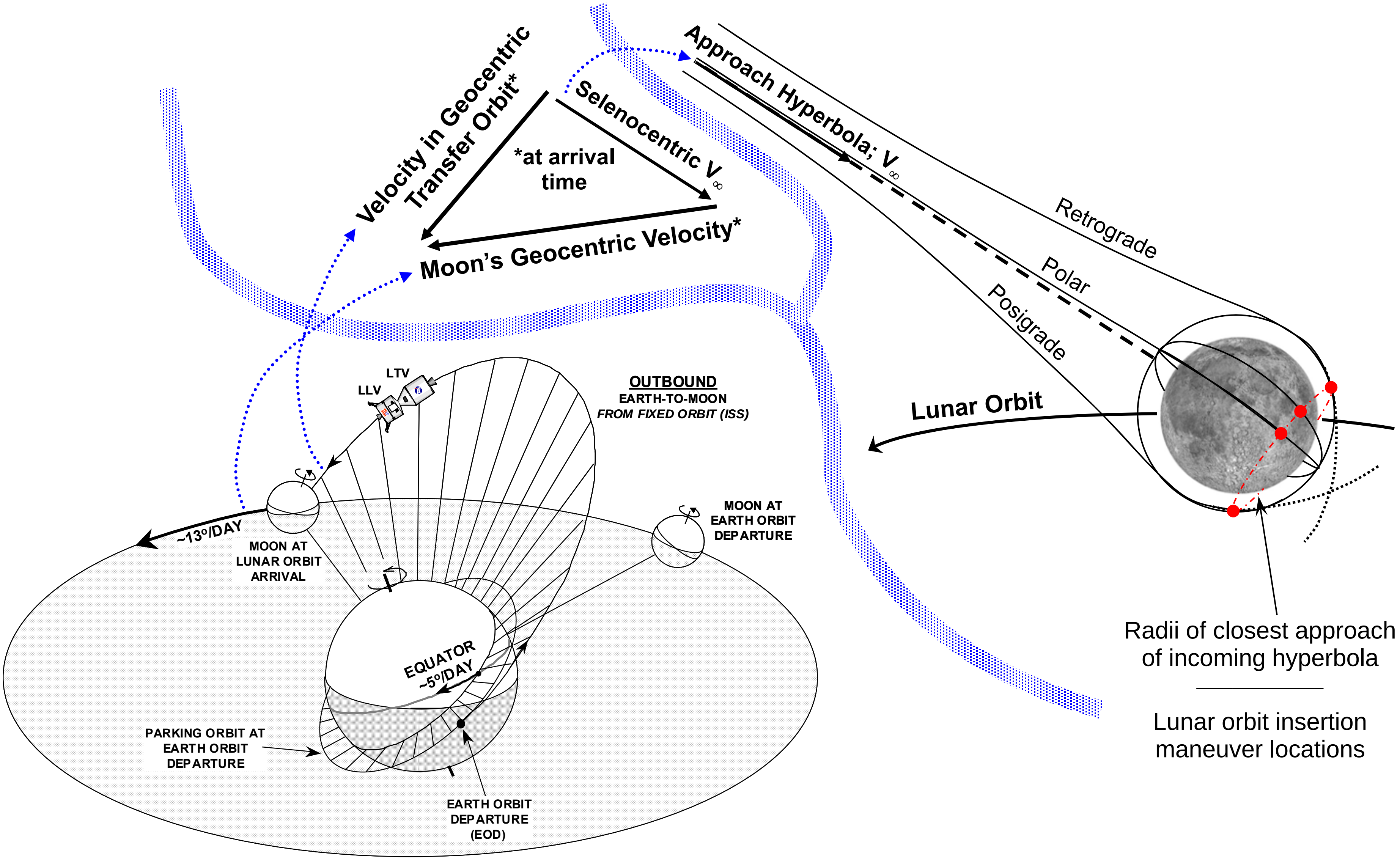
Minimum ΔV LOI occurs at the closest approach to the planet (the periapse radius of the incoming hyperbola, R_{per})



Earth to Moon Transfer

- ❑ In general, the hyperbolic V_∞ approach vector is the vector difference between the geocentric velocity of the spacecraft and Moon's geocentric velocity at the lunar encounter
- ❑ The V_∞ vector can be adjusted at Earth orbit departure, for a negligible ΔV cost, to allow a coplanar LOI to any inclination (greater than or equal to the declination of the incoming V_∞ vector asymptote)
  - ▪ Inclinations lower than the declination of the lunar approach V_∞ vector can also be achieved, but with a required out-of-plane maneuver

Earth to Moon Transfer



Radii of closest approach of incoming hyperbola

Lunar orbit insertion maneuver locations

Lunar Parking Orbits

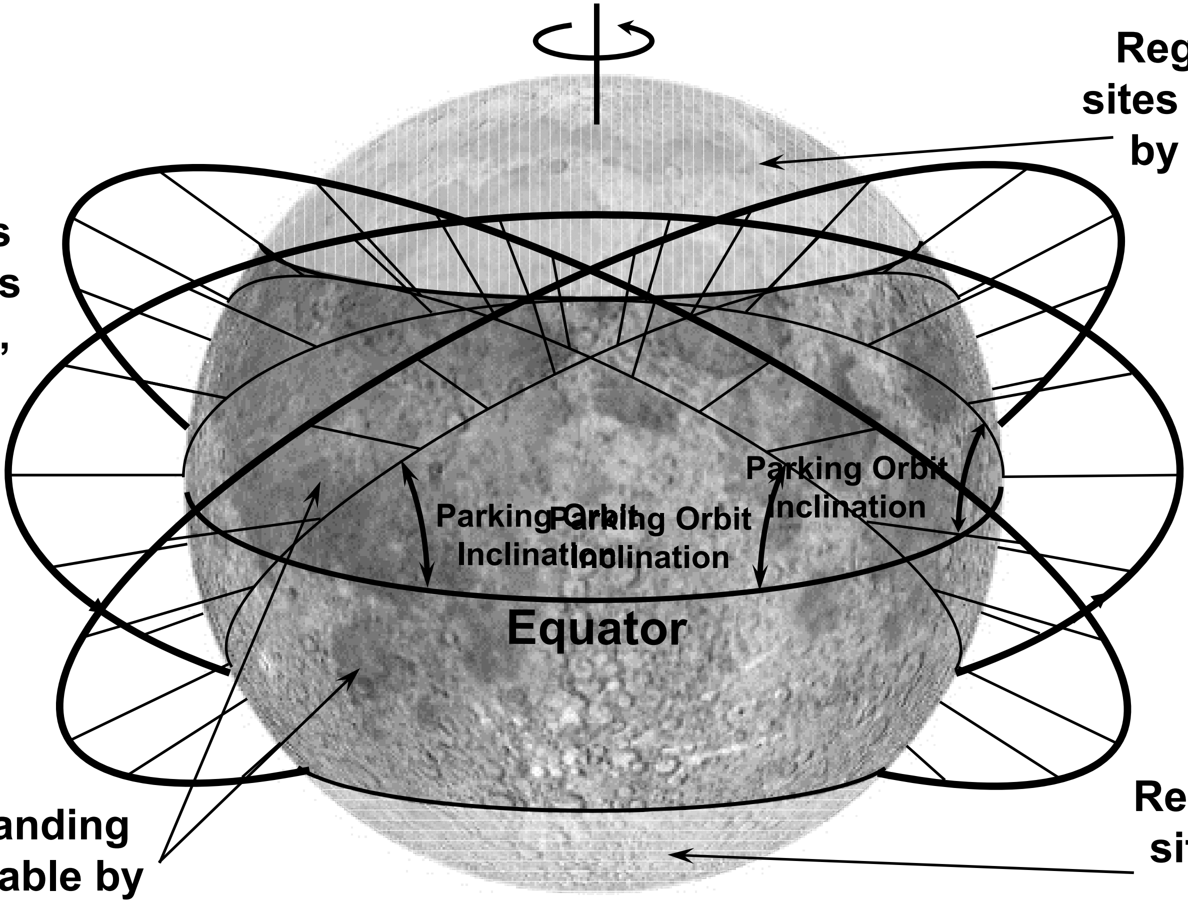
Lunar Parking Orbit Inclination

- ❑ The magnitude of the lunar parking orbit inclination establishes a band of landing site latitudes (equal to the magnitude of the inclination)
  - Within this latitude band, coplanar descent and ascent are possible, given that the rendezvous orbit contains the landing site
  - For landing site latitude magnitudes greater than that of the rendezvous orbit inclination, the descent and ascent are non-coplanar
- ❑ For a given landing site within the latitude band, there are two ascent opportunities every lunar rotation cycle (about 27.3 days)
  - Exception: There is only one opportunity about every 27.3 days when the magnitude of the latitude of the landing site equals that of the rendezvous orbit inclination

Lunar Parking Orbit Inclination

Lunar inclination establishes a band of landing sites achievable with coplanar descent

At least one in-plane lunar ascent & rendezvous available about every 27 days



Region of landing sites achievable only by non-coplanar descent

Rendezvous orbit remains nearly fixed, inertially

Region of landing sites achievable by coplanar descent

Region of landing sites achievable only by non-coplanar descent

Lunar Powered Descent and Landing/Ascent Unplanned Ascent

- ☐ For a given lunar orbit inclination, a coplanar descent/ascent can be achieved to/from any landing site latitude magnitude that is equal to or less than that of the rendezvous orbit inclination
  - Landing can be achieved outside this range, but will be subject to a plane change penalty (i.e., non-coplanar descent or ascent)
- ☐ In general, nominal lunar descent and ascent are coplanar maneuvers
- ☐ Upon nominal landing, the spacecraft moves at a rate equal to the lunar rotation rate (about 360°/27.3 days)
- ☐ An unplanned ascent would require a plane change
  - The magnitude of the plane change will be dependent upon the time of the ascent

Worst Case Lunar Plane Change Cost

Lunar Rendezvous Altitude = 100x100 km

For any lunar orbit inclination, global lunar access dictates that there is a lunar landing site where a 90° plane change could be required

Worst-Case Descent/Ascent Plane Change for Global Lunar Surface Access

Landing Site Latitude	Lunar Parking Orbit Inclination (deg)						
(deg)	0	15	30	45	60	75	90
0	0	15	30	45	60	75	**90**
15	15	30	45	60	75	**90**	75
30	30	45	60	75	**90**	75	60
45	45	60	75	**90**	75	60	45
60	60	75	**90**	75	60	45	30
75	75	**90**	75	60	45	30	15
90	**90**	75	60	45	30	15	0

Worst-Case Descent/Ascent ΔV Cost for Global Lunar Surface Access

Landing Site Latitude	Plane Change ΔV (m/s)						
(deg)	0	15	30	45	60	75	90
0	0	426	845	1250	1633	1988	**2309**
15	426	845	1250	1633	1988	**2309**	1988
30	845	1250	1633	1988	**2309**	1988	1633
45	1250	1633	1988	**2309**	1988	1633	1250
60	1633	1988	**2309**	1988	1633	1250	845
75	1988	**2309**	1988	1633	1250	845	426
90	**2309**	1988	1633	1250	845	426	0

Reference
Coplanar Lunar Ascent Cost

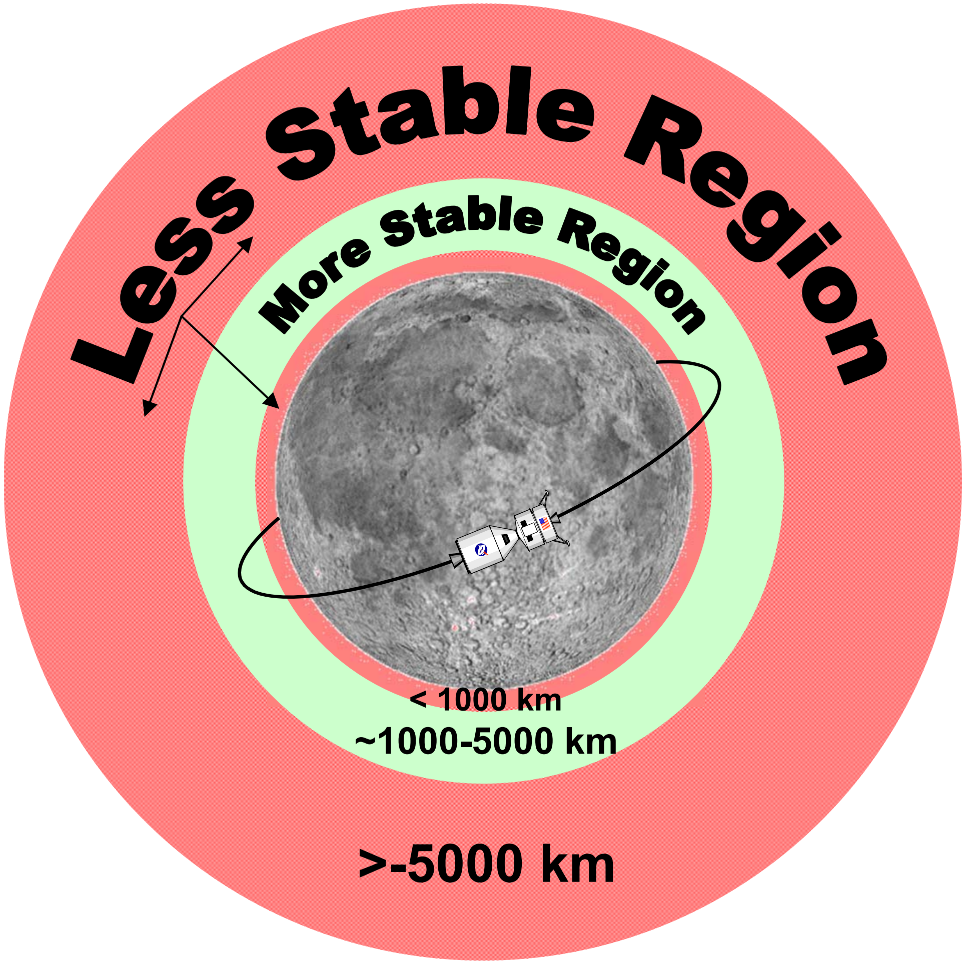
ΔV = 1850 m/s

Target
100x100 km Low Lunar Orbit

Note: The ΔV cost of a 90° plane change is greater than the cost of a coplanar transfer from the lunar surface to a 100x100 km parking orbit.

