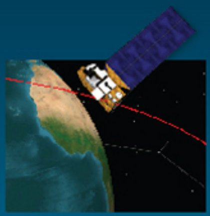
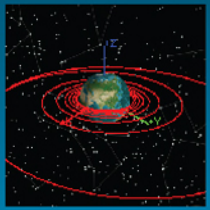
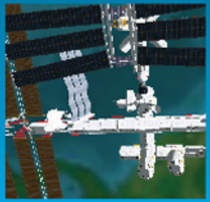
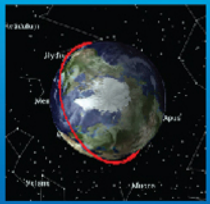
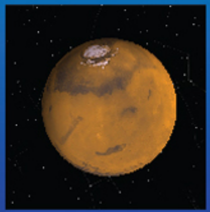
National Aeronautics and Space Administration











Introduction to the GMAT Software

This presentation was written by members of the GMAT team and is used with their permission.

Jason Laing and Mojtaba Abedin

Oct 29, 2014

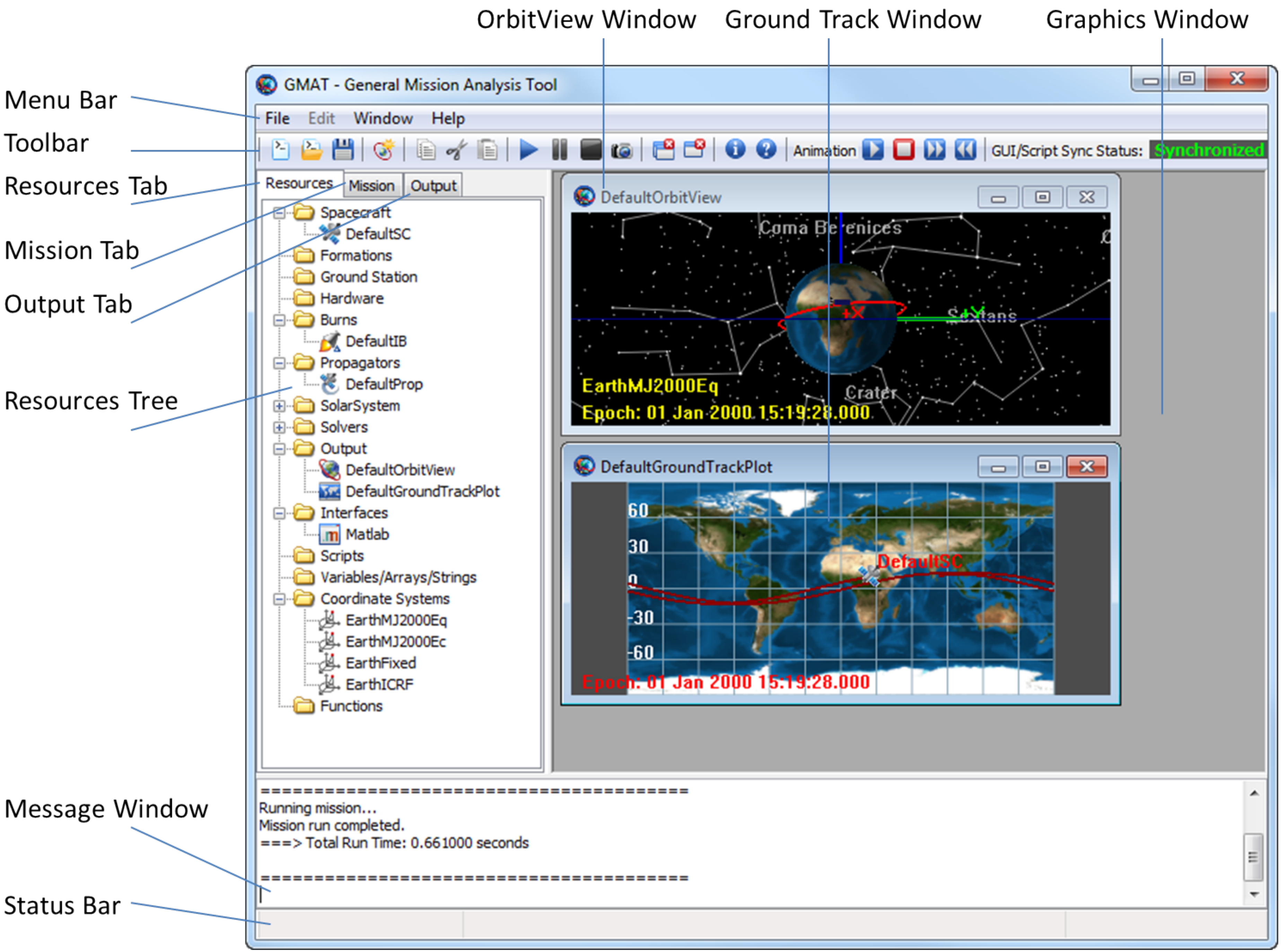
NASA Goddard Space Flight Center

general mission analysis tool

Outline

I. **Key Concepts**
   a. Two Parallel Interfaces
   b. Resources and Commands
   c. Fields and Parameters
   d. Execution Model
II. **Tour of the Graphical User Interface**
   a. GUI Controls
   b. Resources Tree
   c. Mission Tree
   d. Output Tree
   e. OrbitView
III. **Tour of the Script Language**
   a. Basic Syntax
   b. Control Structures
   c. Using Math
   d. Using Parameters
   e. Solvers
