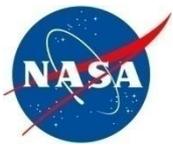


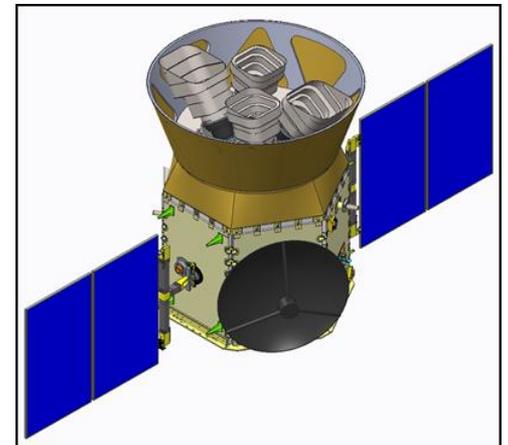
Trajectory Design to Mitigate Risk on the Transiting Exoplanet Survey Satellite (TESS) Mission

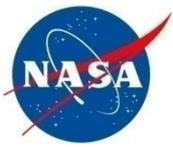
Donald Dichmann
Astrodynamics Specialist Conference
16 Sep 2016



Overview

- The TESS mission utilizes a 2:1 Lunar Resonant orbit similar to IBEX and phasing loops similar to LADEE
- To design the TESS trajectory we draw on understanding of 3- and 4-body dynamics gained from both missions and from some smart people
- For TESS a major trajectory design challenge is to mitigate risk so that statistical DV is not too large, and so that a missed maneuver in the phasing loops would not cause us to lose the mission
- There are also Sun angle constraints on maneuvers, perigee altitude constraints and eclipse constraints that complicate the design
- The team of GSFC and ADS Flight Dynamics engineers has developed GMAT scripts with distributed processing in a generic software architecture to efficiently design trajectories and to run Monte Carlo simulations, with limited user intervention required
- With this architecture we ensure that mission constraints are met and we can explore the mission design trade space





TESS Flight Dynamics Analysts

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Greg Henning

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Applied Defense Solutions (ADS)

Ryan Lebois*

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Stephen Lutz*

John Ferreira

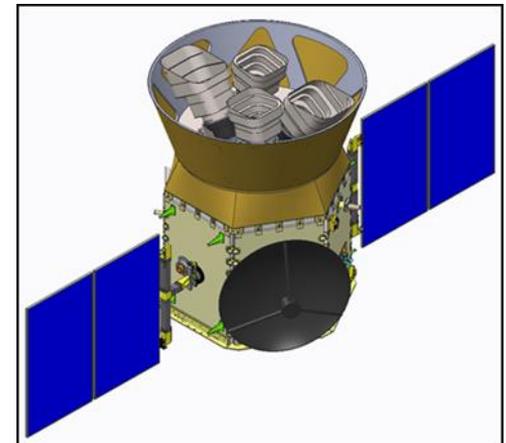
Lisa Policastri (now UCB SSL)

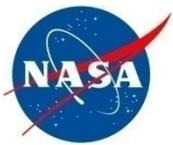
Purdue University

Amanda Haapala Chalk (now JHU APL)

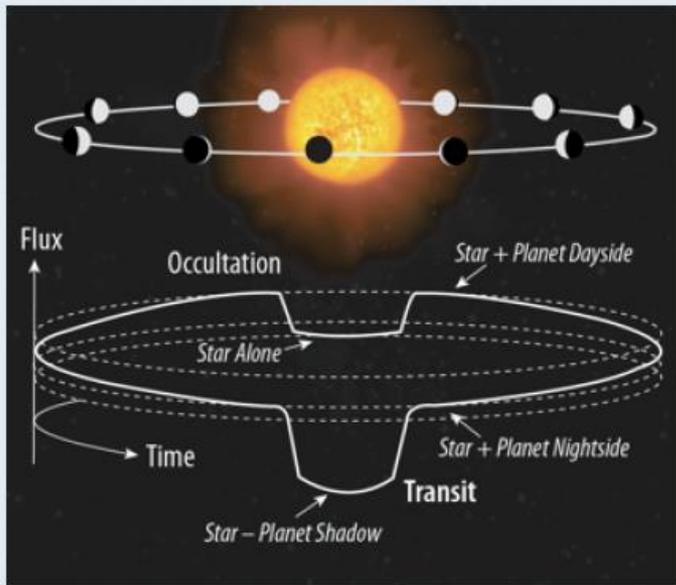
Cody Short (now AGI)

*Authors





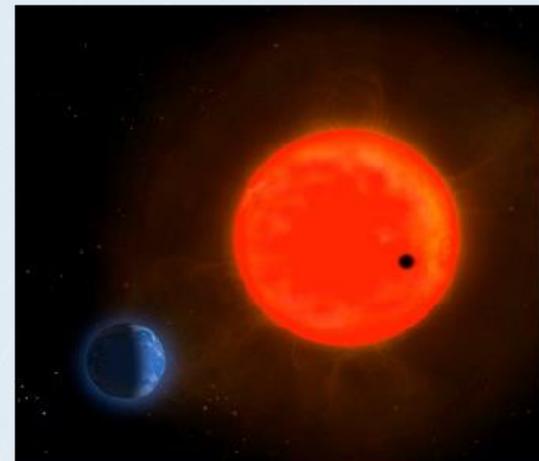
TESS Science Goals and Drivers

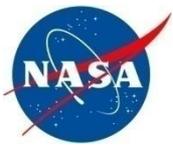


- ◆ **Primary Goal:** Discover Transiting Earths and Super-Earths Orbiting Bright, Nearby Stars
 - *Rocky Planets & Water Worlds*
 - *Habitable Planets*
- ◆ Discover the “Best” ~1000 **Small** Exoplanets
 - “Best” Means “Readily Characterizable”
 - *Bright Host Stars*
 - *Measurable Mass & Atmospheric Properties*
 - Present: Only 3 small transiting exoplanets orbiting bright hosts are known

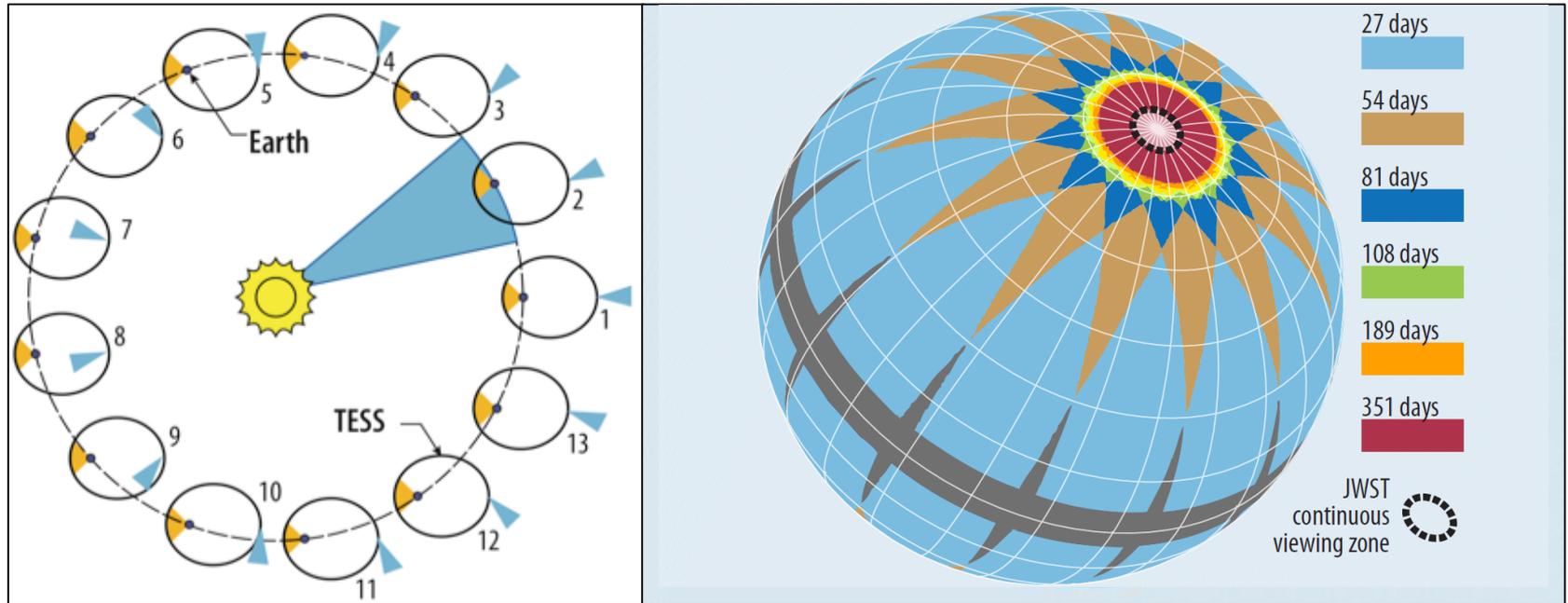
◆ Large Area Survey of Bright Stars

- *F, G, K dwarfs: +4 to +12 magnitude*
- *M dwarfs known within ~60 parsecs*
- “All sky” observations in 2 years:
 - *> 200,000 target stars at <2 min cadence*
 - *> 20,000,000 stars in full frames at 30 min cadence*