Lunar Orbit Stability

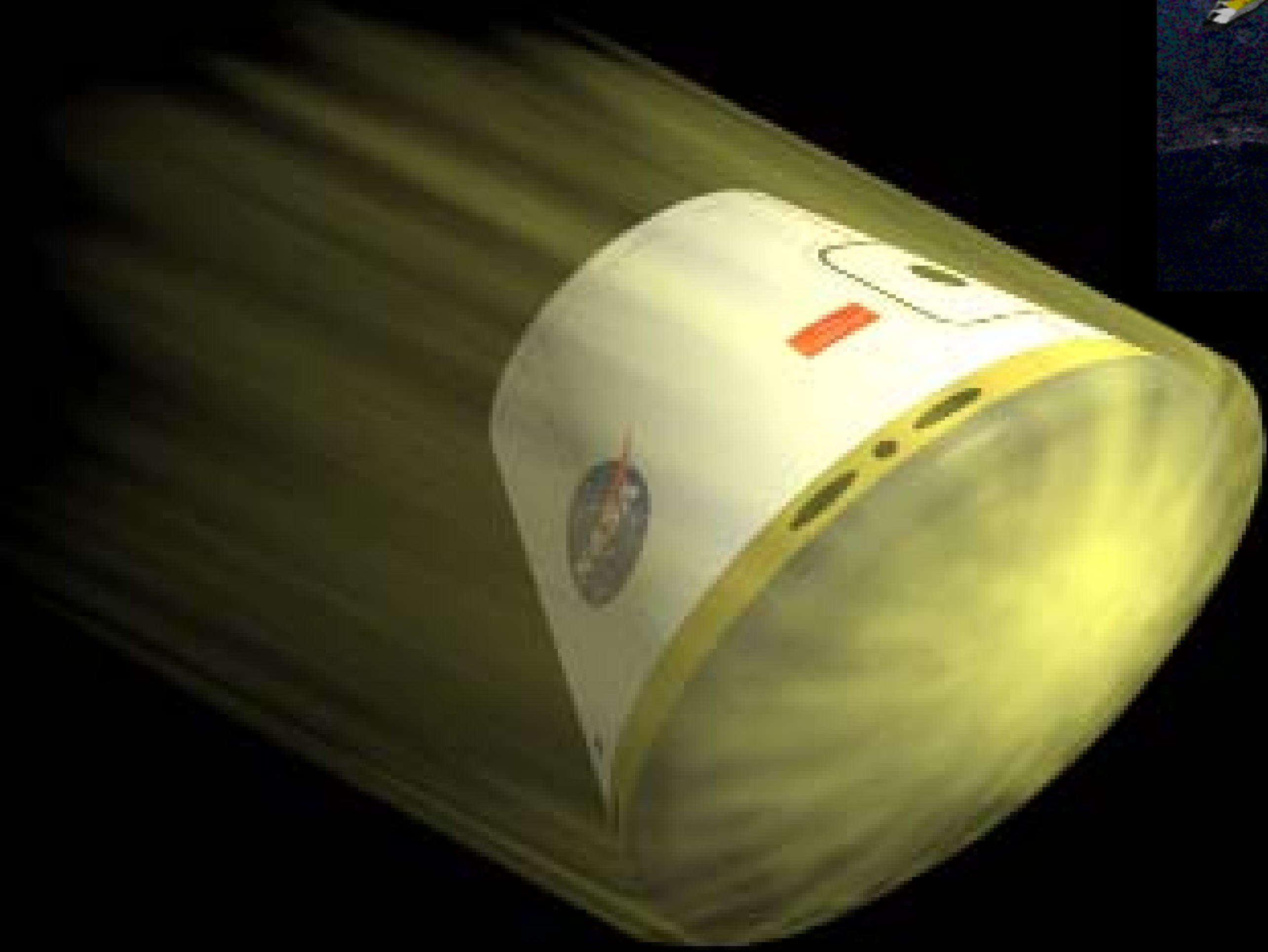
- ☐ Low (circular) lunar orbit altitudes (<~1000 km) are less stable
  - Perturbations to orbit due to non-uniform seleno-potential
- ☐ Higher (circular) lunar orbit altitudes (>~5000 km) are less stable
  - Perturbations to orbit due to Earth and Sun gravity
- ☐ Lunar orbit altitudes in the ~1000 – 5000 km altitude range are more stable

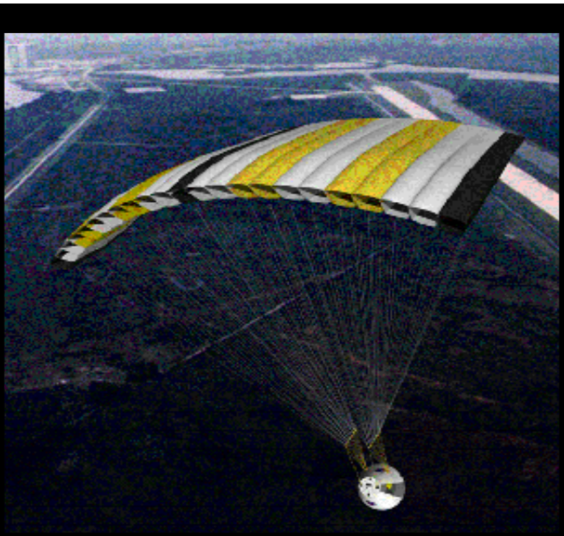


NOT TO SCALE

More …

Earth Return





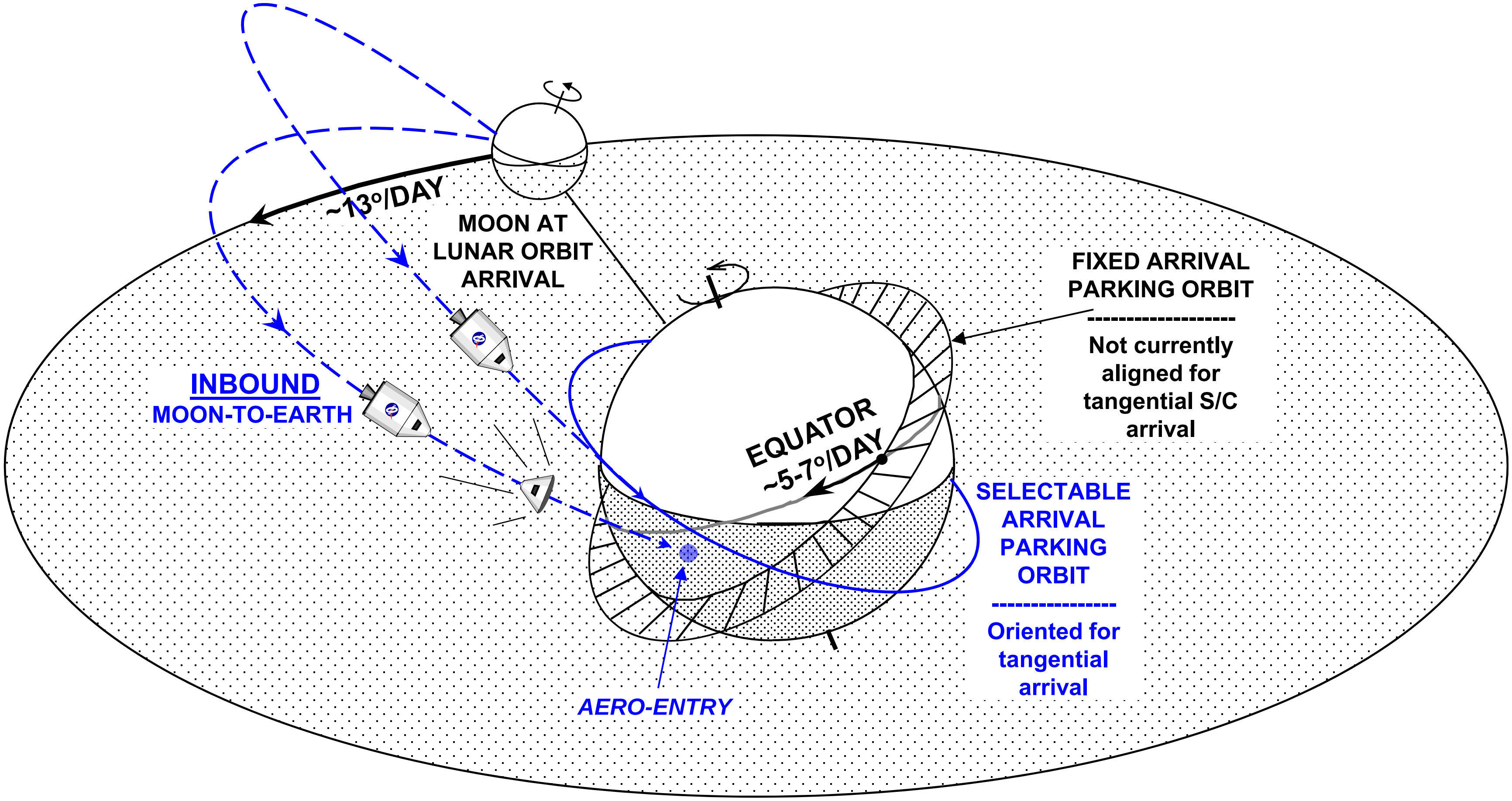
Earth Return and Landing Site Options

- ☐ Major influences on the Moon to Earth trajectory design
  - Lunar departure date
  - Moon to Earth flight time
  - Inclination of the transfer orbit from the Moon to Earth
- ☐ Earth Return Options
  - Direct Entry
    - Lighting conditions at landing depends on the Lunar departure date
    - Limited range of accessible latitudes varies with the departure date
    - Range of accessible longitudes varies with the Moon to Earth flight time
  - Intermediate Low Earth Orbit
    - Insert into orbit using either propulsion or aerobrake
    - Plane change may be required to insert into a specific orbit
    - Orbit inclination is greater than or equal to the landing site latitude
    - Loitering in orbit may be required to target a landing site
- ☐ Landing Site Location
  - Land landing
    - Requires both primary and alternate sites
    - Sites at various latitudes will be required for the direct entry option
  - Water landing
    - Requires ship fleet(s)

Earth Return

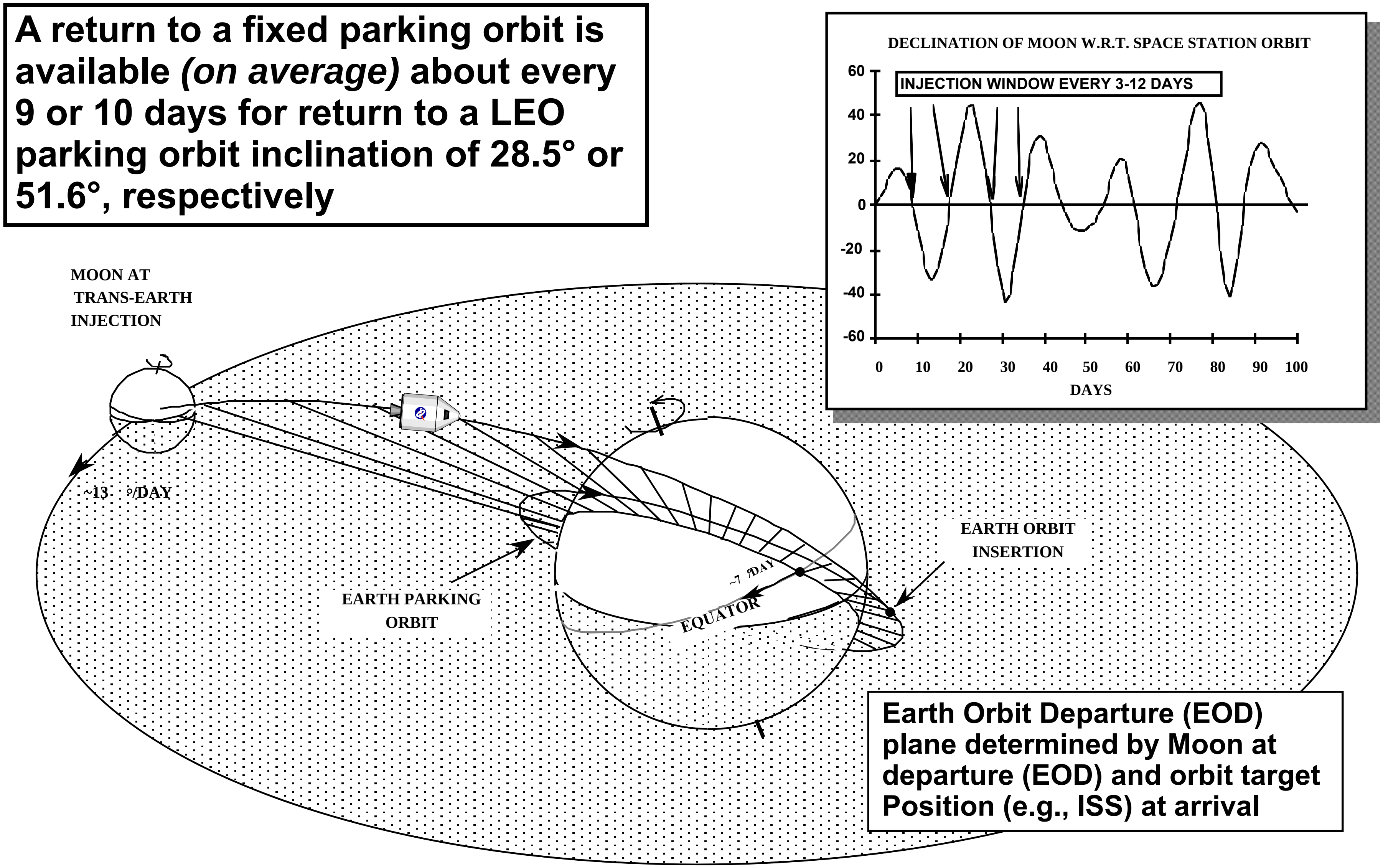
A direct (to surface) return as well as a return to a selectable LEO parking orbit is always available given adequate lunar departure capability

Return to a *fixed* Earth parking orbit requires proper orientation of that orbit



Earth Return: Fixed Orbit Arrival

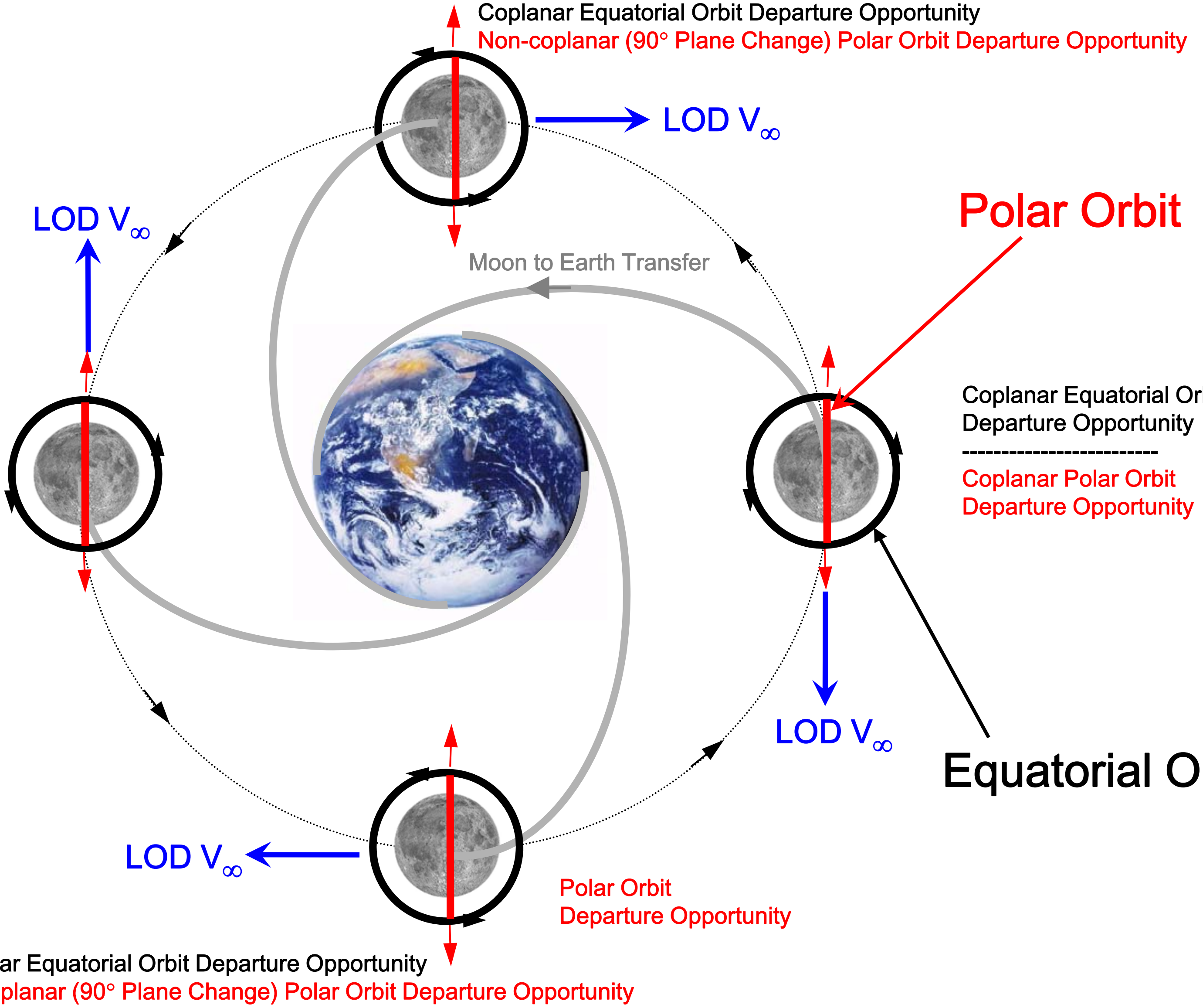
A return to a fixed parking orbit is available *(on average)* about every 9 or 10 days for return to a LEO parking orbit inclination of 28.5° or 51.6°, respectively



Earth Orbit Departure (EOD) plane determined by Moon at departure (EOD) and orbit target Position (e.g., ISS) at arrival

Modified from original. Original courtesy K. Joosten.