   f. Script Editor
   g. Best Practices
IV. **Data Files and Configuration**
V. **Plugins**
VI. **Getting Help**

KEY CONCEPTS

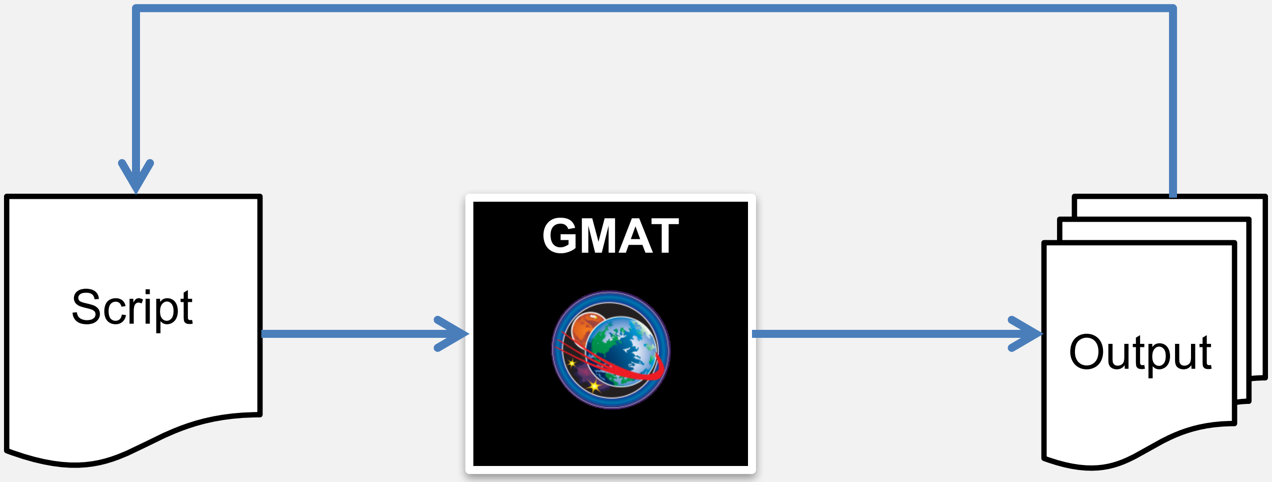

OrbitView Window
Ground Track Window
Graphics Window
Menu Bar
Toolbar
Resources Tab
Mission Tab
Output Tab
Resources Tree
Message Window
Status Bar
GMAT - General Mission Analysis Tool
File
Edit
Window
Help
Animation
GUI/Script Sync Status:
Synchronized
Resources
Mission
Output
Spacecraft
DefaultSC
Formations
Ground Station
Hardware
Burns
DefaultIB
Propagators
DefaultProp
SolarSystem
Solvers
Output
DefaultOrbitView
DefaultGroundTrackPlot
Interfaces
Matlab
Scripts
Variables/Arrays/Strings
Coordinate Systems
EarthMJ2000Eq
EarthMJ2000Ec
EarthFixed
EarthICRF
Functions
DefaultOrbitView
EarthMJ2000Eq
Epoch: 01 Jan 2000 15:19:28.000
DefaultGroundTrackPlot
DefaultSC
Epoch: 01 Jan 2000 15:19:28.000
Running mission...
Mission run completed.
===> Total Run Time: 0.661000 seconds