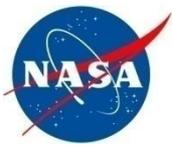
TESS 2-Year Sky Coverage Map



Anti-Solar segments
drive +/- 15 deg

Coverage of ecliptic poles drives
Pitch angle (nominally 54 deg)

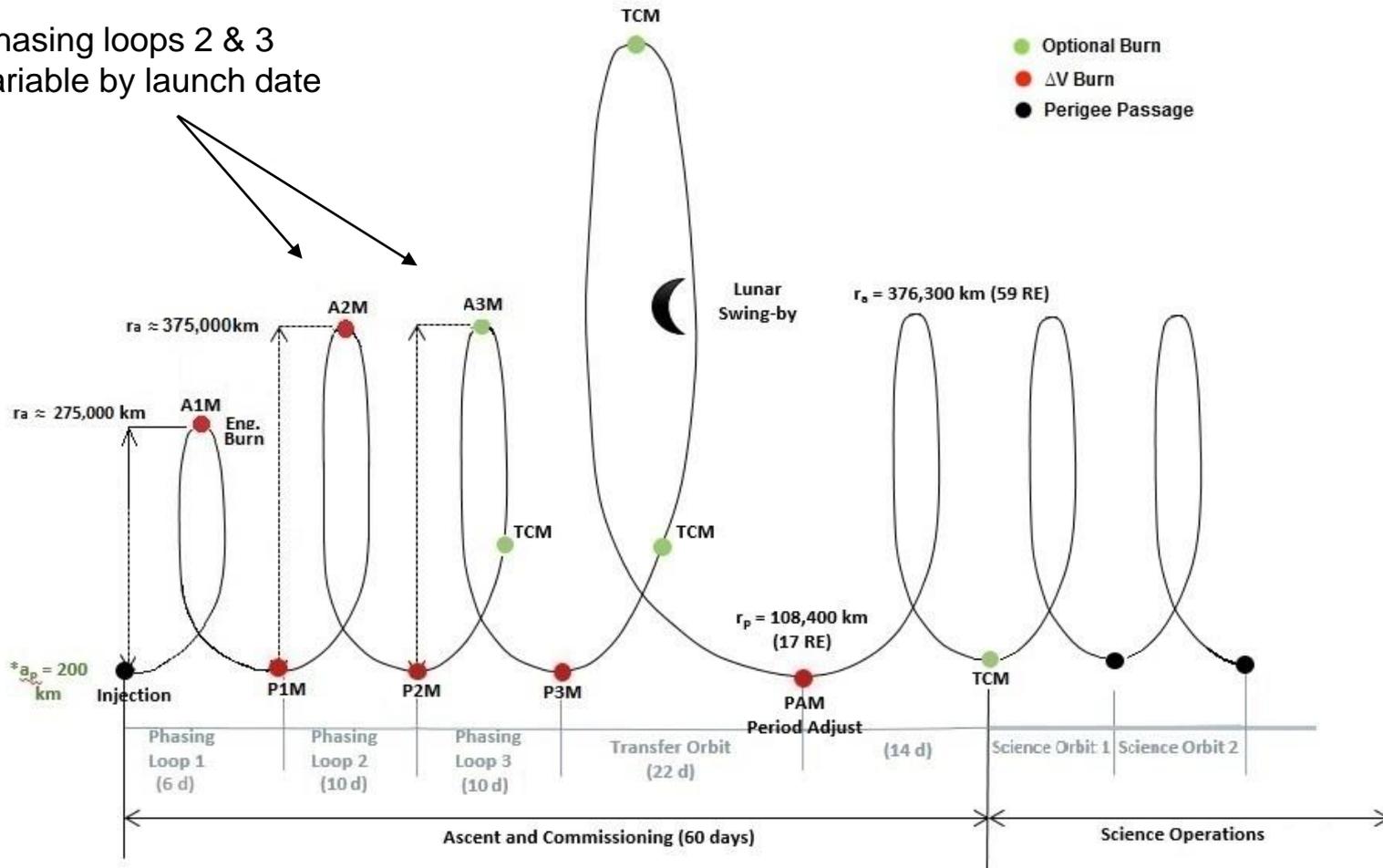
- Concentration of coverage at the ecliptic poles for JWST.
- Sacrifice of coverage in the ecliptic because Kepler-2 is already mapping that region.



Launch to Science Orbit Timeline

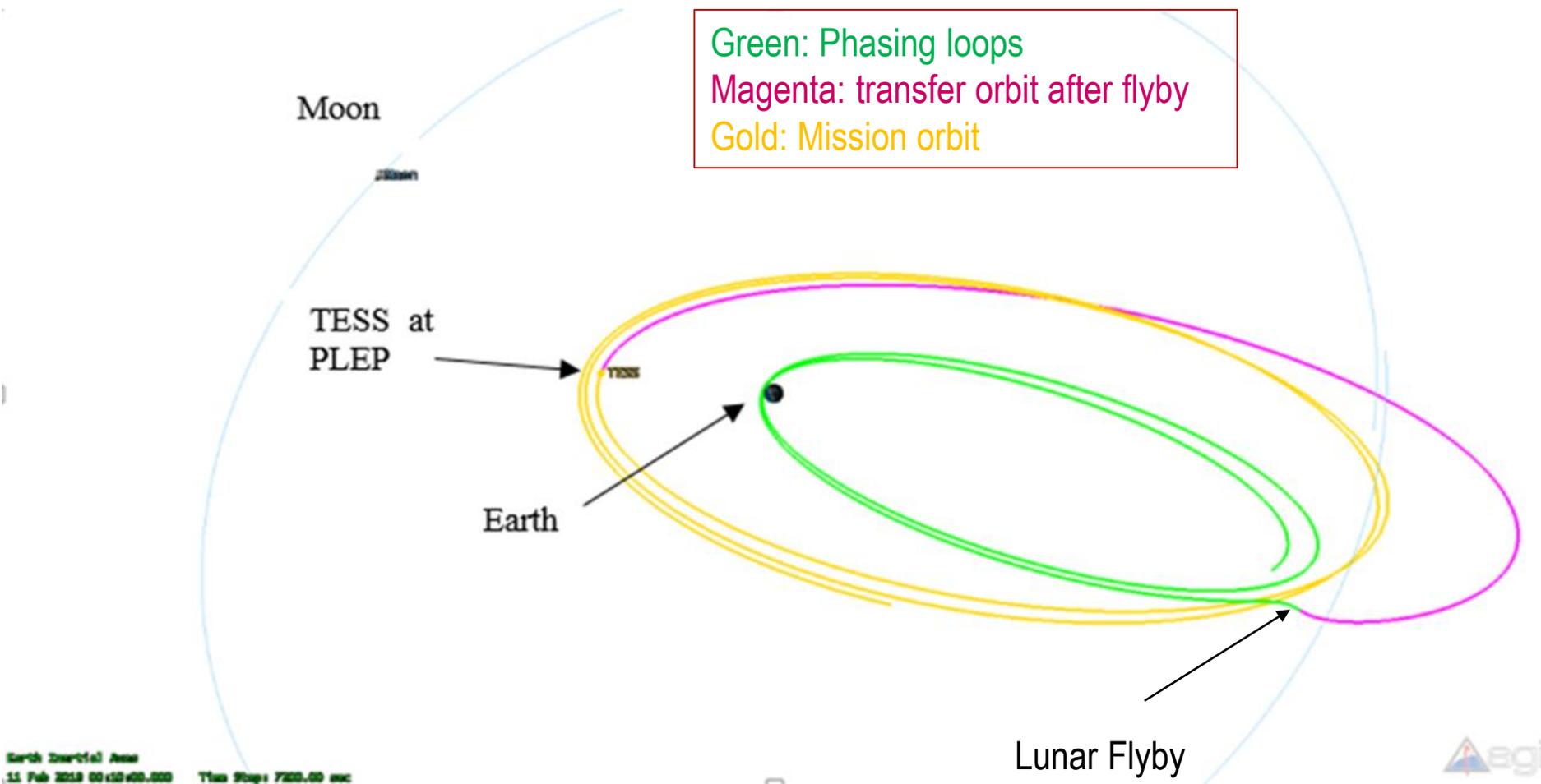
Launch on SpaceX Falcon 9
Launch in Dec 2017