Effect of Lunar Parking Orbit Inclination on Lunar Transfer Opportunities –> Moon to Earth Transfer



Coplanar Equatorial Orbit Departure Opportunity
Non-coplanar (90° Plane Change) Polar Orbit Departure Opportunity

LOD V_∞

Polar Orbit

LOD V_∞

Moon to Earth Transfer

Coplanar Equatorial Orbit
Departure Opportunity

Coplanar Polar Orbit
Departure Opportunity

Coplanar Equatorial Orbit
Departure Opportunity

Coplanar Polar Orbit
Departure Opportunity

LOD V_∞

Equatorial Orbit

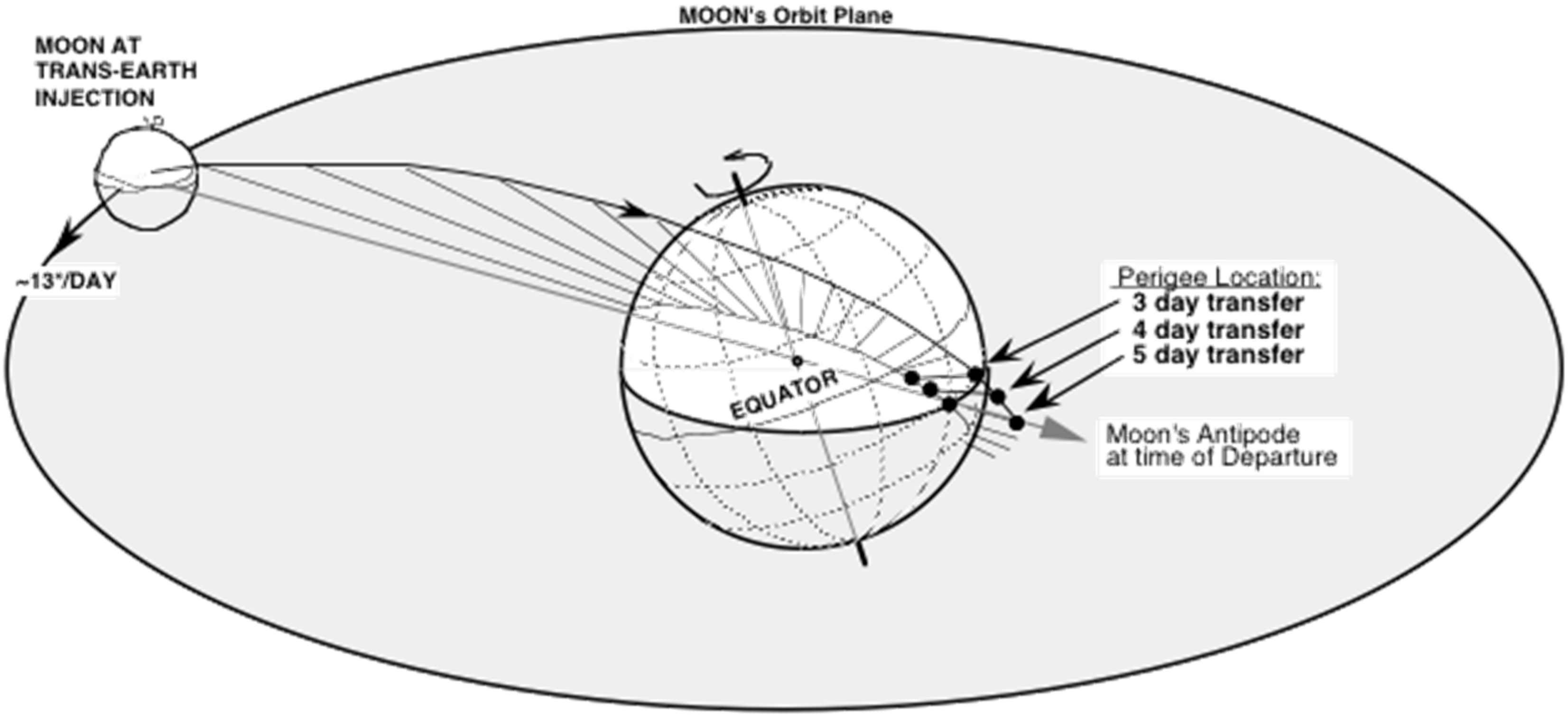
LOD V_∞

Polar Orbit
Departure Opportunity

Coplanar Equatorial Orbit Departure Opportunity
Non-coplanar (90° Plane Change) Polar Orbit Departure Opportunity

Earth Return: Trip Time vs Arrival Location

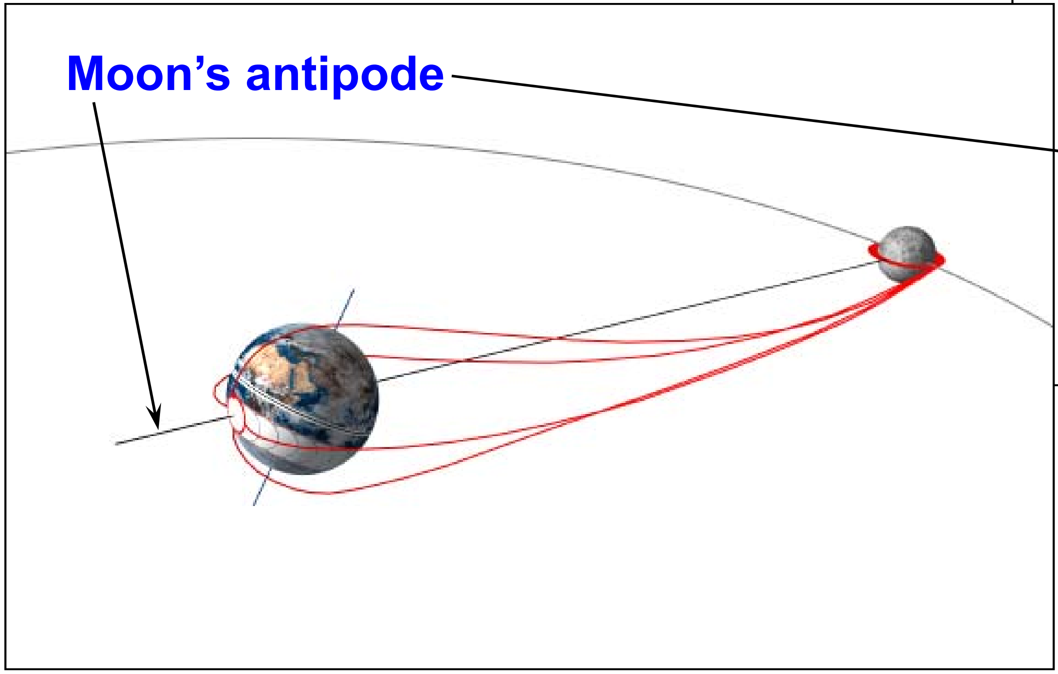
For a given inbound (Earth return) trajectory, a variation in trip time provides some movement of the Earth arrival perigee location



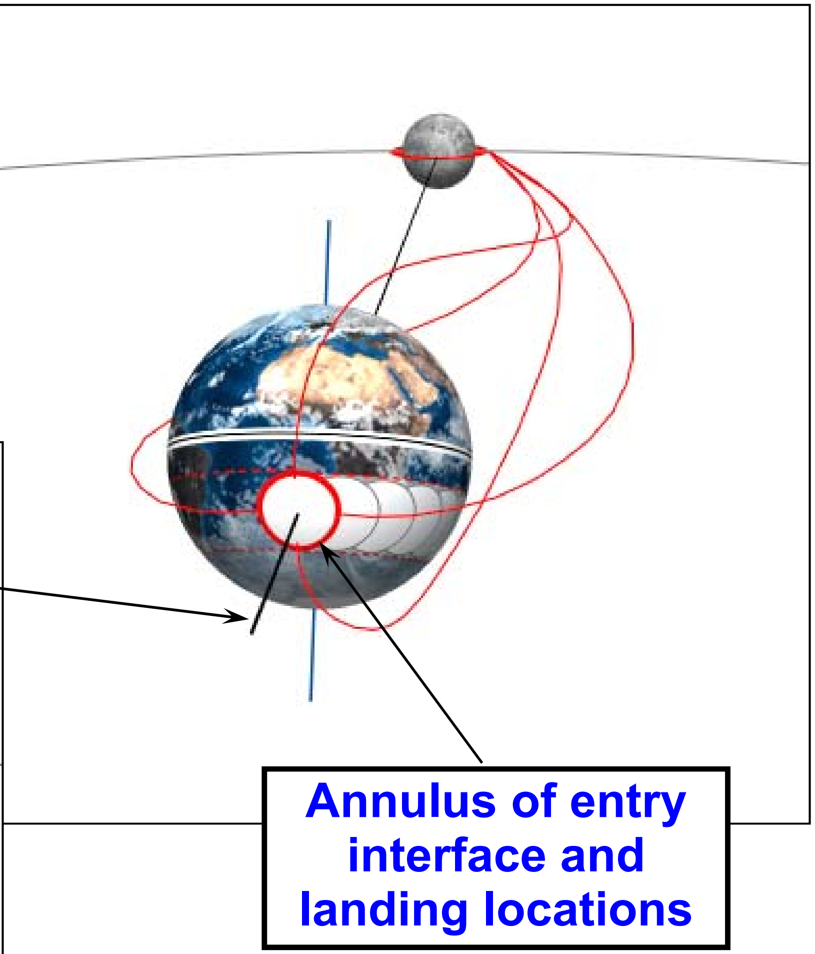
Earth Return: Earth Arrival

The time of lunar departure determines the location of the Moon’s antipode

For a given trip time with a negligibly small ∆V adjustment at lunar departure, the incoming (Moon to Earth) entry interface and landing points can be rotated about the Moon’s antipode