KC1: Execution Model

- GMAT is like MATLAB:
  - You write a program (a “mission”), then run it to generate output
- Not like Excel
  - Cannot generate output or manipulate results without rerunning

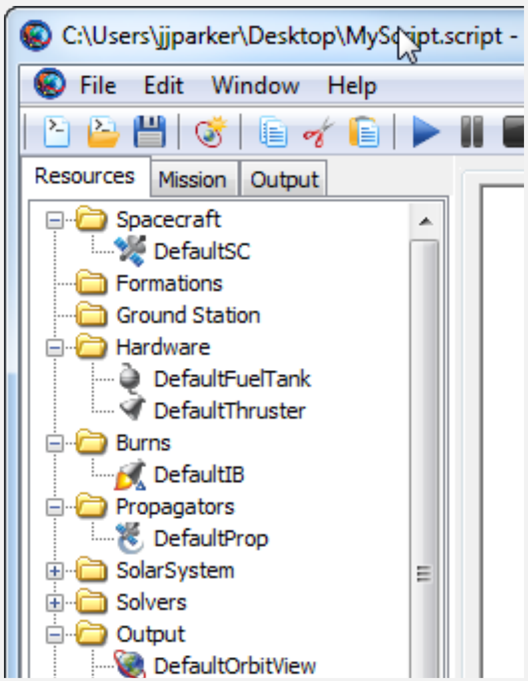
KC1: Execution Model

- Batch execution model



KC2: Two Parallel Interfaces

GUI



Script

```
%General Mission Analysis Tool(GMAT) Script
%Created: 2013-01-23 09:47:07


%----------------------------------------
%---------- Spacecraft
%----------------------------------------

Create Spacecraft DefaultSC;
GMAT DefaultSC.DateFormat = TAIModJulian;
GMAT DefaultSC.Epoch = '21545';
GMAT DefaultSC.CoordinateSystem = EarthMJ2000Eq;
GMAT DefaultSC.DisplayStateType = Cartesian;
GMAT DefaultSC.X = 7100;
GMAT DefaultSC.Y = 0;
GMAT DefaultSC.Z = 1300;
GMAT DefaultSC.VX = 0;
GMAT DefaultSC.VY = 7.35;
GMAT DefaultSC.VZ = 1;
GMAT DefaultSC.DryMass = 850;
GMAT DefaultSC.Cd = 2.2;
```

GUI and script are nearly interchangeable (but not totally).

KC3: Resources and Commands

Resources

- Participants in a GMAT mission
- Represent the “things” that will be manipulated
- Think of them as objects, with properties
- Most are “fixed” when the mission starts

Commands

- Events in a GMAT mission
- Represent the actions taken on the resources
- Think of them as methods or functions

KC4: Fields and Parameters

Fields

- Properties you can set on a resource
- Examples:
  - `Spacecraft.Epoch`
  - `Thruster.DecrementMass`
  - `ReportFile.Filename`

Parameters

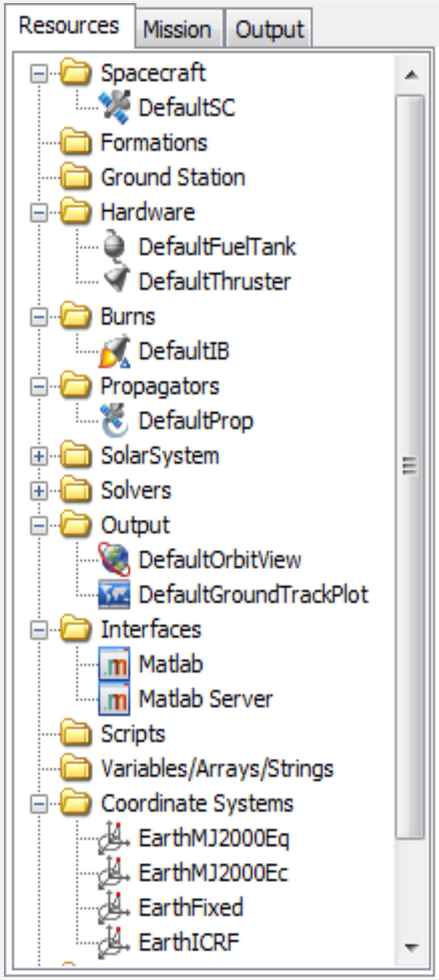
- Properties you can calculate during the mission
- Parameters often have dependencies
- Examples:
  - `Spacecraft.Earth.Altitude`
  - `Spacecraft.EarthMJ2000Eq.BVectorAngle`