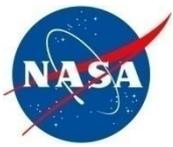
Phasing loops 2 & 3
variable by launch date



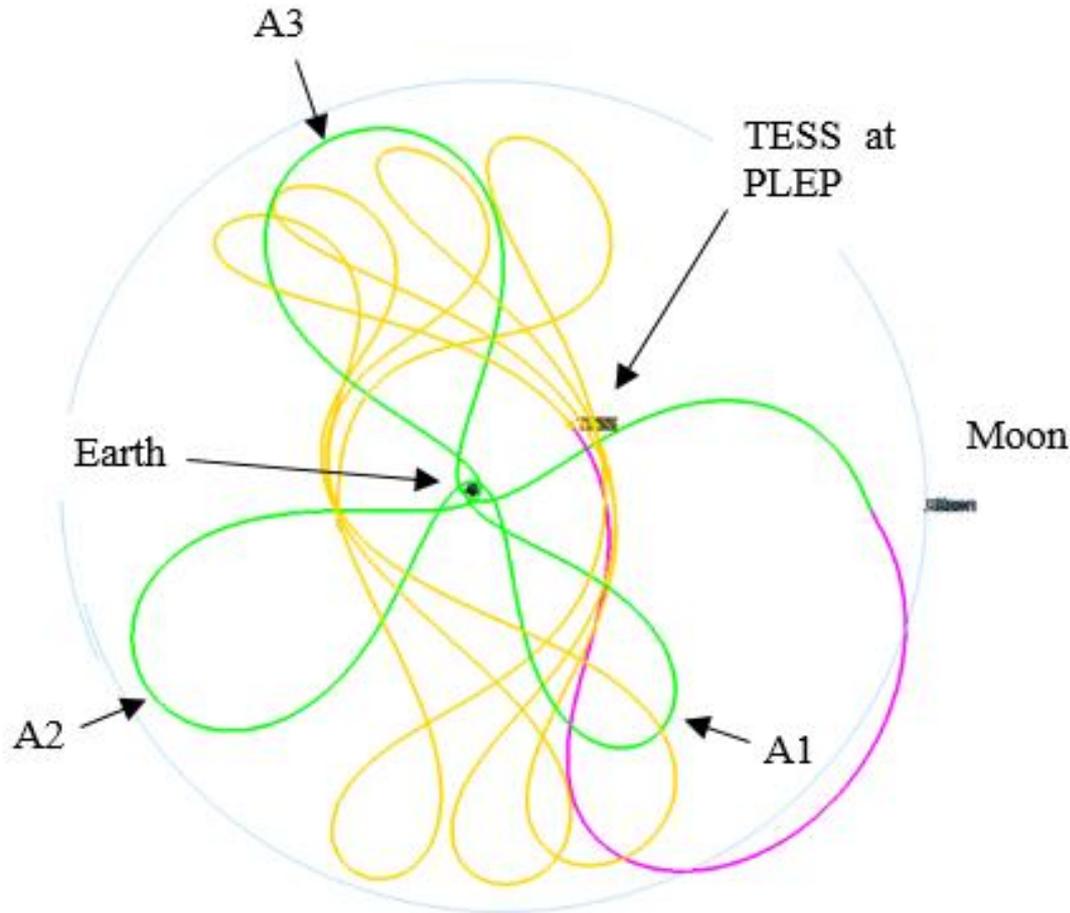


2 Jan 2018 solution: Inertial frame



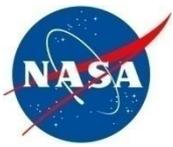


2 Jan 2018 solution: Rotating frame

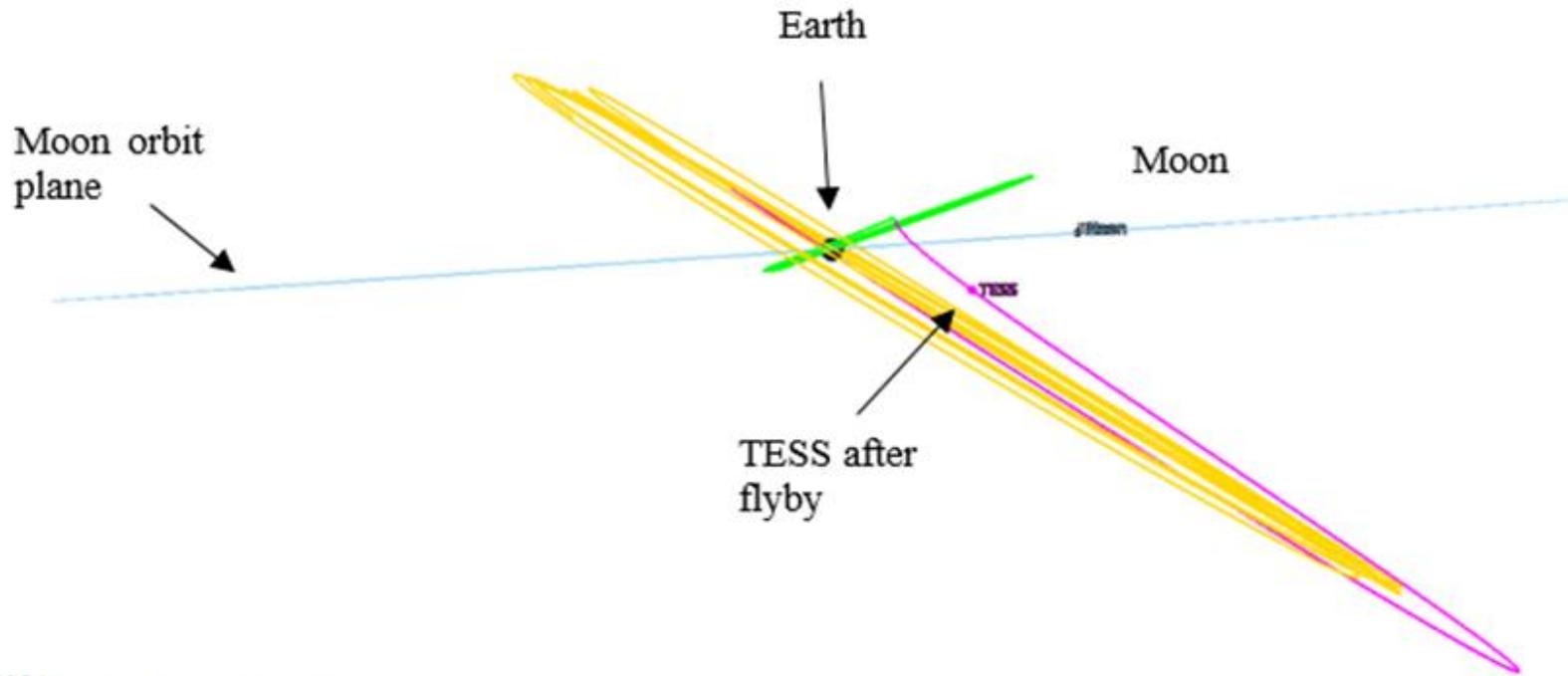


Lunar Resonant Phasing (LRP) condition: At apogee the Moon-Earth-spacecraft angle is large. This removes the need for orbit maintenance maneuvers and makes the orbit operationally stable

The phasing loop duration is chosen so at A1, the Moon is ahead of the spacecraft. This raises perigee naturally, so A1M is not critical for the mission



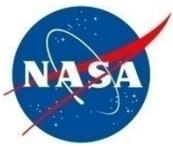
Plane Change at Flyby



Earth Orbital Area
20 Jan 2018 18:10:00.000 Time Step: 7200.00 sec

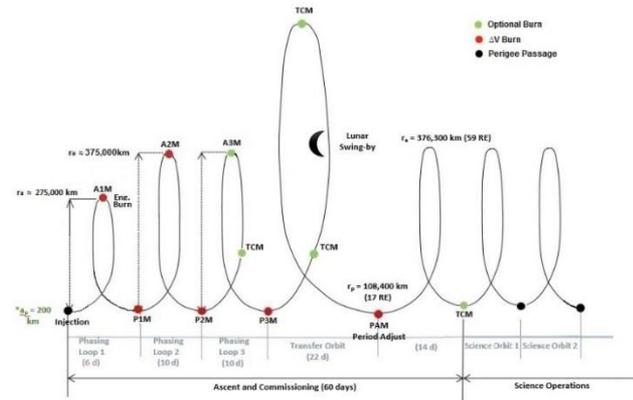
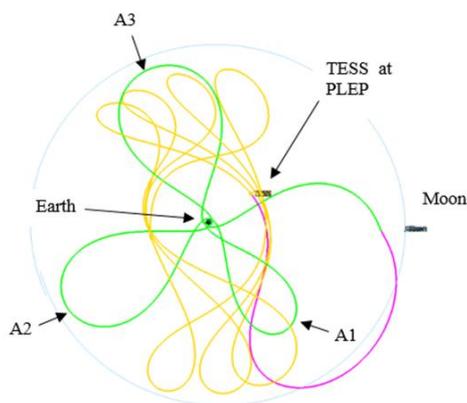