Moon’s antipode

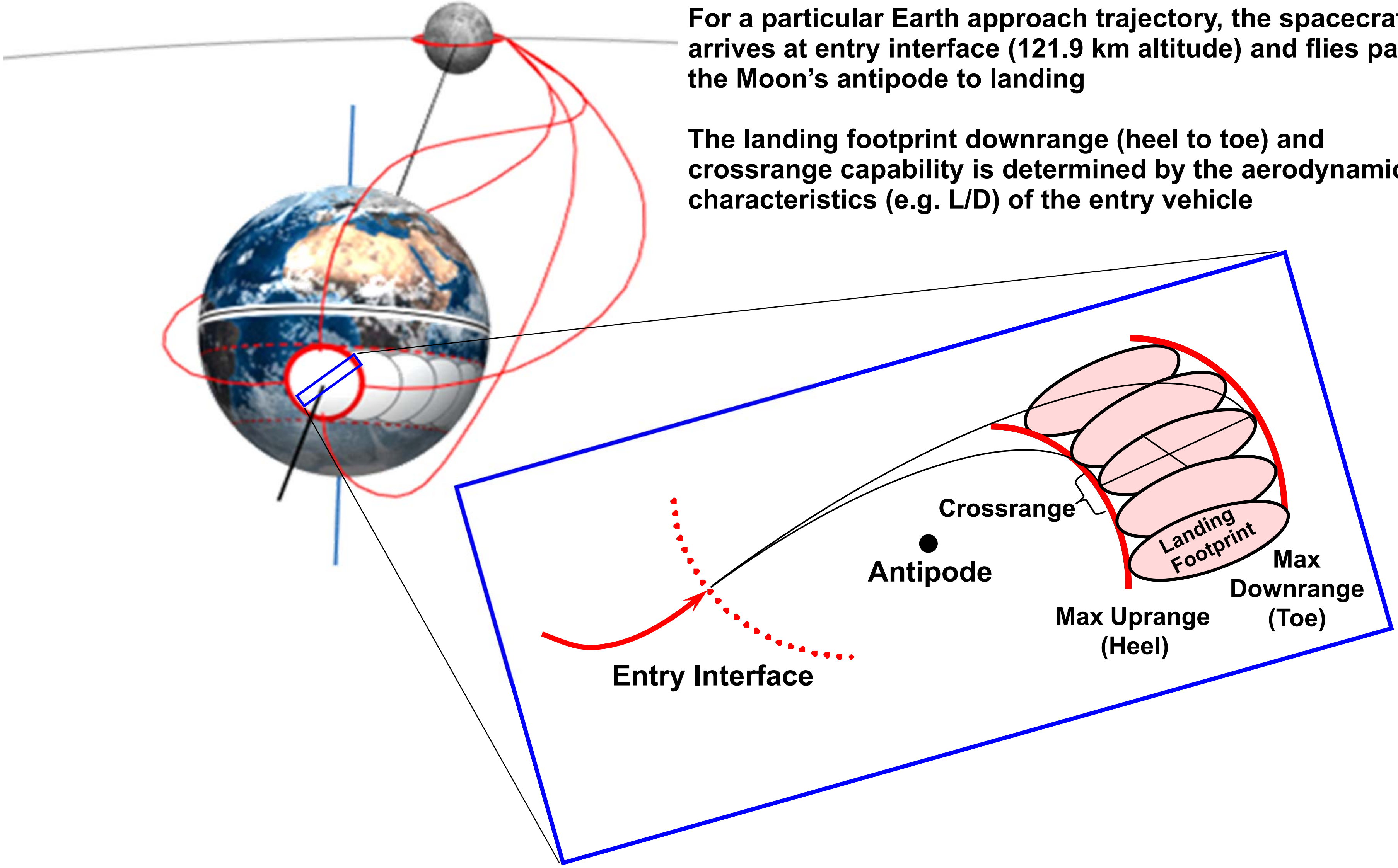


Annulus of entry interface and landing locations

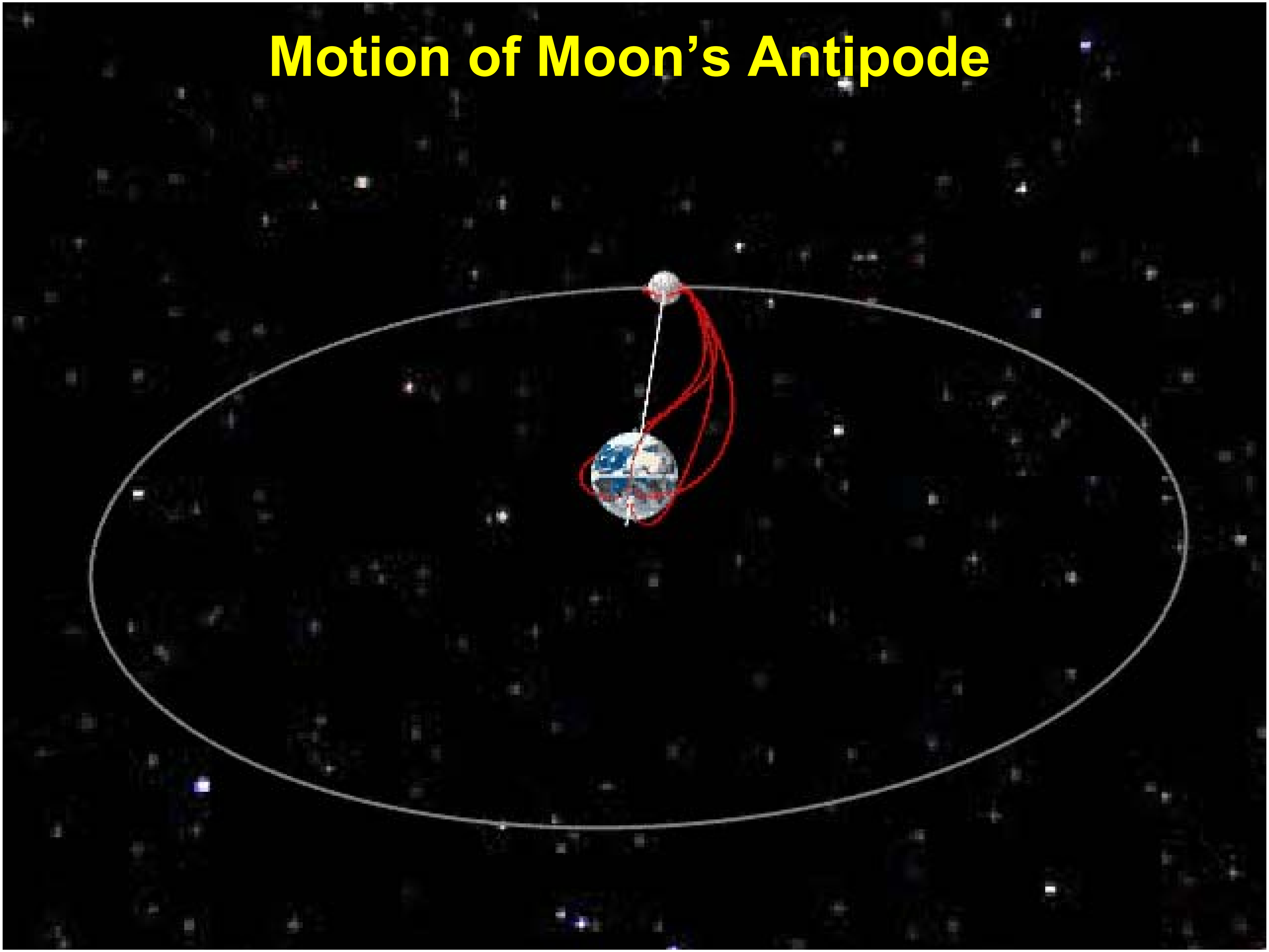
Earth Return: Earth Arrival

For a particular Earth approach trajectory, the spacecraft arrives at entry interface (121.9 km altitude) and flies past the Moon's antipode to landing

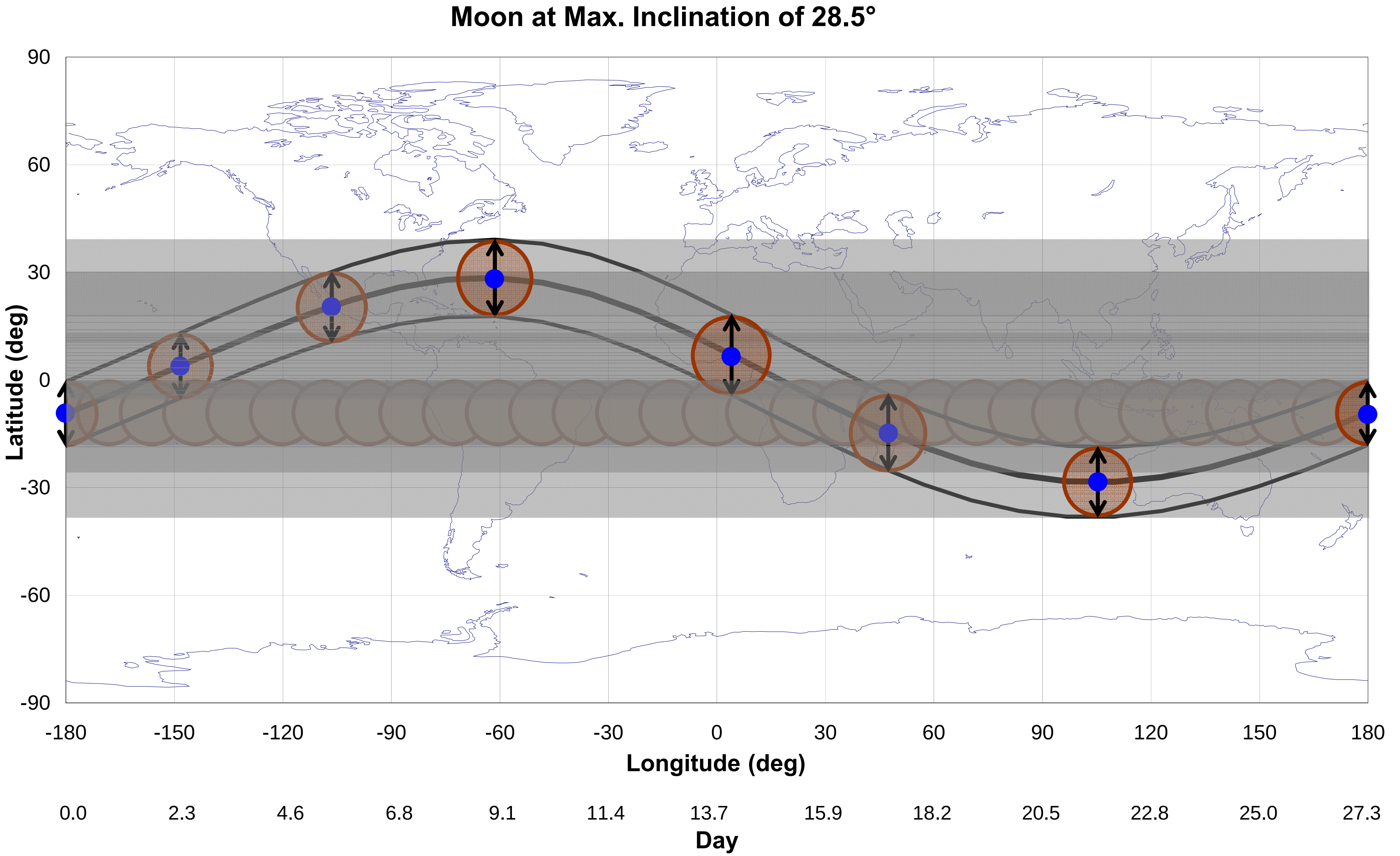
The landing footprint downrange (heel to toe) and crossrange capability is determined by the aerodynamic characteristics (e.g. L/D) of the entry vehicle



Motion of Moon’s Antipode



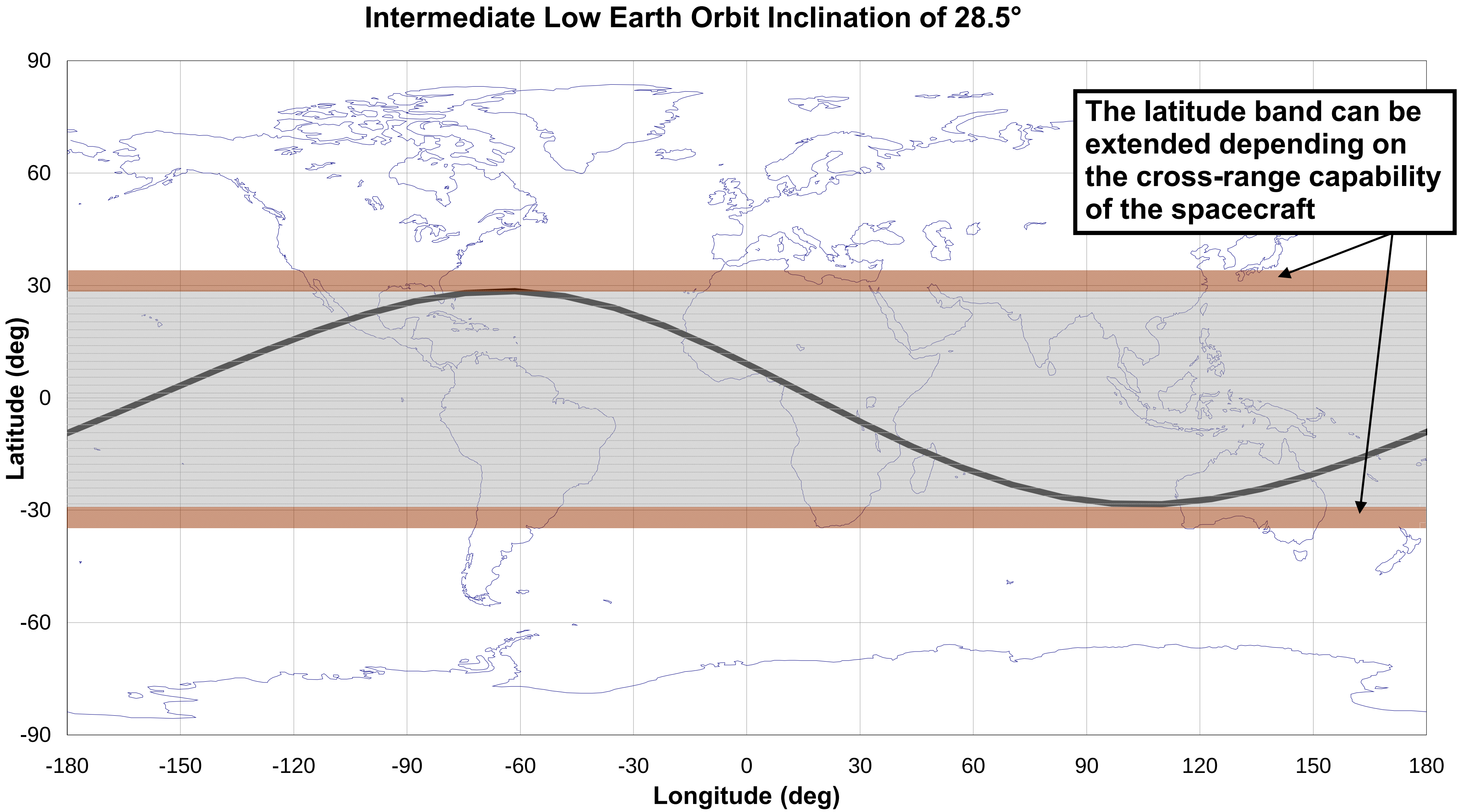
Variation in Accessible Latitudes during a Sidereal Month (Direct Entry)



Accessible Latitudes from Intermediate Low Earth Orbit

- A return to a LEO parking orbit provides a landing latitude band equal to the magnitude of the arrival inclination
  - For example: Arrival into a 28.5° parking orbit provides a 57° latitude band (- 28.5° to + 28.5°)
- This latitude band covers 360° of longitude
- Any (land or water) landing site within the latitude band is accessible provided:
  - Adequate on-orbit loiter time (about 24 hours min.)
  - Adequate spacecraft cross-range capability

Accessible Latitudes from Intermediate Low Earth Orbit



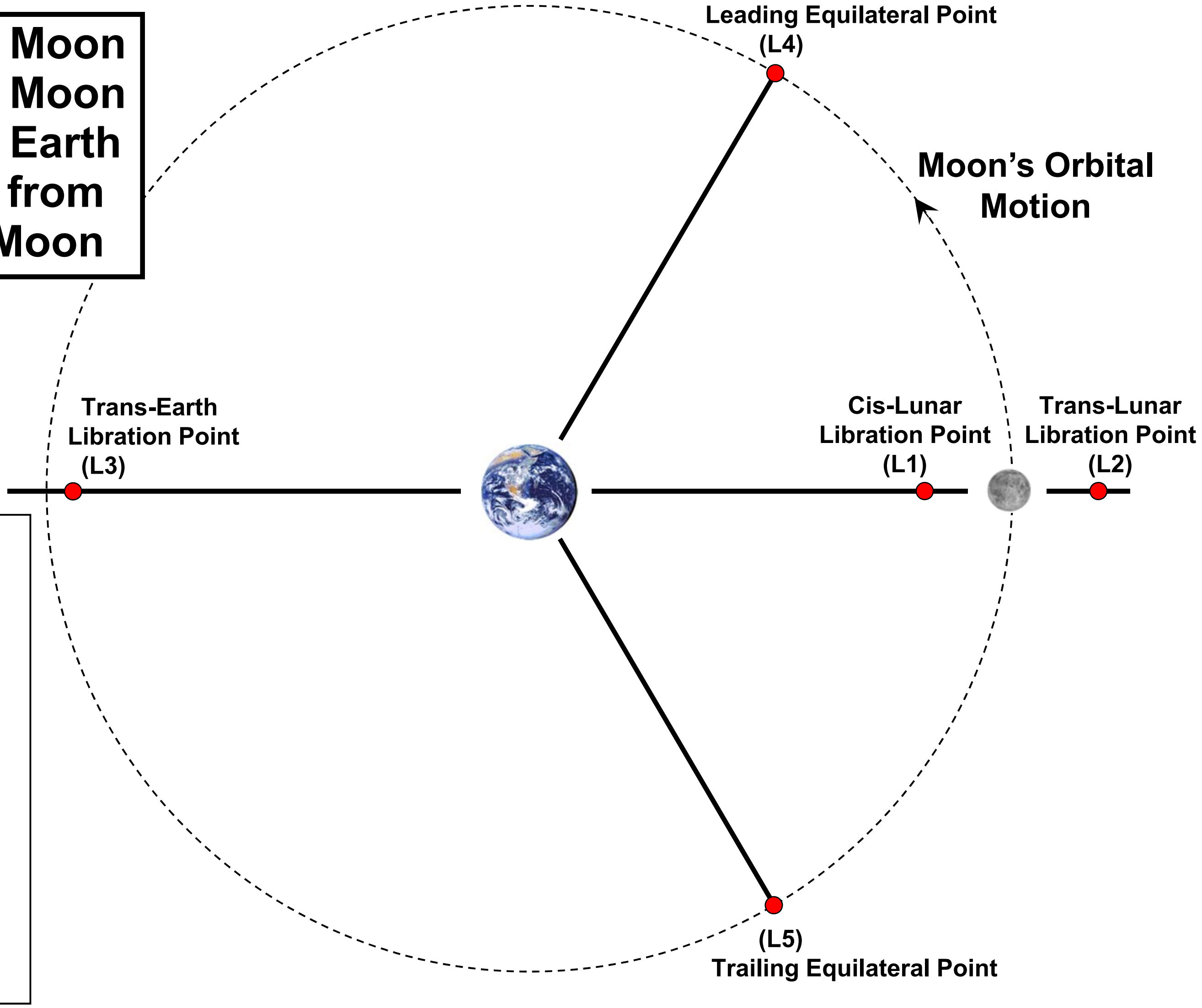
The latitude band can be extended depending on the cross-range capability of the spacecraft