- Sometimes a property is both a field and a parameter.
- Examples: `Spacecraft.SMA`, `FuelTank.FuelMass`

TOUR OF THE GRAPHICAL USER INTERFACE

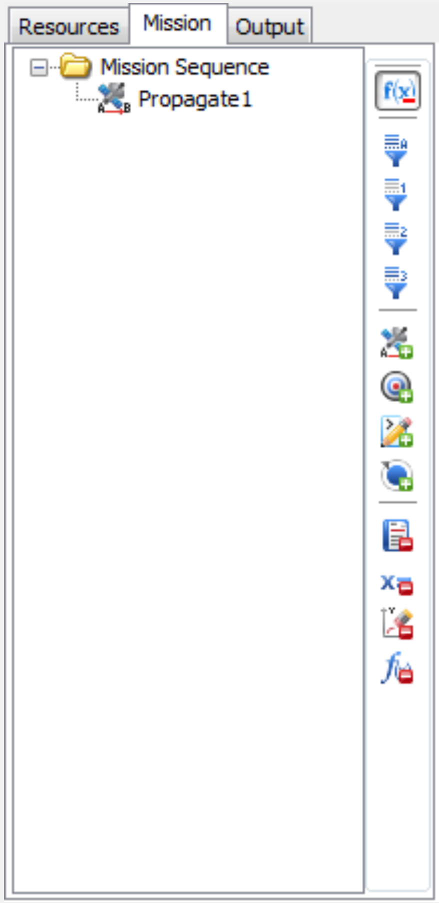
Resource Tree

- Contains all configured resources in the mission
- Grouped into folders by type:
  - Spacecraft
  - Hardware
  - Burns
  - Output
  - SolarSystem



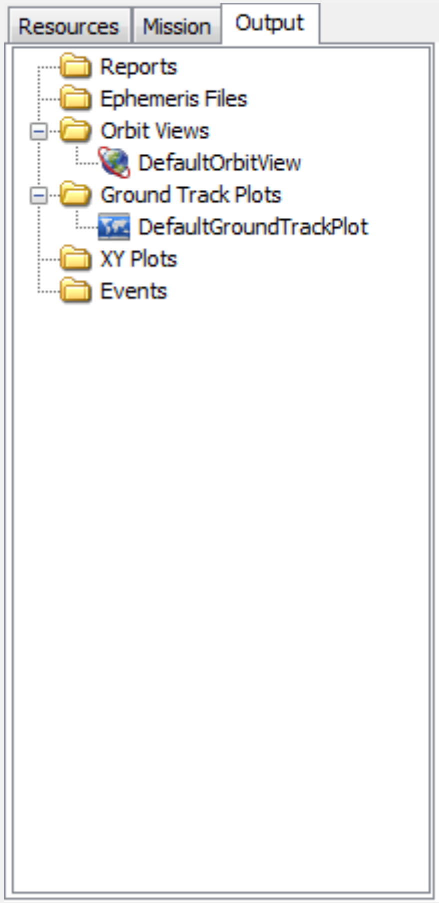
Mission Tree

- Contains the Mission Sequence—sequence of all configured commands
- Special features:
  - Docking & undocking
  - Filtering controls
  - Command Summary



Output Tree

- Contains all output products
- Populated *after* mission execution



OrbitView

- 3D graphics window
- Most complex of the graphical output types
  - Others include: XYPlot (2D plotting), GroundTrackPlot (2D mapping)
- Mouse controls:
  - Left button: rotation
  - Right button: zoom (horizontal motion)
  - Middle button: rotation normal to screen
- Configuration includes:
  - Camera controls
  - Resources to draw
  - Visual elements

TOUR OF THE SCRIPT LANGUAGE

Basic Syntax

- Syntax is based on MATLAB
- Single-line statements w/ optional line continuations
- Case sensitive
- Loosely typed
- Begin/End block statements
- Resources are created before used (except special defaults like SolarSystem)