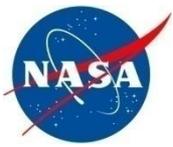
Lunar flyby changes inclination by about 47 deg, so apogee is far outside the ecliptic plane



Mission constraints

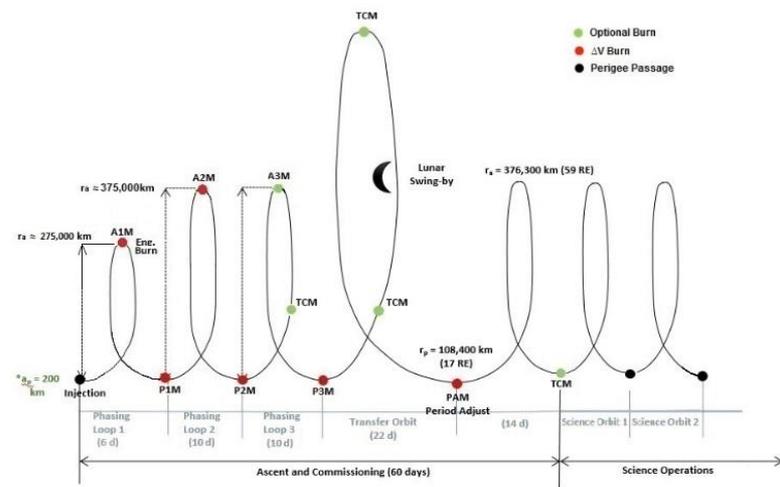
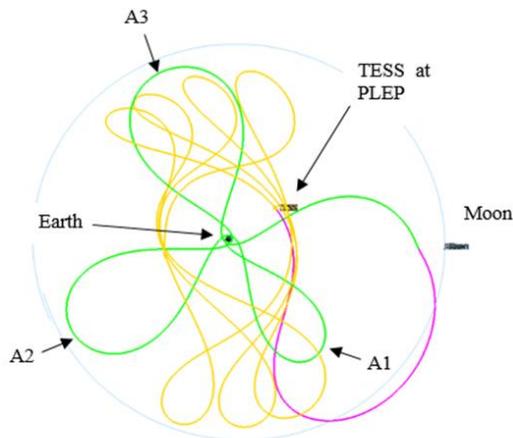
- Mission orbit period 2:1 resonant with Moon = 13.65 days
 - Semimajor axis 38 Re
- At least five launch opportunities in a lunar cycle
- Achieve mission orbit within 2 months after launch
- Total DV \leq 215 m/s
 - \sim 150 m/s deterministic, \sim 25 m/s for injection error, \sim 20 m/s for statistical DV and \sim 20 m/s margin
- Initial mission orbit perigee = 17 Re
- Mission orbit perigee remains below 22 Re for communications
- Eclipses
 - At most 16
 - No more than two between 4 to 5 hours; the rest less than 4 hours
- Phasing loop perigee P1, P2, P3 above 600 km
- Sun angle at maneuvers: Sun-to-boresight angle can be less than 30 deg for no more than 15 minutes at a time
- Mission orbit perigee stays above GEO radius for 30 years

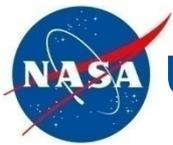




Mission goals

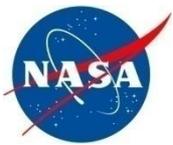
- **Operational Stable Mission Orbit**
 - No orbit maintenance maneuvers required
- **No critical maneuvers**
 - A maneuver is deemed critical if missing the maneuver may mean we cannot complete the mission
- **Initial ecliptic AOP ≤ 35 deg**
 - This is known to help us achieve eclipse constraint
- **Select injection AOP to achieve desired injection coverage**





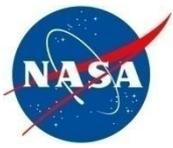
Using Astrodynamics to Meet Mission Constraints & Goals

- Eclipse constraint
 - **Kozai mechanism**
 - Avoid flyby near full Moon
- Mission orbit perigee constraint
 - **Kozai mechanism**
- Operational orbit stability
 - **Lunar Resonant Phasing (LRP) condition: lunar resonant orbit & proper phasing with the Moon**
- No critical maneuvers
 - Introduce phasing loops, to allow time to correct for maneuver errors and to calibrate engine
 - **Select phasing loop duration so A1M, P2M & P3M are not critical**
- GEO constraint
 - **Kozai mechanism (may include adjustment in initial mission orbit period)**
 - **LRP**
- Total DV \leq 215 m/s
 - Flyby to perform perigee raise, apogee raise and inclination change
 - **LRP to eliminate need for orbit maneuvers**
 - **Select phasing loop duration: raise perigee at A1, keep P2M & P3M small**
 - Optimization
 - Bound P2M & P3M to limit stat. DV
- Sun angle constraint: Shift maneuver away apsis