Note: May require some loitering time for groundtrack to intersect with desired landing site

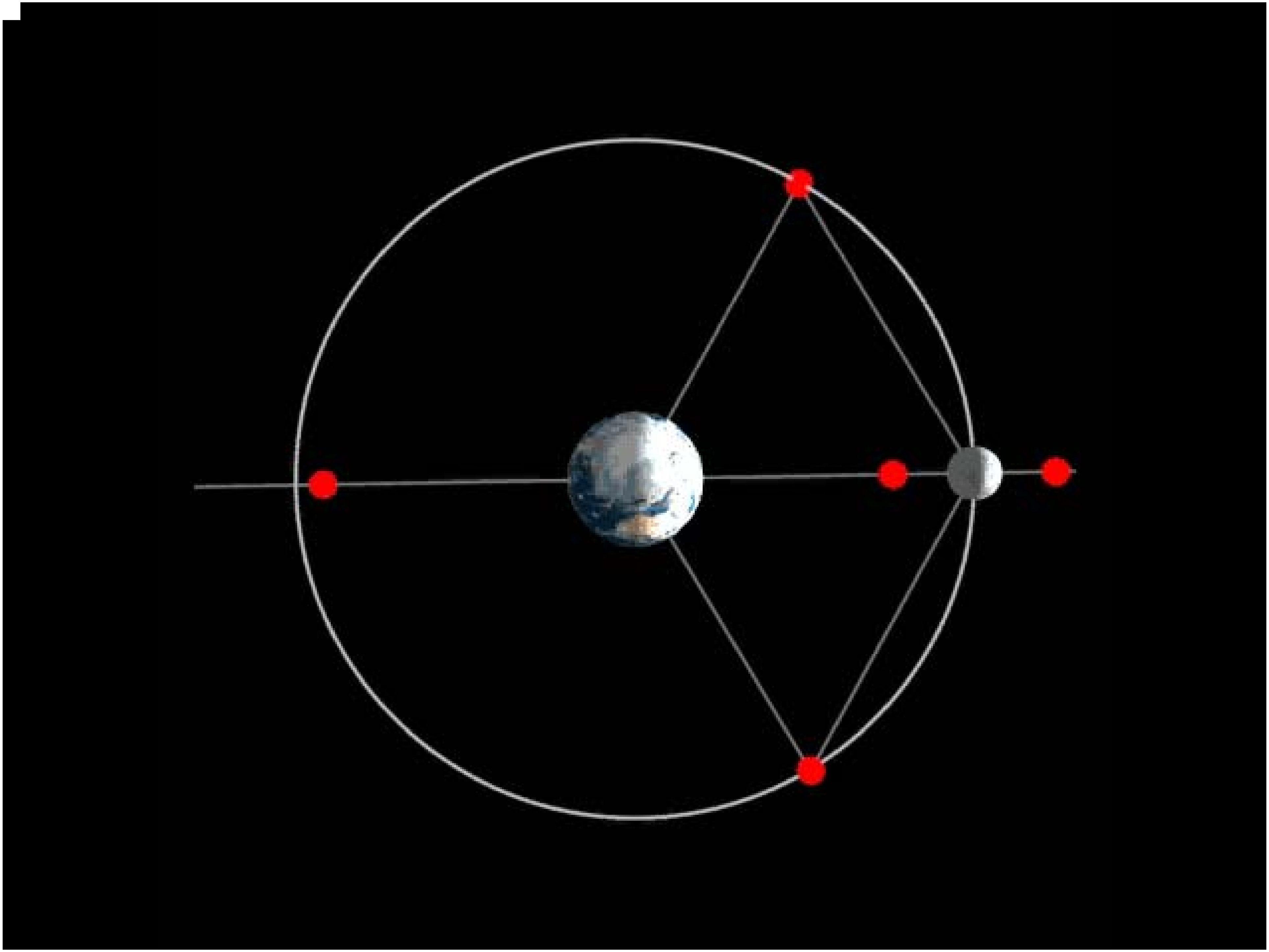
Libration Points

Earth-Moon Libration Points

L1 – 57,731 km from Moon
L2 – 64,166 km from Moon
L3 – 381,327 km from Earth
L4 & L5 – 384,400 km from Earth and Moon



- **Equilibrium points in a 2-body system**
  - **Earth-Moon, Sun-Earth, Sun-Mars, etc.**
- **Collinear points (L1, L2, L3), "unstable"**
- **Equilateral points (L4, L5), "stable"**
- **Station-keeping achievable for very small ΔV (<10 m/s/yr)**



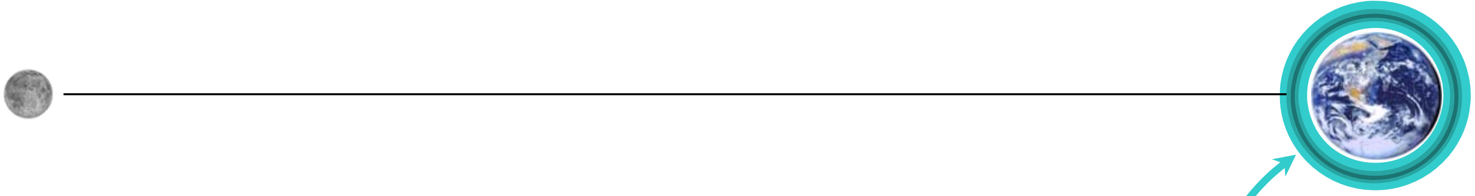
Libration Points

- ☐ Possible staging point for robotic and human missions
  - Lunar Gateway Mission for lunar sorties

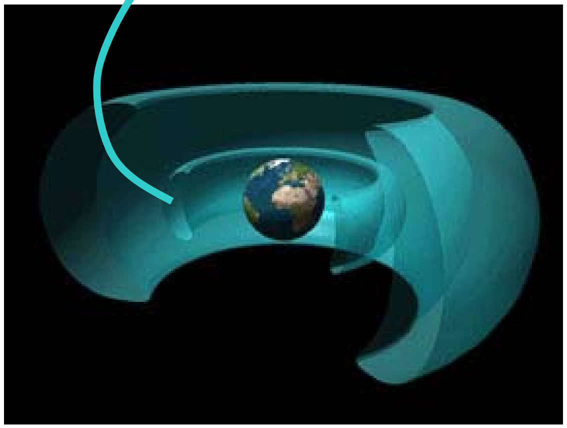
  - Possible telescope (e.g. Webb Telescope, NGST) deploy/maintenance point
    - Minimal ΔV transfer costs between Earth-Moon and Sun-Earth libration points

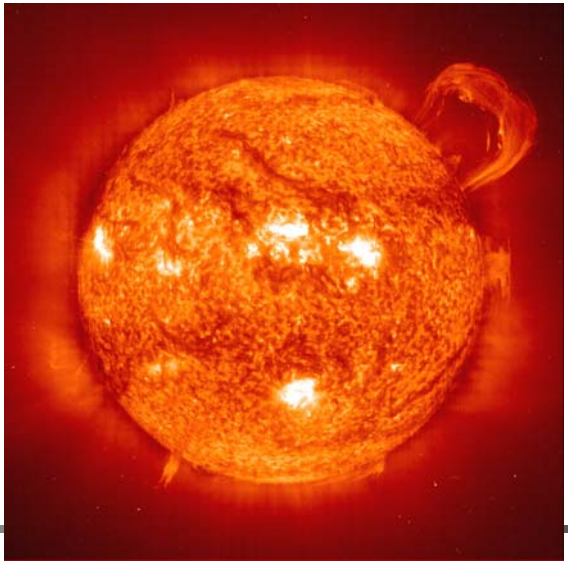
Environment Considerations

Radiation

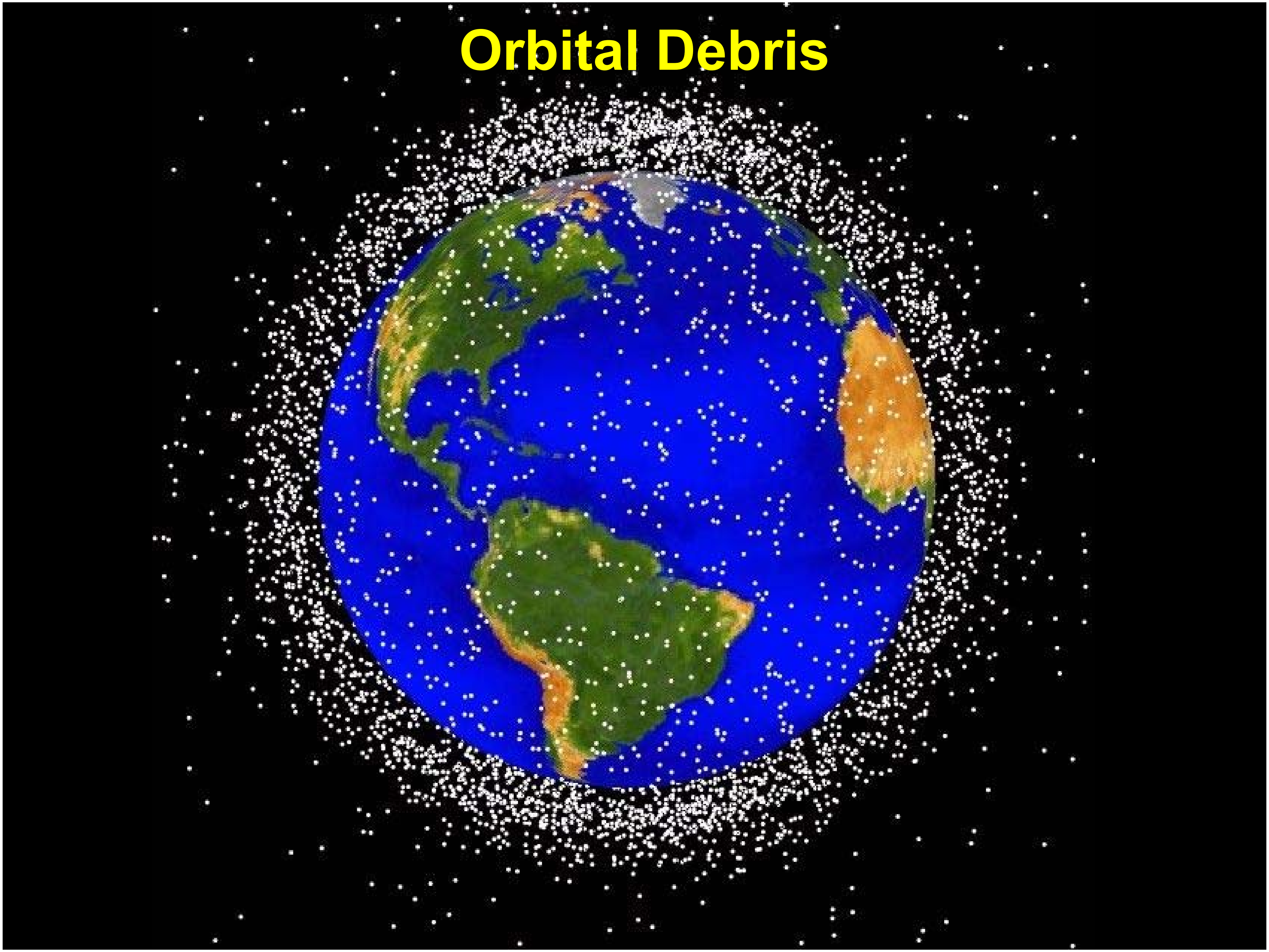


- Van Allen Radiation Belts **More …**
  - Inner (proton) belt 1,000-12,000 km altitude
  - Outer (electron) belt 19,000-57,000 km altitude
- Solar Flares
  - Can cause ionization damage and single-event effects in sensitive devices
  - Energetic protons reach Earth within 30 minutes
  - Other solar materials and magnetic fields reach Earth in 1 to 4 days
- Solar Flux
  - 9 to 13 year cycle
  - Proton energy range from 10 MeV to 1 GeV
- Galactic Cosmic Ray (GCR) Flux
  - Causes single-event effects in sensitive devices
  - Peaks around solar minimum
  - Particle energy up to and over 10 GeV