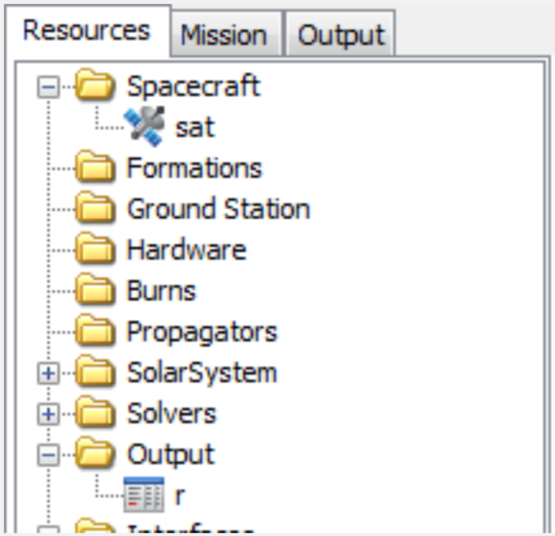
Basic Syntax

- Script is divided into two sections:
  - Initialization (at the top)
  - Mission Sequence (at the bottom)
  - Divided by the `BeginMissionSequence` command
- Initialization -> Resources Tree
  - Static assignment only
- Mission Sequence -> Mission Tree
  - Manipulation of existing resources, cannot create new ones

Basic Syntax

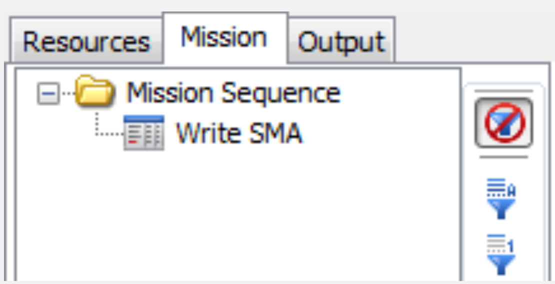
```
Create Spacecraft sat
sat.SMA = 7000

Create ReportFile r
r.Filename = 'MyReport.txt'
```




```
BeginMissionSequence

Report 'Write SMA' r sat.SMA
```



Using Math

- Math syntax is based on MATLAB
- Operators are matrix-aware

Operators	
+	add
-	subtract
*	multiply
/	divide
'	transpose
^	power

Built-in Functions	
sin	cos
tan	asin
acos	atan
atan2	log
log10	exp
DegToRad	RadToDeg
abs	sqrt
norm	det
inv	

Using Math

```
Create Spacecraft SC
SC.SMA = 7100
Create Variable period, mu, pi
mu = 398600.4415

BeginMissionSequence

pi = acos(-1)
period = 2 * pi * sqrt(SC.SMA^3/mu)
```

Using Parameters

- Parameters can have one of two types of dependencies (or neither):
  - Central body
  - Coordinate system
- They are calculated on the fly when they are used:
  - `Spacecraft.MarsFixed.X`
  - `Spacecraft.Earth.BetaAngle`
- If omitted, default dependency is used

Using Parameters

```
Create Spacecraft SC
SC.CoordinateSystem = MarsFixed
Create ReportFile r
BeginMissionSequence

% using parameters
Report r SC.EarthMJ2000Eq.X
Report r SC.Earth.BetaAngle
```

Control Flow

- Three control flow statements:
  - If/Else – execute if a conditional is true
  - While – loop while a condition is true
  - For – loop a certain number of times

```
If SC.Earth.Altitude < 300
     % do a maneuver
Else
     % continue
EndIf
```

Solvers

- Threey types of solvers:
  - `Target` (using `DifferentialCorrector`)
  - `Optimize` (using either optimizer)
  - Estimator
- Similar to loops, with specific nested commands:
  - `Target`: `Vary`, `Achieve`
  - `Optimize`: `Vary`, `NonlinearConstraint`, `Minimize`
- See the tutorials for examples

Getting Help

- For feature-specific information:
  - Help button on feature panel
- For scripting help:
  - "Show Script" button on feature panel
- Overall information:
  - GMAT User Guide (Help > Contents)
  - Updated copy: http://gmat.sf.net/docs/nightly

Community Resources

- GMAT Wiki:
  - http://gmatcentral.org/
- User Forum
  - http://forums.gmatcentral.org/
- Mailing lists:
  - gmat-users@lists.sourceforge.net
  - gmat-developers@lists.sourceforge.net
  - Subscribe at http://sf.net/projects/gmat