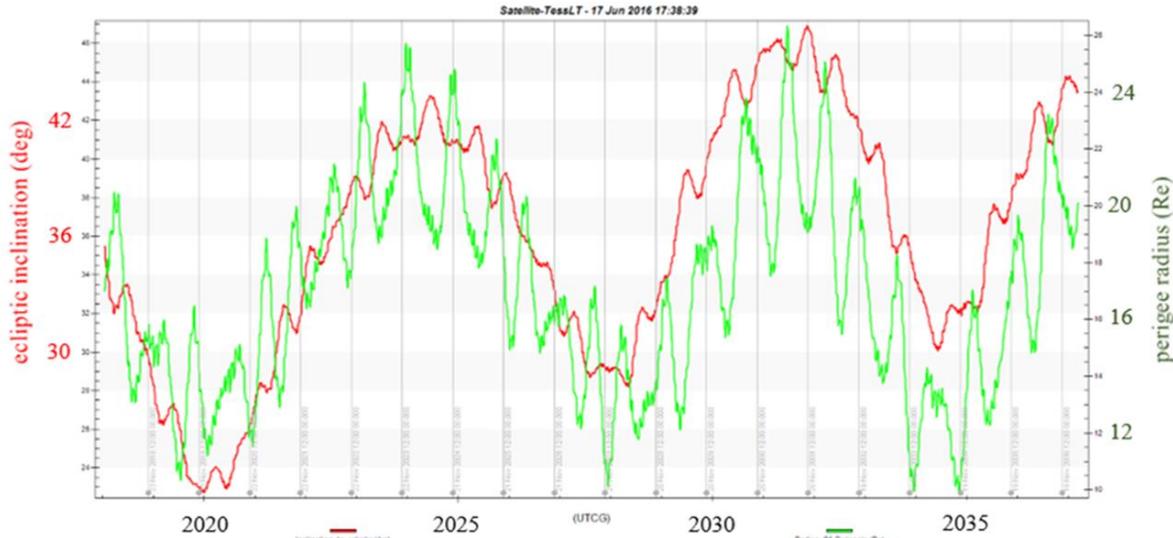


Kozai Mechanism

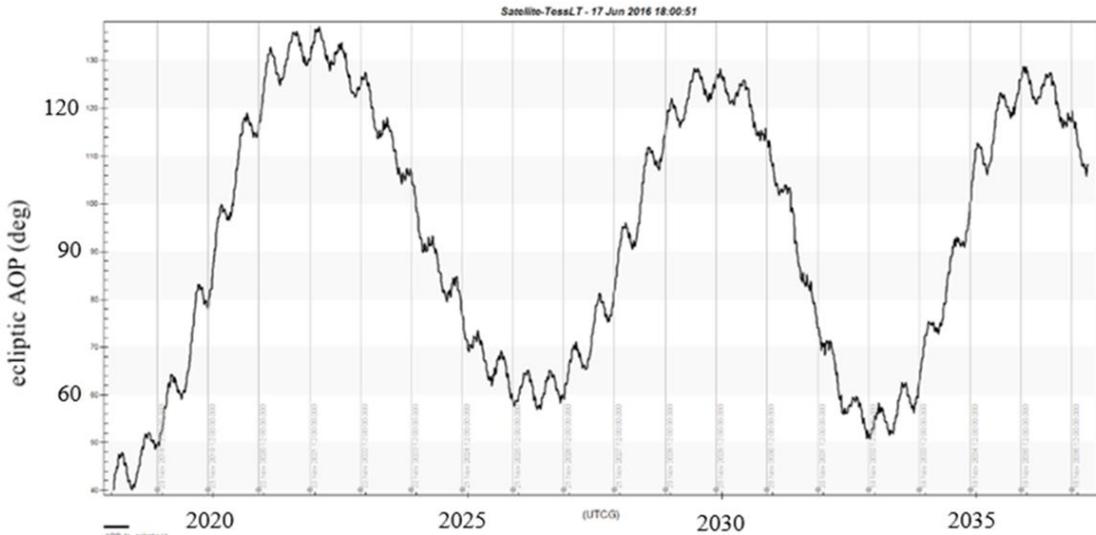
- The Kozai Mechanism describes the long-term evolution of a highly eccentric, highly inclined orbit due to a third body (for us, the Moon).
- The Kozai model implies that:
 - Orbit semimajor axis is conserved
 - Kozai parameter $K = \cos(i)\sqrt{1 - e^2}$ is constant, where e is eccentricity and i is inclination to the Moon orbit plane
 - Eccentricity and inclination oscillate in unison, with a period of about 10 years for a TESS-like orbit. (Therefore, perigee radius and inclination oscillate together.)
 - AOP relative to the Moon librates around 90 deg or 270 deg
- Kozai mechanism is relevant to TESS because
 - We want mission perigee radius to remain between 6.6 Re (GEO) and 22 Re
 - We want mission ecliptic AOP to remain near 90 deg or 270 deg, so line of apsides stays out of ecliptic plane, and so long eclipses cannot occur near apogee
- For TESS mission orbit, $e = 0.55$ so $K = 0.65$ implies $i = 39$ deg
- We exploit the fact that the lunar plane and ecliptic plane are near the same, only 5 deg apart.
- Perturbing forces (especially the Sun) imply that the Kozai mechanism does not work exactly in the full force model. Nevertheless, like CR3BP, the Kozai mechanism is a useful technique for orbit design



Kozai Mechanism (cont'd)

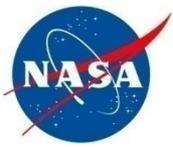


Evolution of perigee radius (green) and lunar inclination (red) over 20 years. The oscillation period is about 10 years

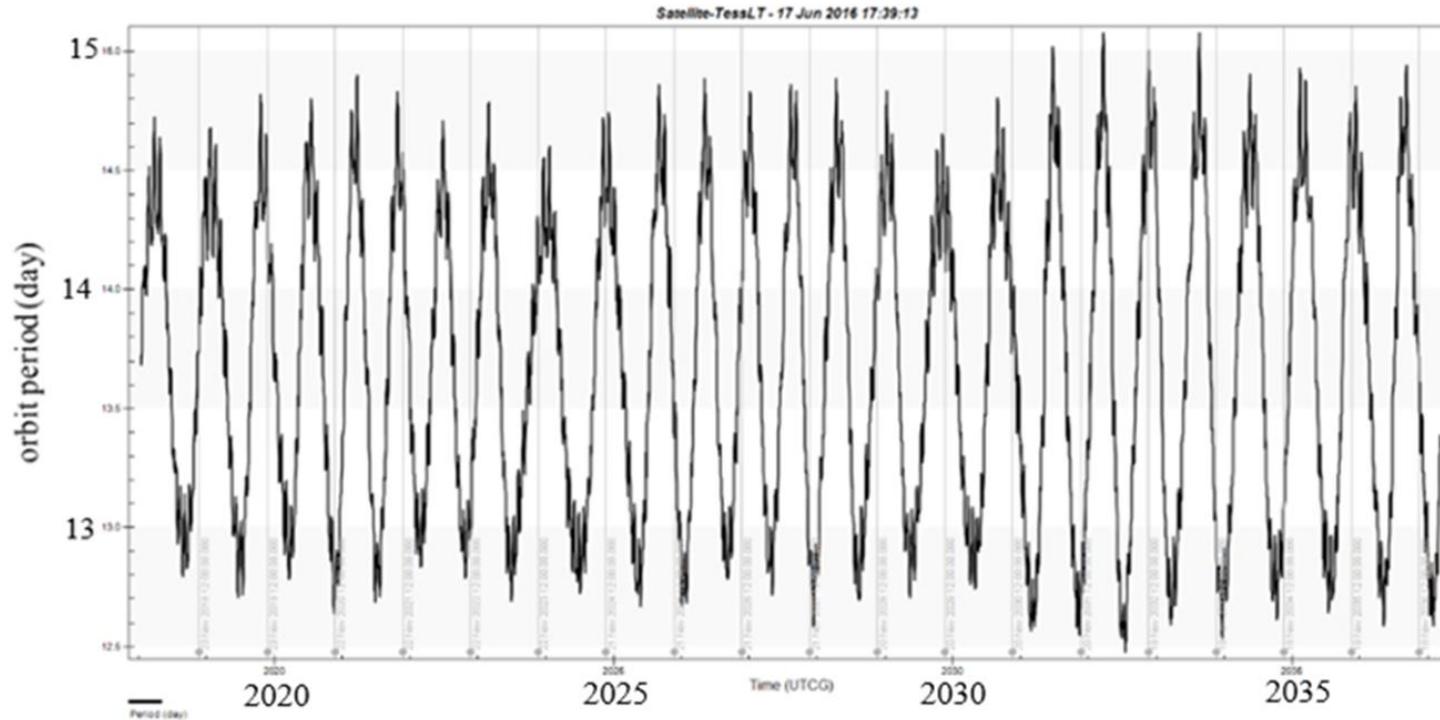


Evolution of ecliptic AOP over 20 years.

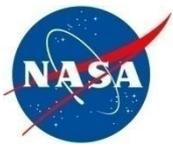
See also: Dichmann, Parker, Williams, Mendelsohn: Trajectory Design for the Transiting Exoplanet Survey Satellite. ISSFD 2014



Kozai Mechanism (cont'd)

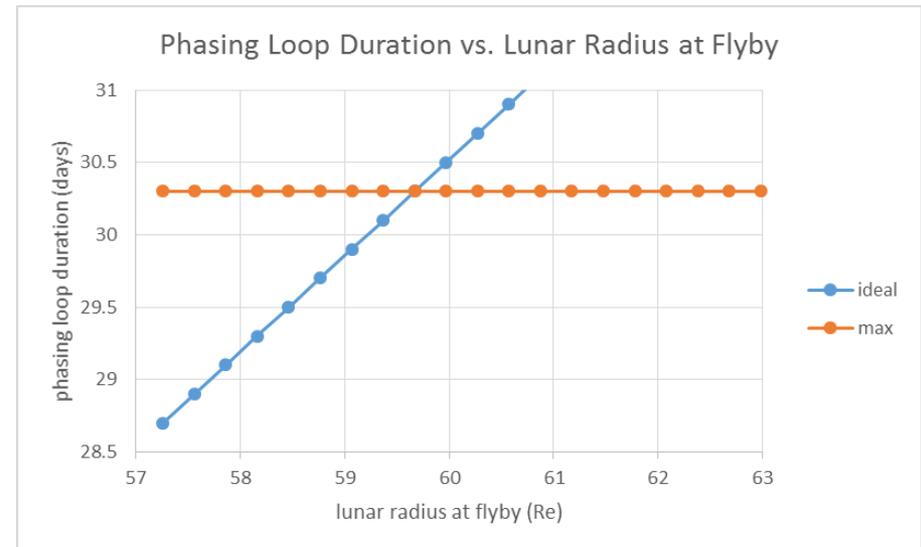
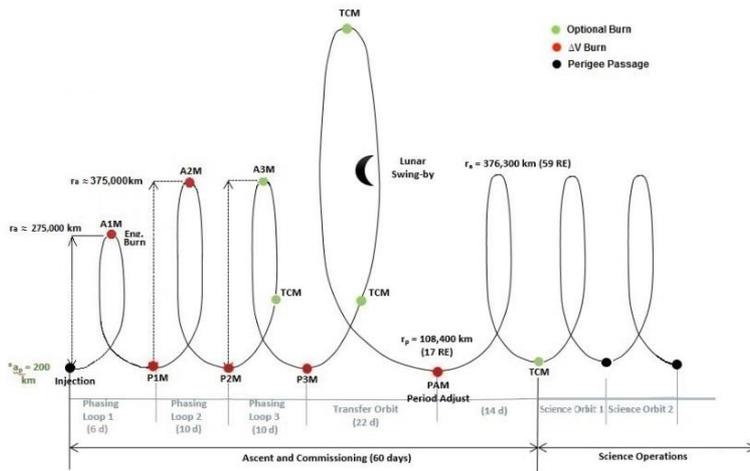
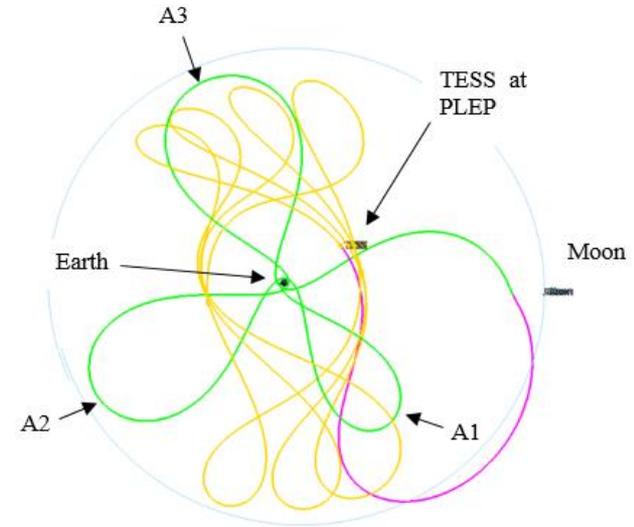