Orbital Debris



End Part 1

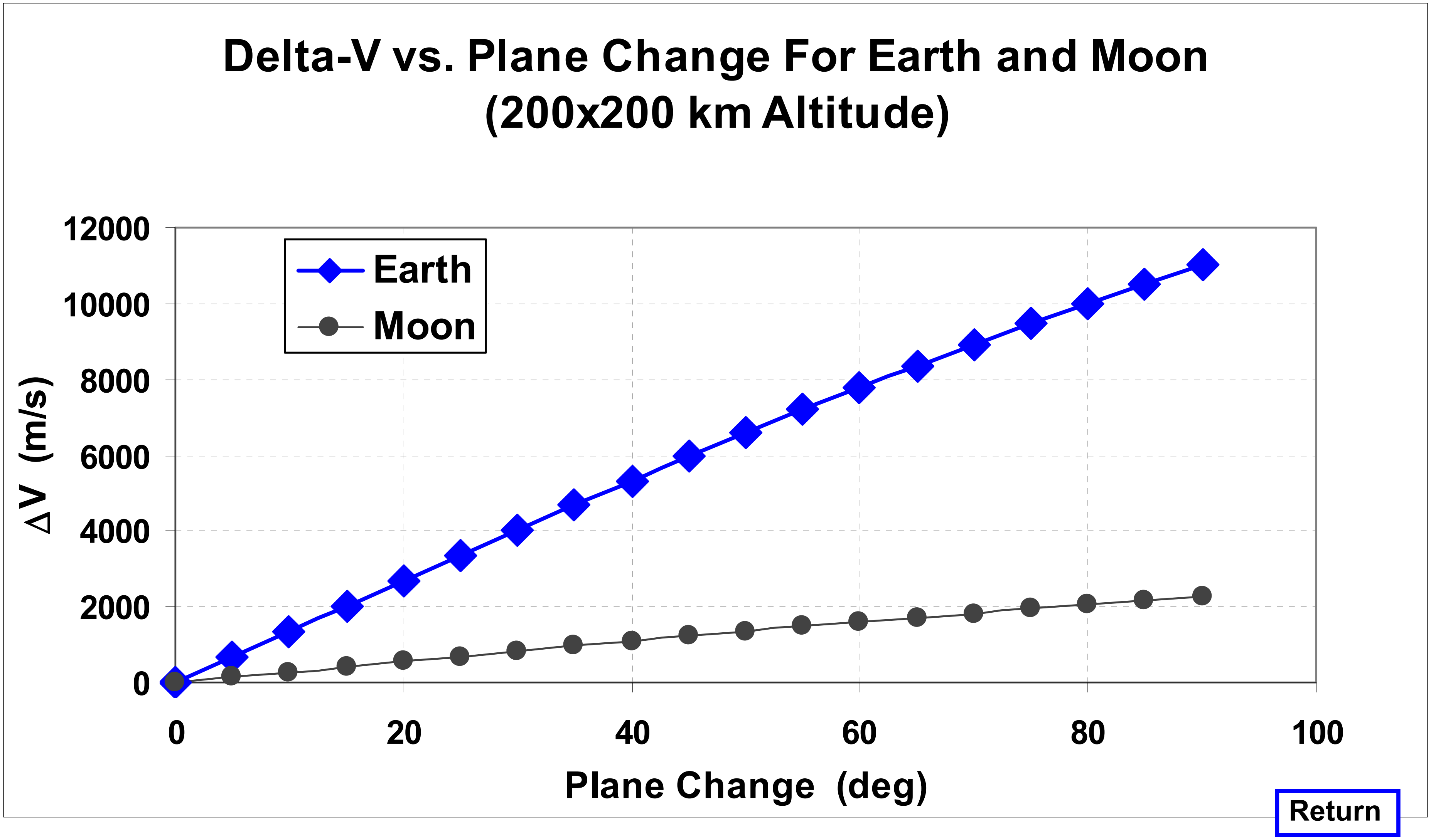
Lunar Orbit Mechanics Tutorial

Back Up Charts for:

Part 1

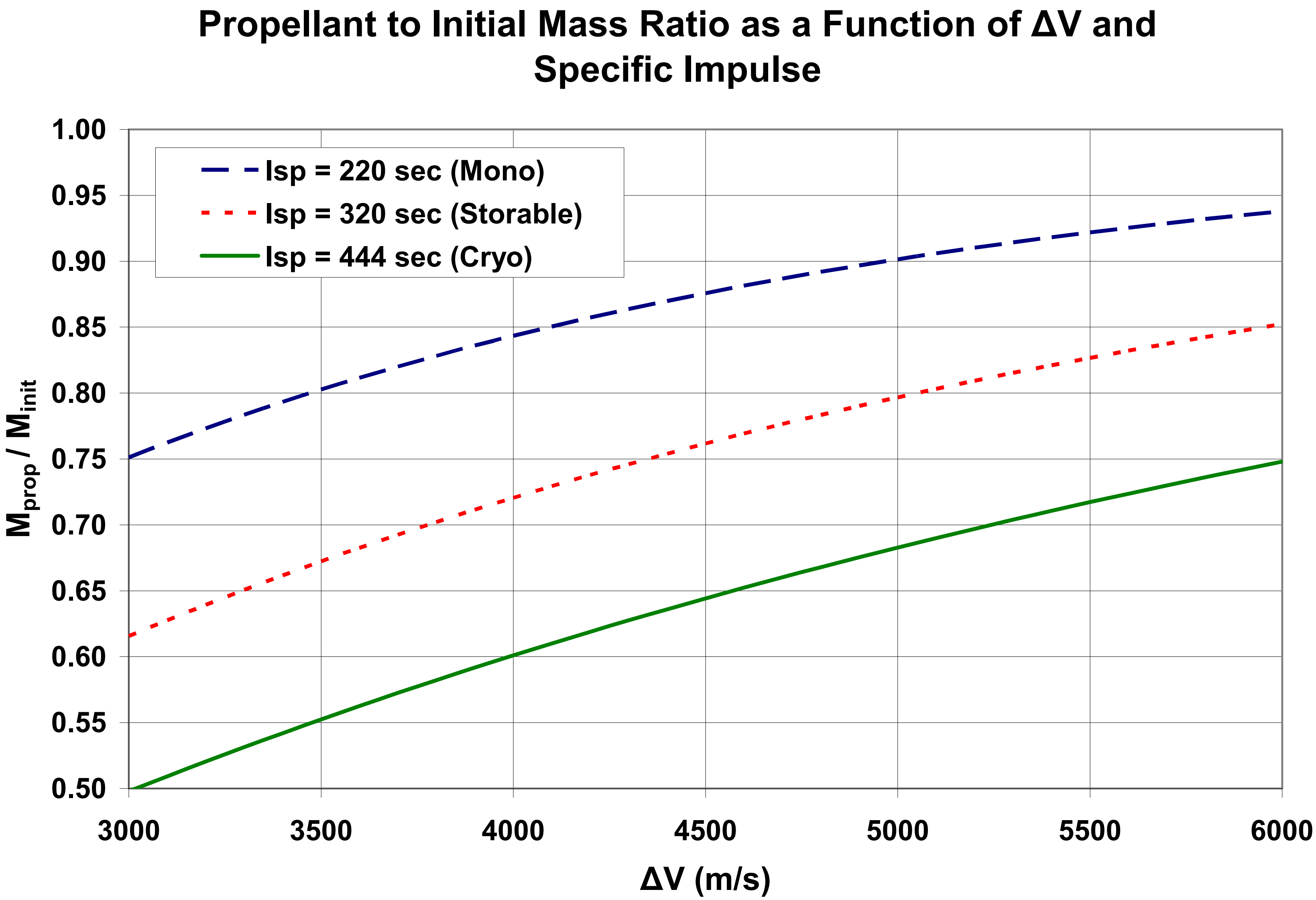
Lunar Orbit Mechanics Tutorial

On-Orbit Plane Changes
Earth and Moon



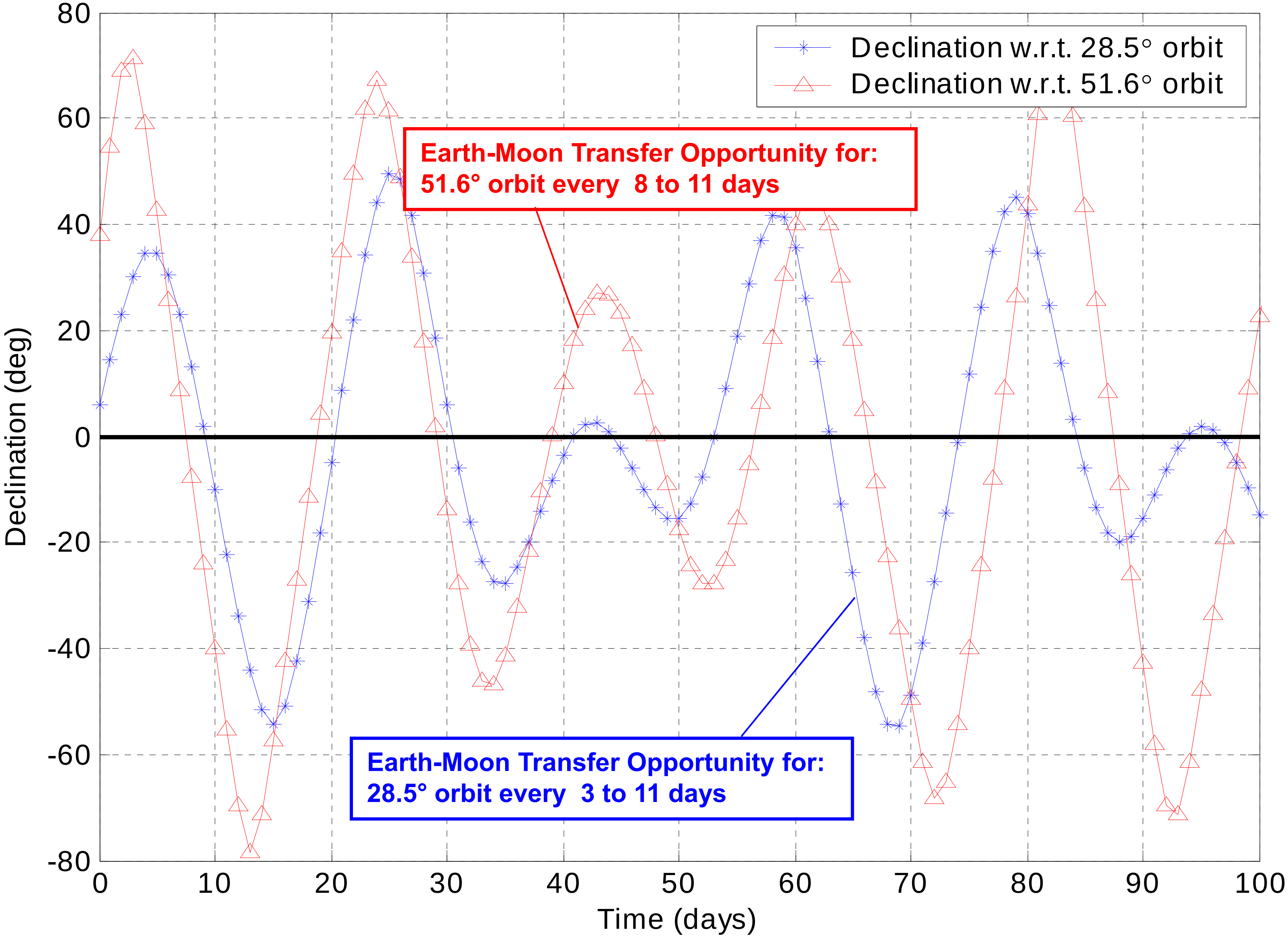
Return

Mass Ratio versus ΔV



Declination of Moon At Arrival w.r.t. Fixed Departure Plane

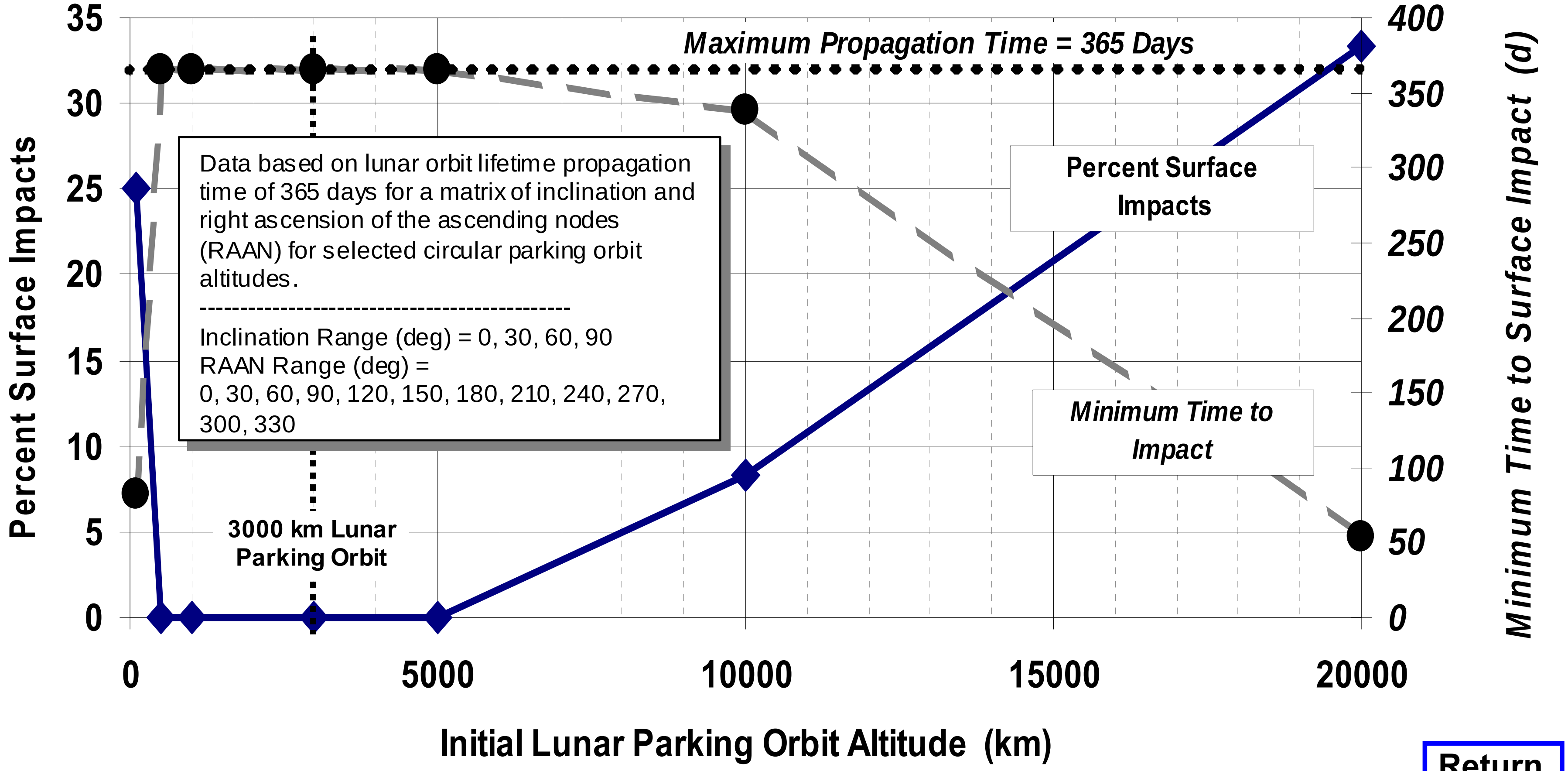
Departure Planes: 51.6° ISS and 28.5° Construction Orbit
Time = 0 at Jan 9, 2009, RAAN = 0°, Altitude = 407x407 km



Earth-Moon Transfer Opportunity for:
51.6° orbit every 8 to 11 days

Earth-Moon Transfer Opportunity for:
28.5° orbit every 3 to 11 days

Lunar Orbit Stability



Percentage of Impacts and Minimum Time to Impact vs. Initial Lunar Parking Orbit Altitude