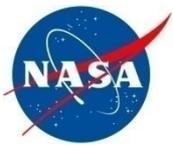
- Orbit period oscillates around ideal 13.65 days with amplitude near 1 day
- Thus we do not need to start at ideal resonance to achieve resonance on average
 - We change mission orbit period by changing the Period Adjust Maneuver (PAM) at mission orbit insertion
- IBEX extended mission design used this feature to select an initial orbit that achieved low eclipse durations for several years



Select Phasing Loop Duration to Avoid Critical Maneuvers

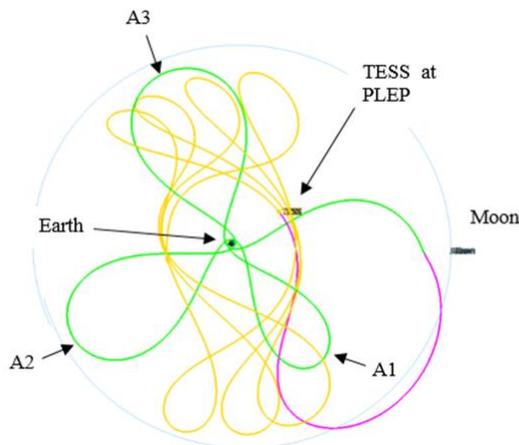
- **Ideal:** To reduce DV and to ensure P2 and P3 are not critical, it is desirable for loops 2 and 3 to have apogee equal to lunar radius at flyby. This would imply P2M and P3M near 0
- **Maximum:** To ensure that A1 is not critical, the interval from A1 to flyby can be no more than 27.3 days. The interval from injection to A1 is 3 days, so phasing loop duration should be no more than 30.3 days.
 - If ideal duration $>$ max duration, then A2 radius is below flyby radius, and P2M is nonzero

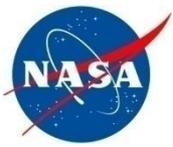




Operational Stability of Mission Orbit

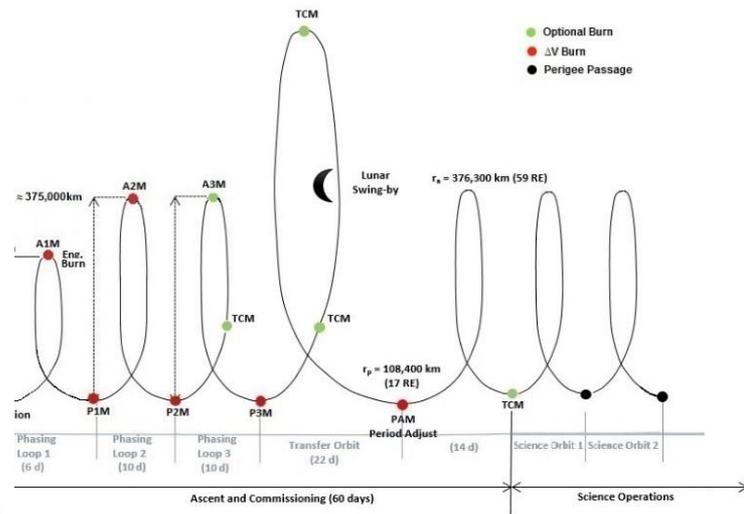
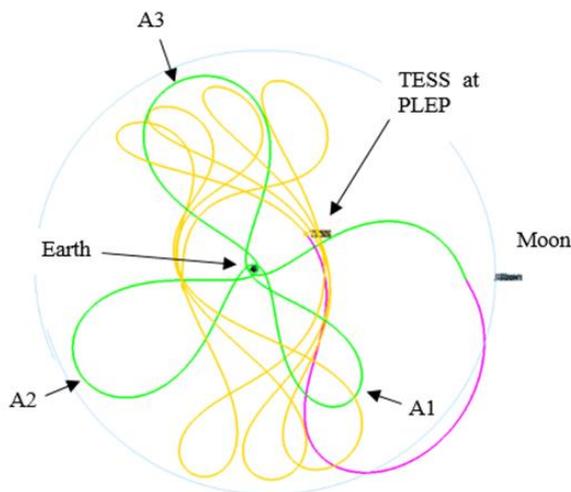
- “Operational stability” means no orbit maintenance maneuvers are required
- Operational stability is achieved by the LRP condition:
 - Orbit period oscillates around ideal resonance
 - Apogee is kept away from the Moon: Initially no more than 37 deg away from the ideal direction orthogonal to Moon.
 - Resonance implies that, if apogee-Earth-Moon angle is large at the start then it will remain large
- Dynamic stability analysis in Dichmann, Lebois, Carrico (2014) shows this type of orbit does exhibit operational stability for decades (but perhaps not forever)

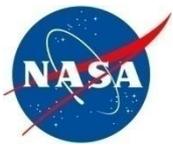




Total DV: Deterministic + Statistical ≤ 215 m/s

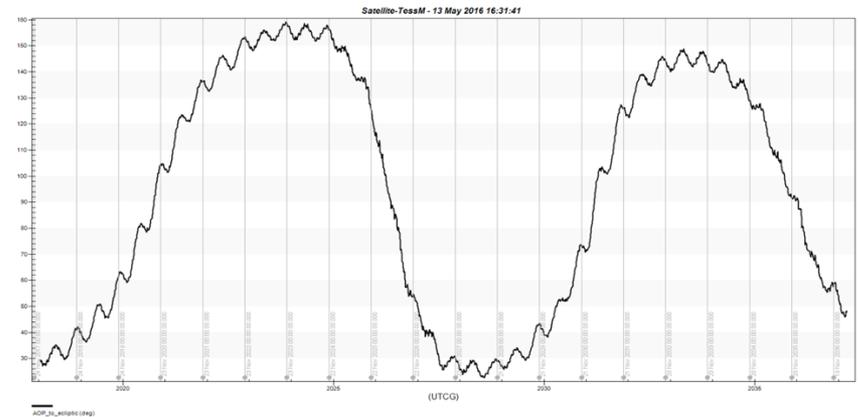
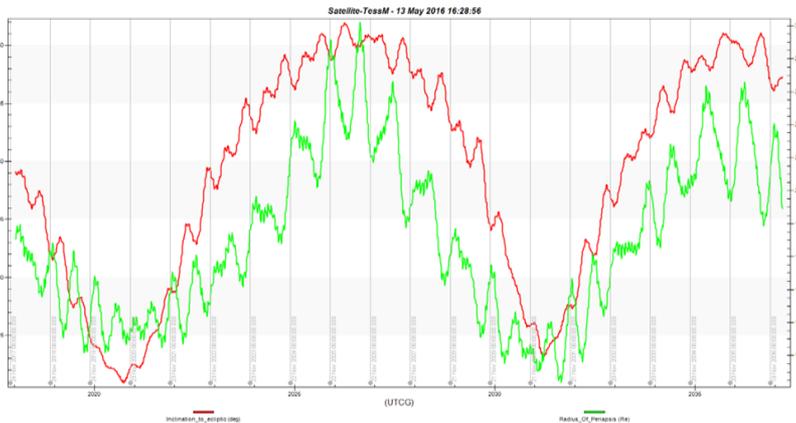
- Use lunar flyby to achieve transfer orbit plane change, apogee raise and perigee raise
- Use gravity assist from the Moon near A1 to raise P1 radius with no DV cost
- Whenever practical, use ideal phasing loop duration so P2M and P3M are near zero.
- Employ constrained optimization in GMAT to reduce DV
 - It is not practical to include statistical DV from Monte Carlo in trajectory optimization
 - Instead we introduce a DV proxy where we penalize P2M and P3M more, because they contribute most to the statistical DV.
- Explicitly bound P2M and P3M





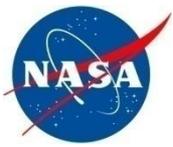
Eclipses: Kozai Mechanism to the Rescue

- Mission orbit: We avoid long eclipses by placing apogee well outside the ecliptic
 - Ecliptic inclination (left, red)
 - Lunar flyby achieves inclination ~ 40 deg
 - Kozai mechanism keeps inclination above 20 deg
 - Ecliptic AOP (right): Ideal value is 90 deg
 - Lunar flyby achieves initial AOP > 30 deg
 - Kozai mechanism causes AOP to oscillate about 90 deg with period ~ 8 years



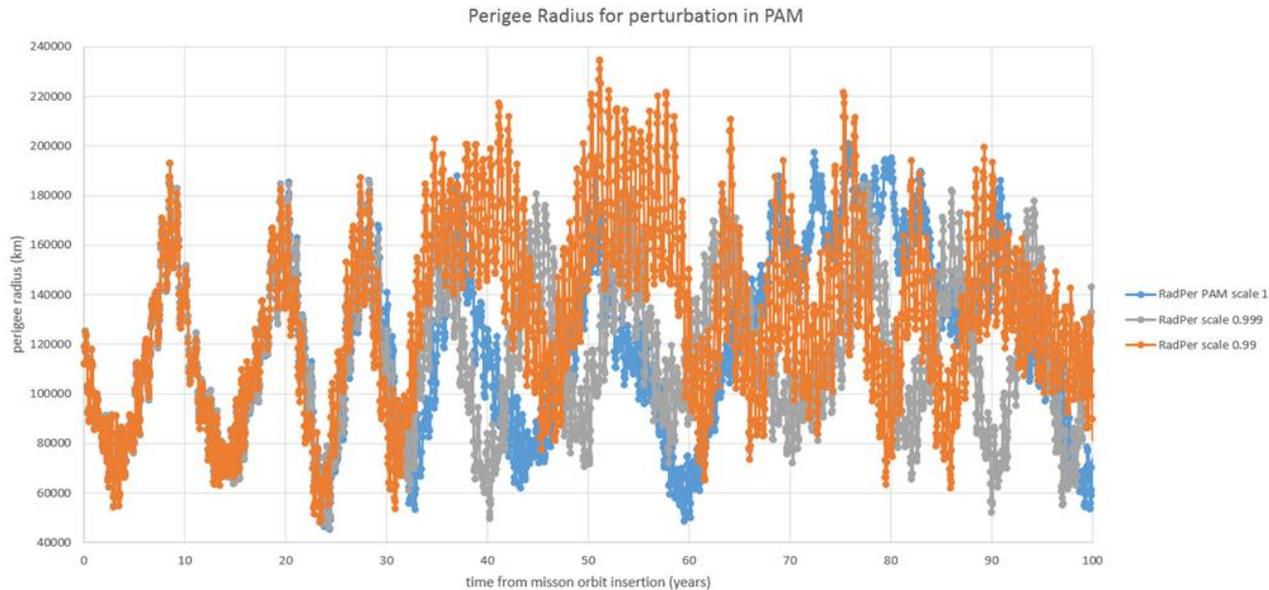
Phasing loops

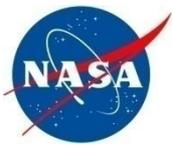
- *Avoid lunar flyby near full Moon, so TESS is not caught in shadow near apogee. In particular, avoid total lunar eclipse 31 Jan 2018 (so no launch for some dates near 4 Jan 2018)*
- *Likewise avoid any apogee near full Moon (so no launch for some dates near 23 Dec 2017)*
- *The project has expressed concern previously about eclipses inside the radiation belt near the Earth. To reduce such eclipses we have considered changing flyby epoch or switching from short coast to long coast.*



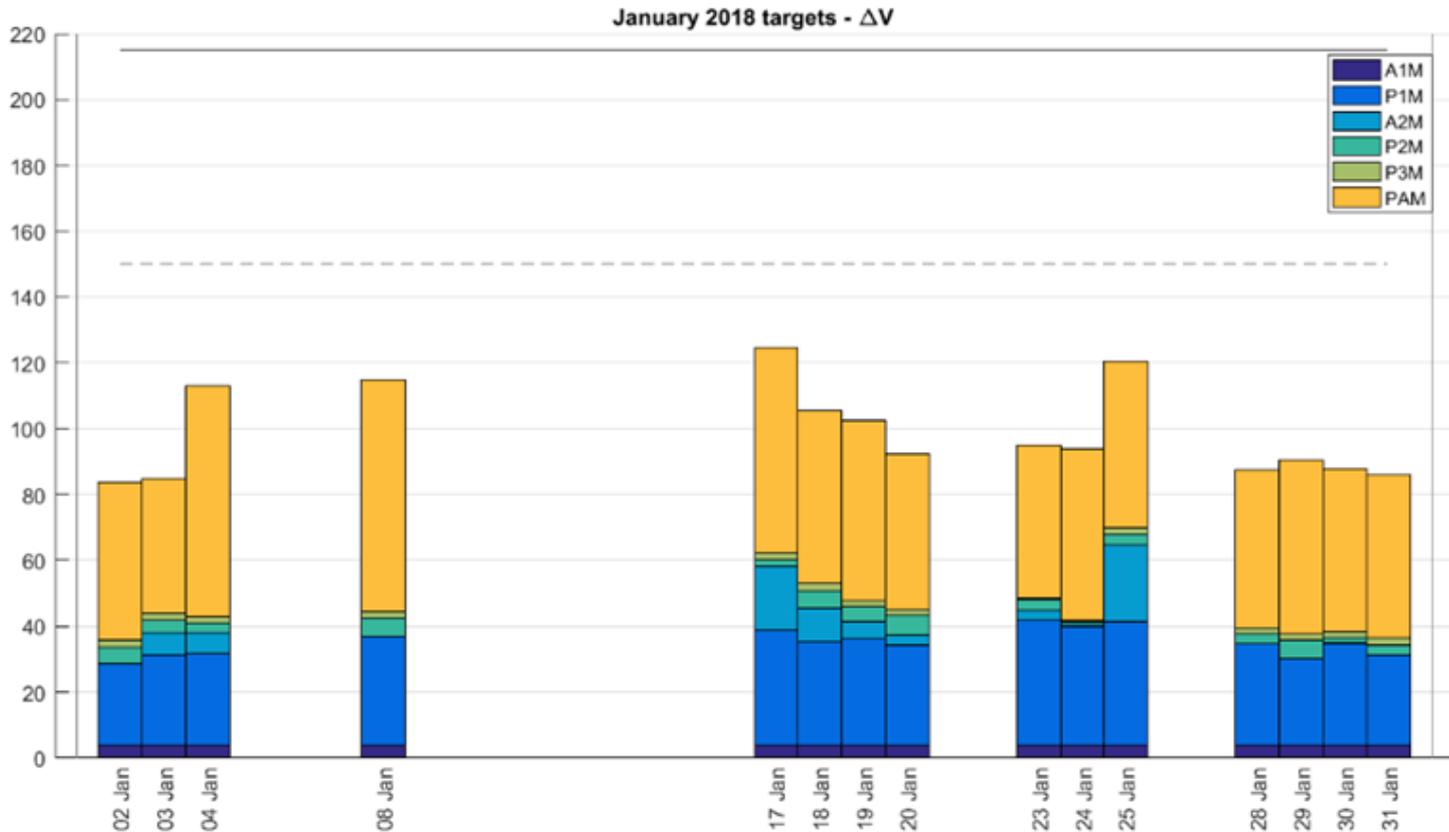
Long-Term Predictability & GEO constraint

- IBEX primary mission orbit became unpredictable after 2 years
- IBEX extended mission and TESS mission orbits were selected to achieve operational stability
- TESS mission is predictable for about 30 years
 - Maximum Floquet multiplier suggests that orbit becomes unpredictable after about 30 years
 - Simply changing the orbit propagator leads to notable change after 75 years
 - Error in PAM or introduction of Momentum Unloads leads to notable change in orbit after 30 years
- Based on this analysis, the TESS project is designing mission orbit to keep mission orbit perigee above GEO for 30 years
- There is no plan for an End of Life maneuver. However, earlier in mission analysis Amanda Haapala and Cody Short at Purdue found that it may be possible to carefully design the PAM so cumulative Sun perturbations after years lead to a jump in mission orbit perigee (see paper by Short, Howell, Haapala and Dichmann, 2016)