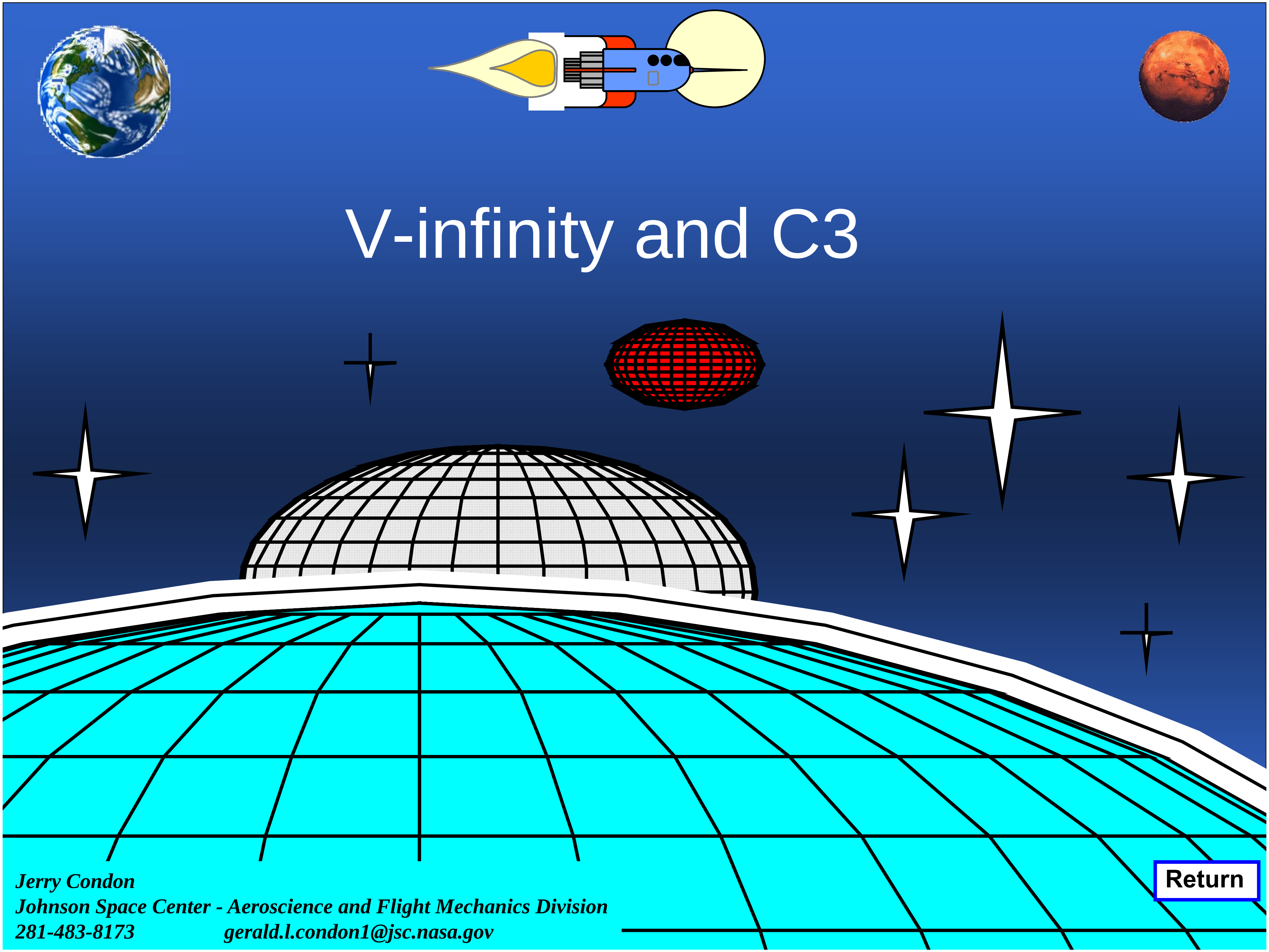
Data based on lunar orbit lifetime propagation time of 365 days for a matrix of inclination and right ascension of the ascending nodes (RAAN) for selected circular parking orbit altitudes.

Inclination Range (deg) = 0, 30, 60, 90

RAAN Range (deg) =
0, 30, 60, 90, 120, 150, 180, 210, 240, 270, 300, 330

Return

V-infinity and C3



Jerry Condon
Johnson Space Center - Aeroscience and Flight Mechanics Division
281-483-8173 gerald.l.condon1@jsc.nasa.gov

Return

Vis-Viva Equation

Vis-Viva “Life Force” Equation is a statement of conservation of energy

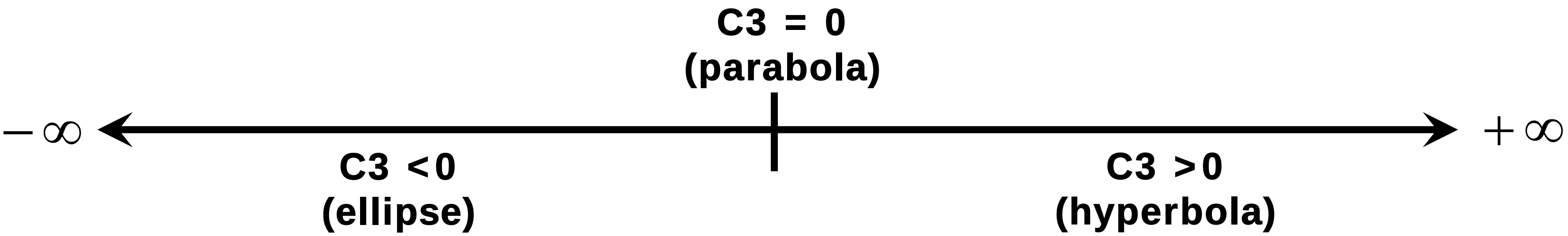
$$V^2 - \frac{2\mu}{r} = const. = C3$$

$$\frac{1}{2} m V^2 \qquad\qquad -\frac{\mu m}{r}$$

Kinetic Energy

Specific Mechanical Energy (total energy per 2 units of mass

C3 determines the type of conic section describing the orbit.



C3 = 0
(parabola)

$-\infty$

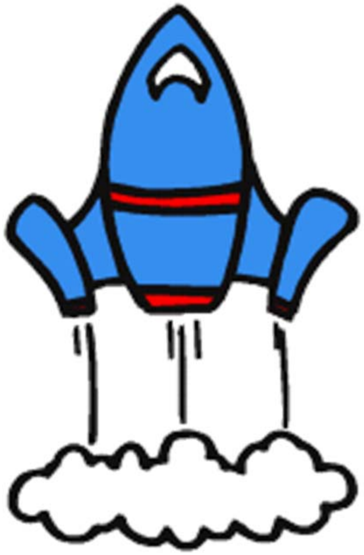
$+\infty$

C3 <0
(ellipse)

C3 >0
(hyperbola)

Typical Form of Vis-Viva Equation

From the general equation for a conic:



$$V^2 = \frac{2\mu}{r} - \frac{\mu}{a} \quad \Longrightarrow \quad \frac{C3}{\mu} = -\frac{1}{a}$$

dynamic quantity

geometric quantity

This is valid for all equations

Typical form of Vis-Viva equation used by flight mechanics:

$$V^2 = \frac{2\mu}{r} - \frac{\mu}{a} \qquad (1)$$

- or -

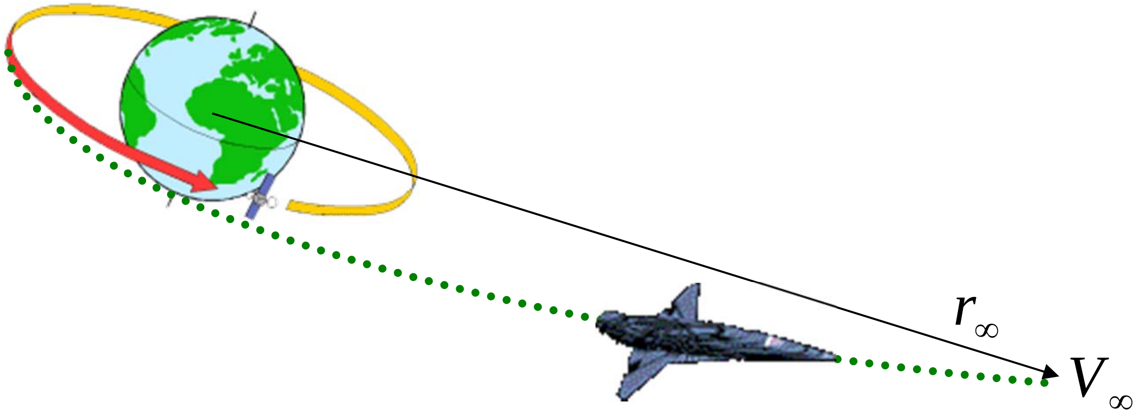
$$V^2 = \frac{2\mu}{r} + C3 \qquad (2)$$

Vis-Viva Equation and Hyperbolic Excess Speed

Let’s look at (1) $V^2 = \frac{2\mu}{r} - \frac{\mu}{a}$

as r → ∞ $V_\infty{}^2 = \overset{0}{\cancel{\frac{2\mu}{r_\infty}}} - \frac{\mu}{a} \Rightarrow V_\infty{}^2 = -\frac{\mu}{a}$

(1) becomes $V^2 = \underbrace{\frac{2\mu}{r}}_{\text{escape speed}} + \underbrace{V_\infty^2}_{\text{hyperbolic excess speed}} \Rightarrow V_\infty^2 = C3$ **(in units of km²/s²)**



Circular & Parabolic Orbit Case

Circular orbit case

Using (1) for a circular orbit, a = r

$$V^2 = \frac{2\mu}{r} - \frac{\mu}{r} = \frac{\mu}{r} \Longrightarrow \boxed{V_{circ} = \sqrt{\frac{\mu}{r}}}$$

(Circular orbit speed)

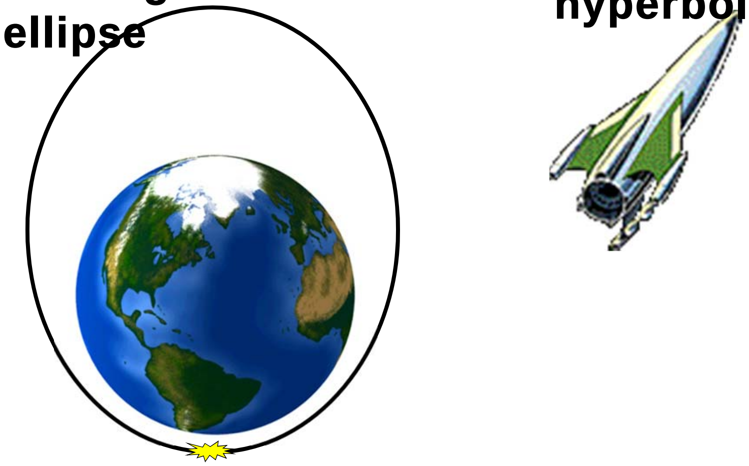
Parabolic orbit case

Using (1) for a parabolic orbit, $C3 = -\frac{\mu}{a} = 0$

$$V^2_{parabola} = \frac{2\mu}{r} \Longrightarrow \boxed{V_{parabola} = \sqrt{\frac{2\mu}{r}}}$$

(Escape speed)

Using Vis-Viva Equation to determine ΔV requirements



Parking orbit ellipse

Departure hyperbola

$$\Delta V_{departure} = V_{required} - V_{current}$$

$$V_{required} = \sqrt{\frac{2\mu}{r_{perigee}} + C3}$$

$$V_{current} = \sqrt{\frac{2\mu}{r_{perigee}} - \frac{\mu}{a}}$$

$$\boxed{\Delta V_{departure} = \sqrt{\frac{2\mu}{r_{perigee}} + C3} - \sqrt{\frac{2\mu}{r_{perigee}} - \frac{\mu}{a}}}$$

For an Earth departure (robotic mission)

μ = 398602.3 km^3/s^2

r_{earth} = 6378 km

$h_{perigee}$ = 407 km

h_{apogee} = 1000 km

C3 (2) = 10 km^2/s^2

$r_p = r_{perigee}$ = $r_{earth} + h_{perigee}$ = 6785 km

$r_a = r_{apogee}$ = $r_{earth} + h_{apogee}$ = 7378 km

$a = (r_a + r_p)/2$ = 7081.5 km

$\Delta V_{departure}$ = 11.2205 – 7.7211 km/s

$\boxed{\Delta V_{departure} = 3.4994 \text{ km/s}}$

Note: C3 target is independent of initial parking orbit

Typical C3 Values

Robotic Mars Missions

Mars Sample Return

2011	C3 (km^2/s^2)	Type	Arrival Entry Speed (km/s)
	9.8	II	5.6
	12.5	I	6.2
	17.7	IV	6.4
2013			
	10.2	II	5.9
	13.1	I	6.7
	14.7	IV	5.9

Human Mars Missions

Mars Combo Lander

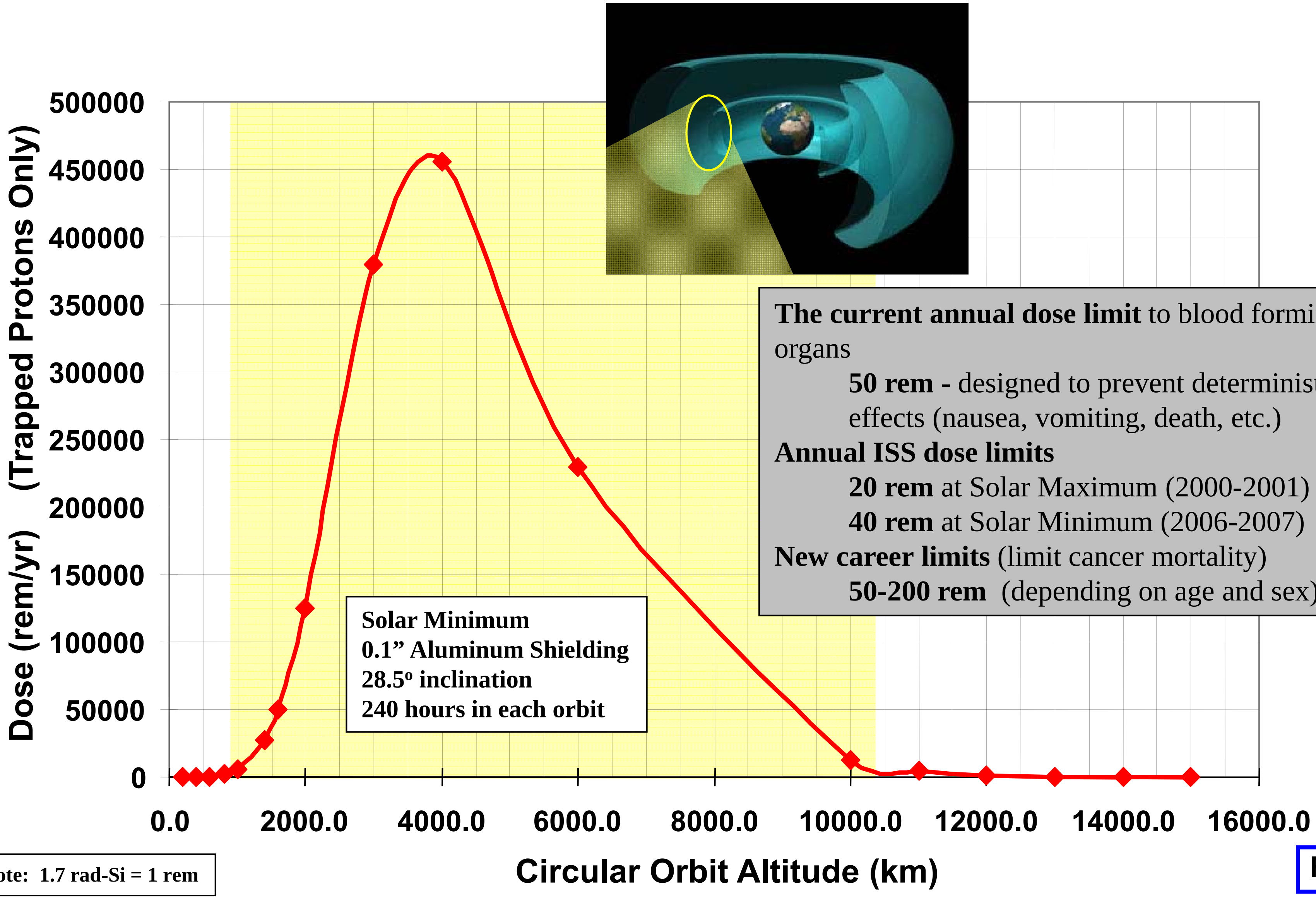
2014			
	$\leq$18.8	I	$\leq$7.36

Lunar Missions

0.9	na	na

Return

Van Allen Radiation Belts - Trapped Proton Belt Dose Rate for Circular Orbits



Solar Minimum
0.1” Aluminum Shielding
28.5° inclination
240 hours in each orbit

The current annual dose limit to blood forming organs

- **50 rem -** designed to prevent deterministic effects (nausea, vomiting, death, etc.)

Annual ISS dose limits

- **20 rem** at Solar Maximum (2000-2001)
- **40 rem** at Solar Minimum (2006-2007)

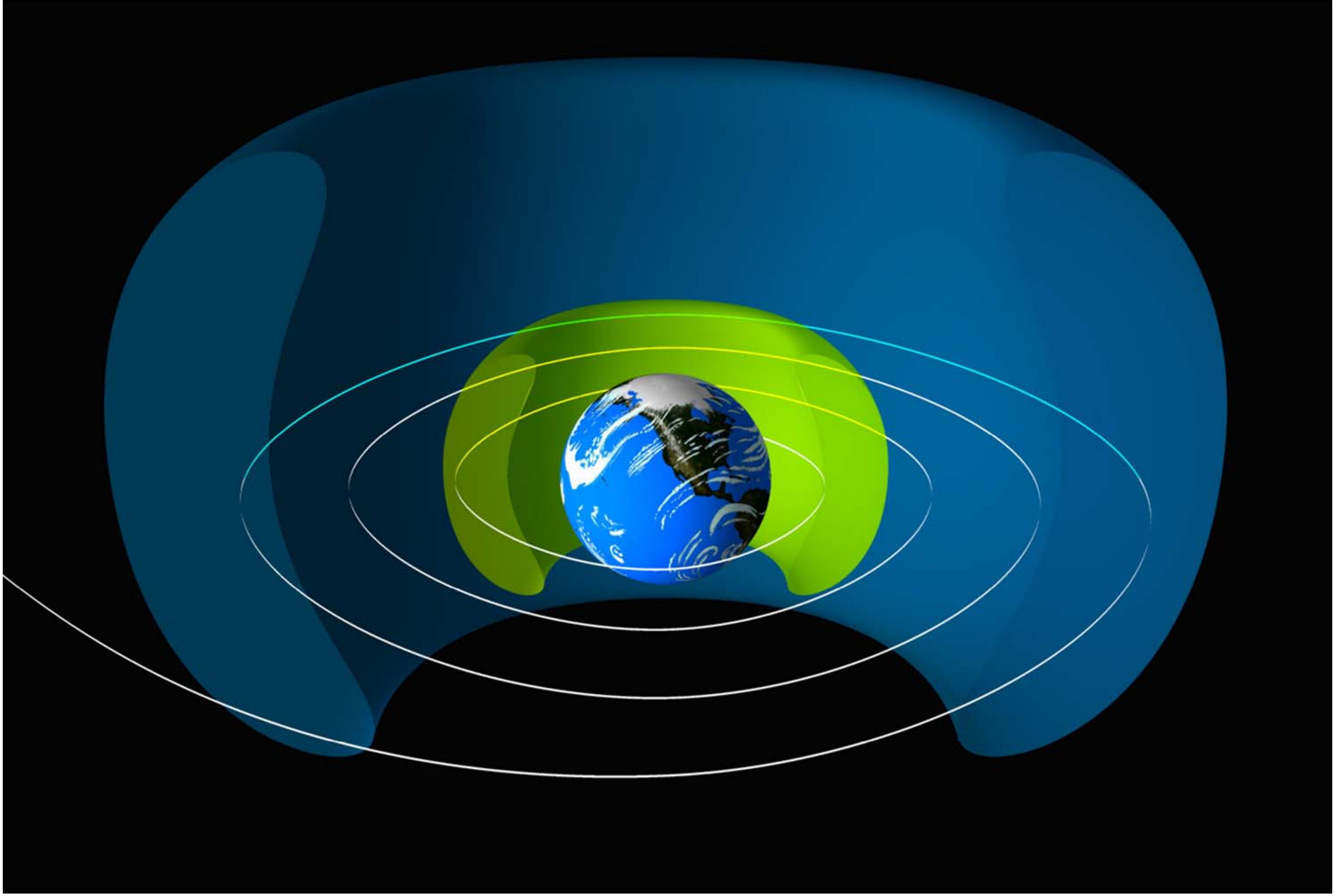
New career limits (limit cancer mortality)

- **50-200 rem** (depending on age and sex)

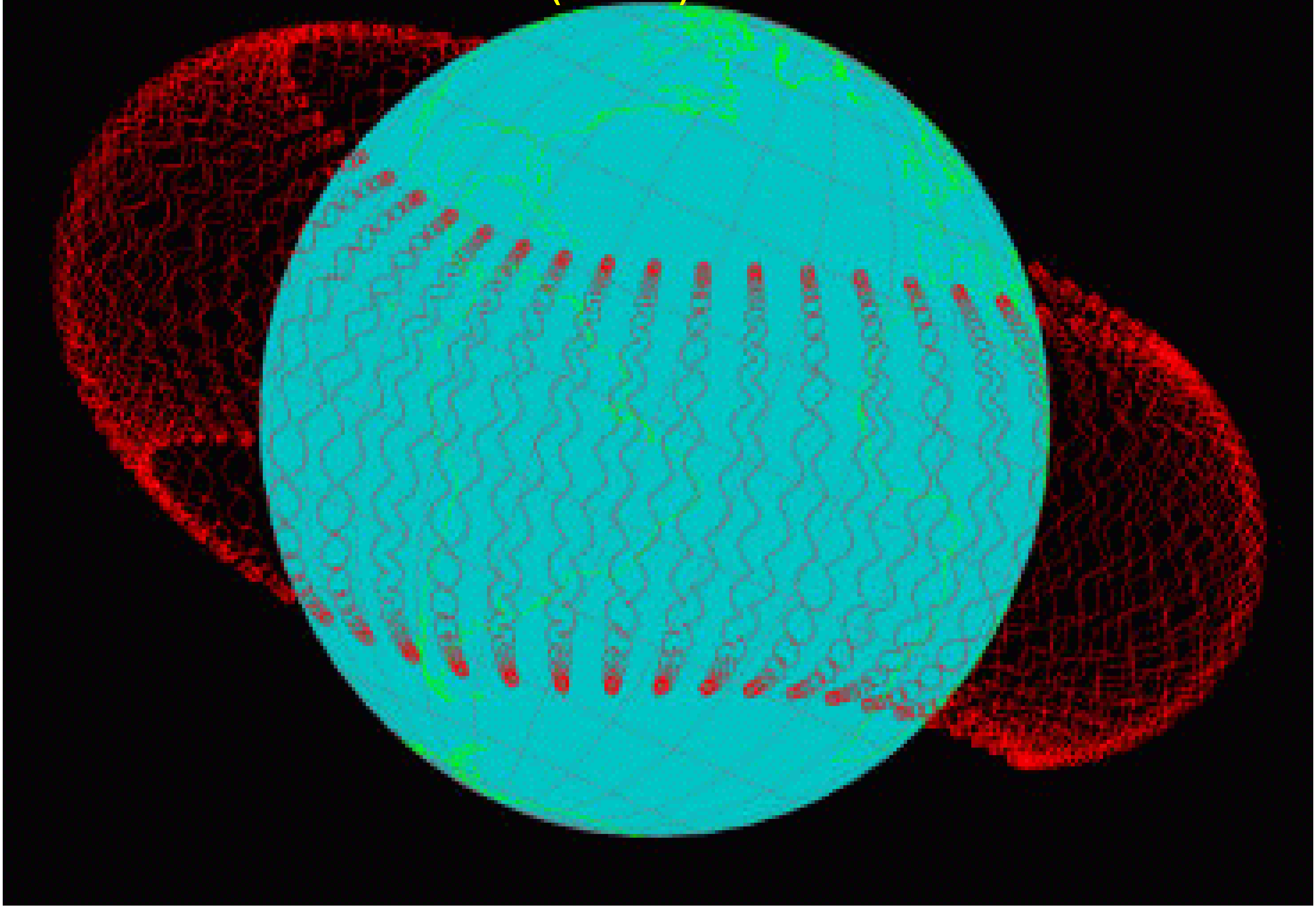
Note: 1.7 rad-Si = 1 rem

Return

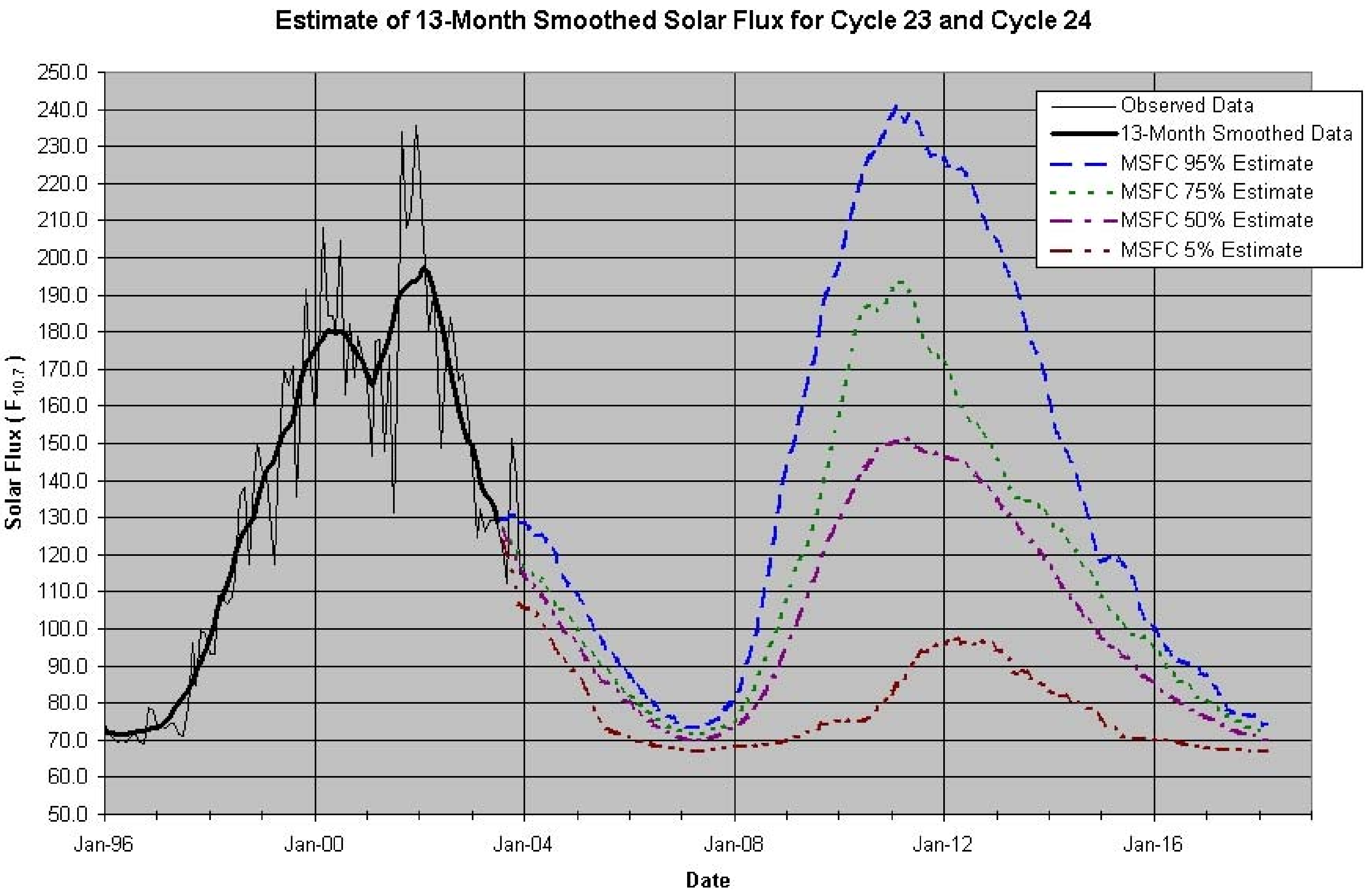
Van Allen Radiation Belts



Trapped Particle Trails
Van Allen (Proton) Radiation Belt



Solar Flux



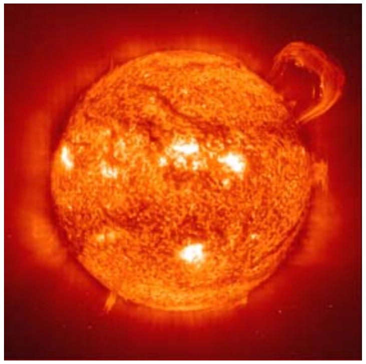
April

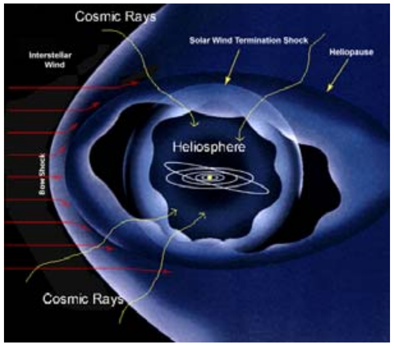
Source Data: "Future Solar Activity Estimates for Use in Prediction of Space Environmental Effects On Spacecraft" by H. Euler and S. Smith

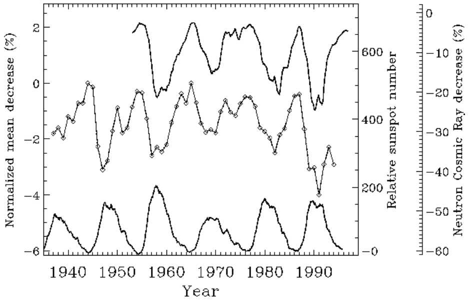
February 3, 2004

Solar and GCR

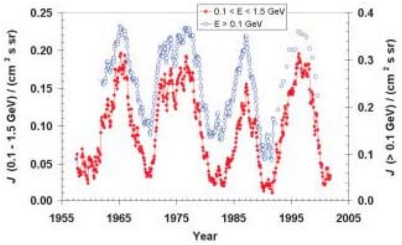
- ❑ Solar Flares
  - Can cause ionization damage and single-event effects in sensitive devices
  - Energetic protons reach Earth within 30 minutes
  - Other solar materials and magnetic fields reach Earth in 1 to 4 days
- ❑ Solar Flux
  - 9 to 13 year cycle
  - Proton energy range from 10 MeV to 1 GeV
- ❑ Galactic Cosmic Ray (GCR) Flux
  - Causes single-event effects in sensitive devices
  - Peaks around solar minimum
  - Particle energy range from 0 to over 10 GeV







H.S. Ahluwalia, 1997, *J. Geophy. Res.* **102**



Yu.I. Stozhkov, et al., *28th Int'l Cosmic Ray Conf.*

End of Part I

Lunar Orbit Mechanics