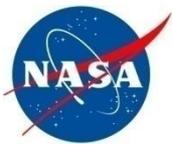




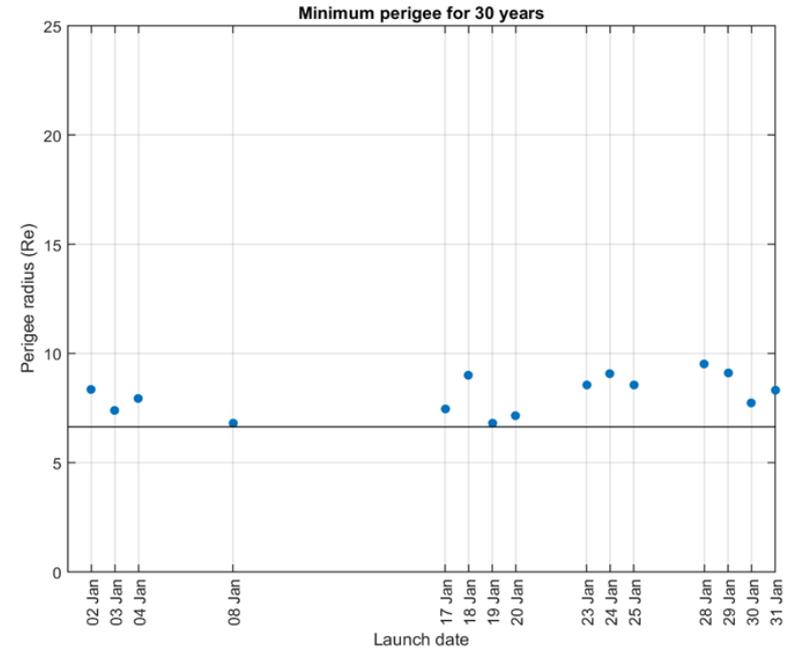
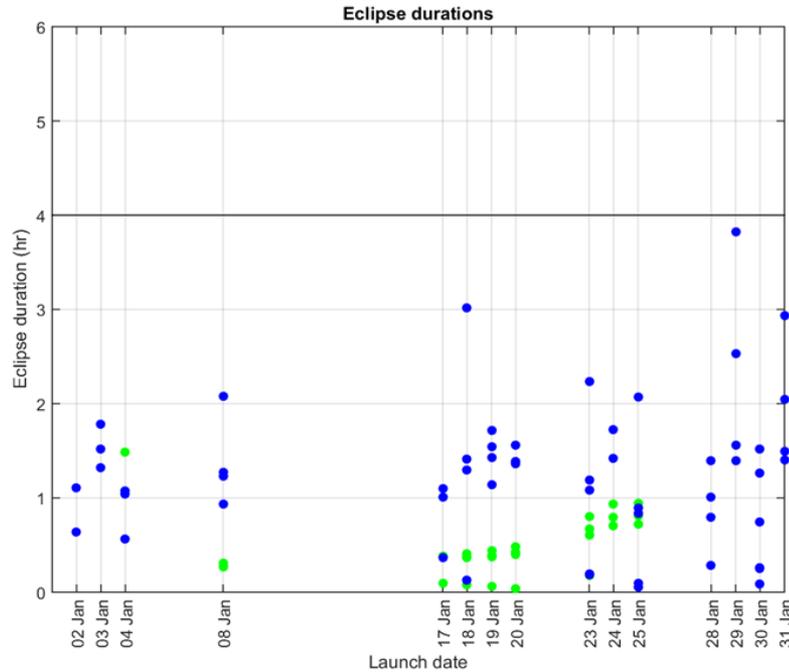
Solutions for Jan 2018



Plot shows deterministic DV only, not statistical DV.
We find 15 feasible launch dates in Jan 2018, well above the 5 required dates.
(Including statistical DV removes 3 dates, 17-19 Jan, leaving 12 launch dates.)



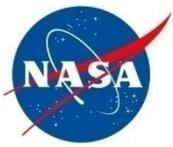
Eclipse Duration & Minimum Perigee



- Green for phasing loops
- Blue for mission orbit

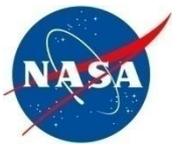
In most cases, eclipse durations remain well below the 4 hour limit

In some cases (8 and 19 Jan), to keep perigee above GEO we adjust PAM to adjust initial mission orbit period by as much as a day

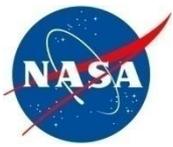


Summary

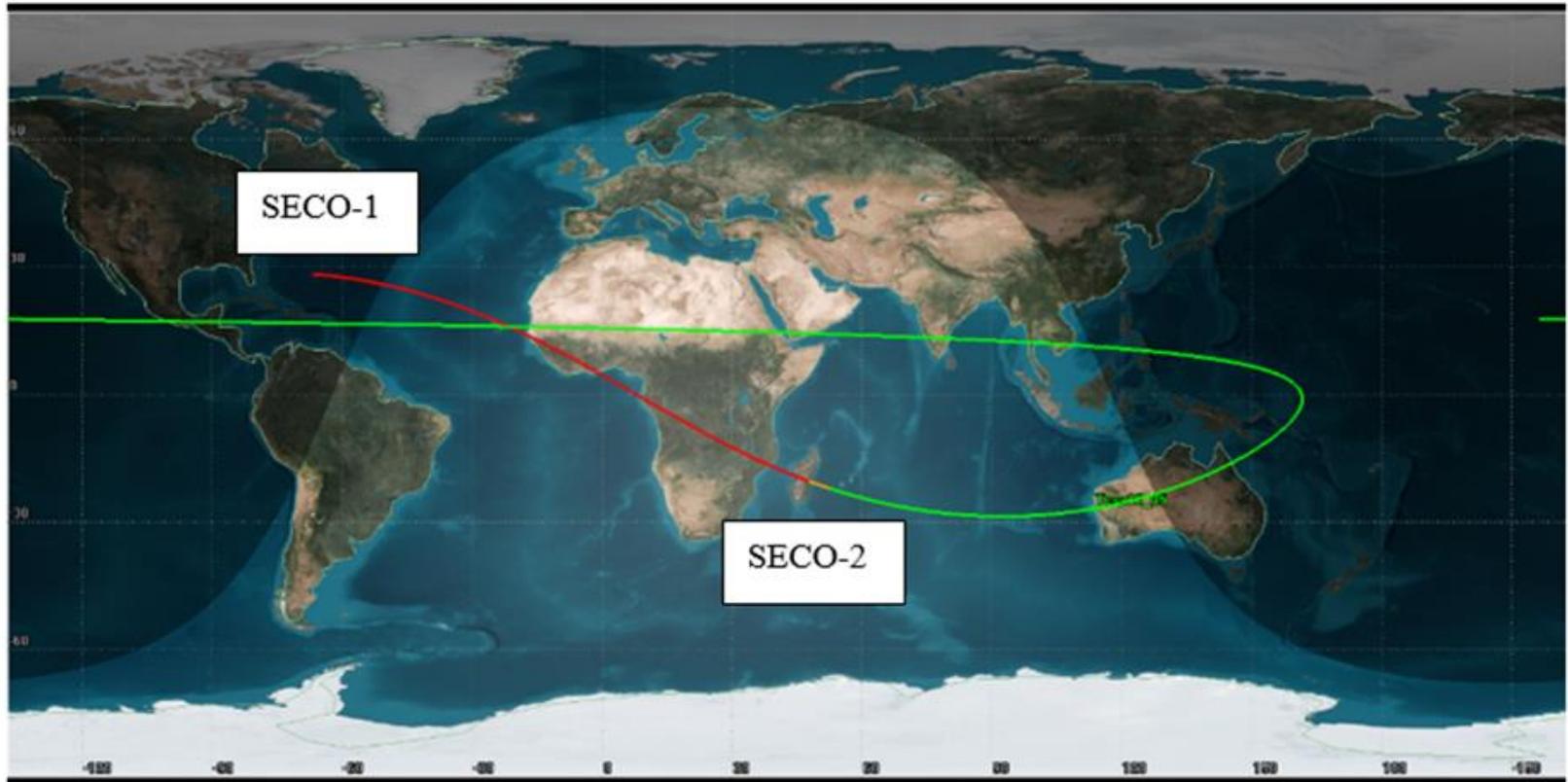
- The TESS mission presents significant trajectory design challenges to satisfy mission constraints and to mitigate risk
- To overcome these challenges we utilize our knowledge of the 3- and 4-body orbit dynamics, including the Tisserand criterion, Kozai mechanism and long-term predictability
 - This knowledge was obtained in part through support of the IBEX and LADEE missions
- Kozai mechanism helps us to meet constraints on eclipse and on mission orbit perigee radius
- LRP condition helps us to achieve operational orbit stability and GEO constraint
- Mission orbits such as TESS and IBEX that meet the LRP condition are predictable for about 30 years



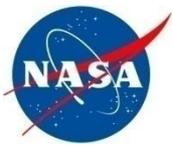
Backup Charts



Ground Track



Short coast: Injection northwest of Africa to southeast of Africa, depending on flyby epoch



Injection Point for range of injection AOP values