National Aeronautics and Space Administration

General Mission Analysis Tool (GMAT)

GMAT Application to GSFC Mission Design

Steven P. Hughes

14 Mar. 2016

This presentation is a based on presentations provided by the GMAT project, the TESS project, and the OSIRIS-REx project used with their permission. Author attributions are listed at the beginning of each section.

general mission analysis tool

www.nasa.gov

Outline

- GMAT Overview and Status
- Usage Basics
- GMAT Application to TESS
- GMAT Application to OSIRIS

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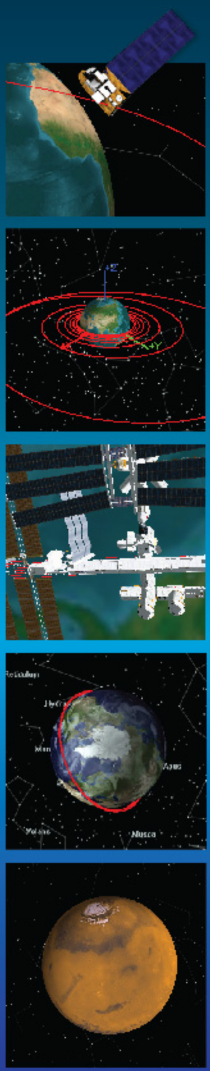

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GMAT Overview and Status

This presentation was written by members of the GMAT team and is used with their permission.

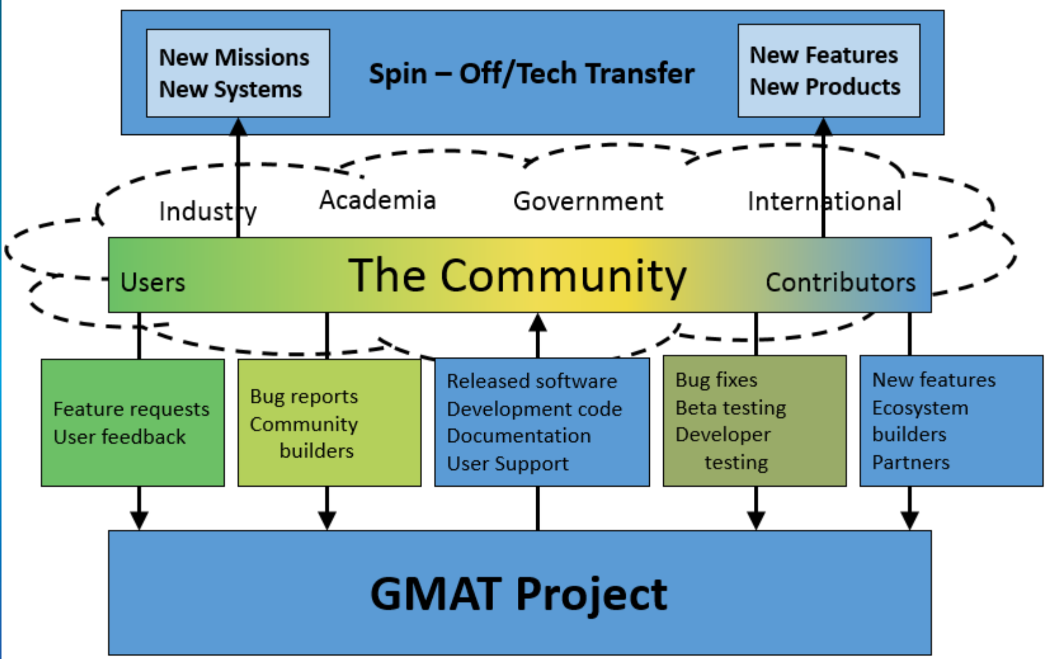
S. Hughes and T. Grubb,
July 17, 2015

Mission Design and Nav. Applications

- Orbit design, optimization, and selection
- Control design
- Visualization
- Orbit product generation and delivery
- Event detection/prediction
- Fuel bookkeeping & lifetime analysis
- Propulsion system sizing
- Launch window analysis
- Sensitivity and Monte Carlo analysis
- Navigation data simulation
- Orbit determination
- Maneuver planning and calibration
- Maneuver Support and reconstruction
- End-of-Life modelling
- Ephemeris prediction

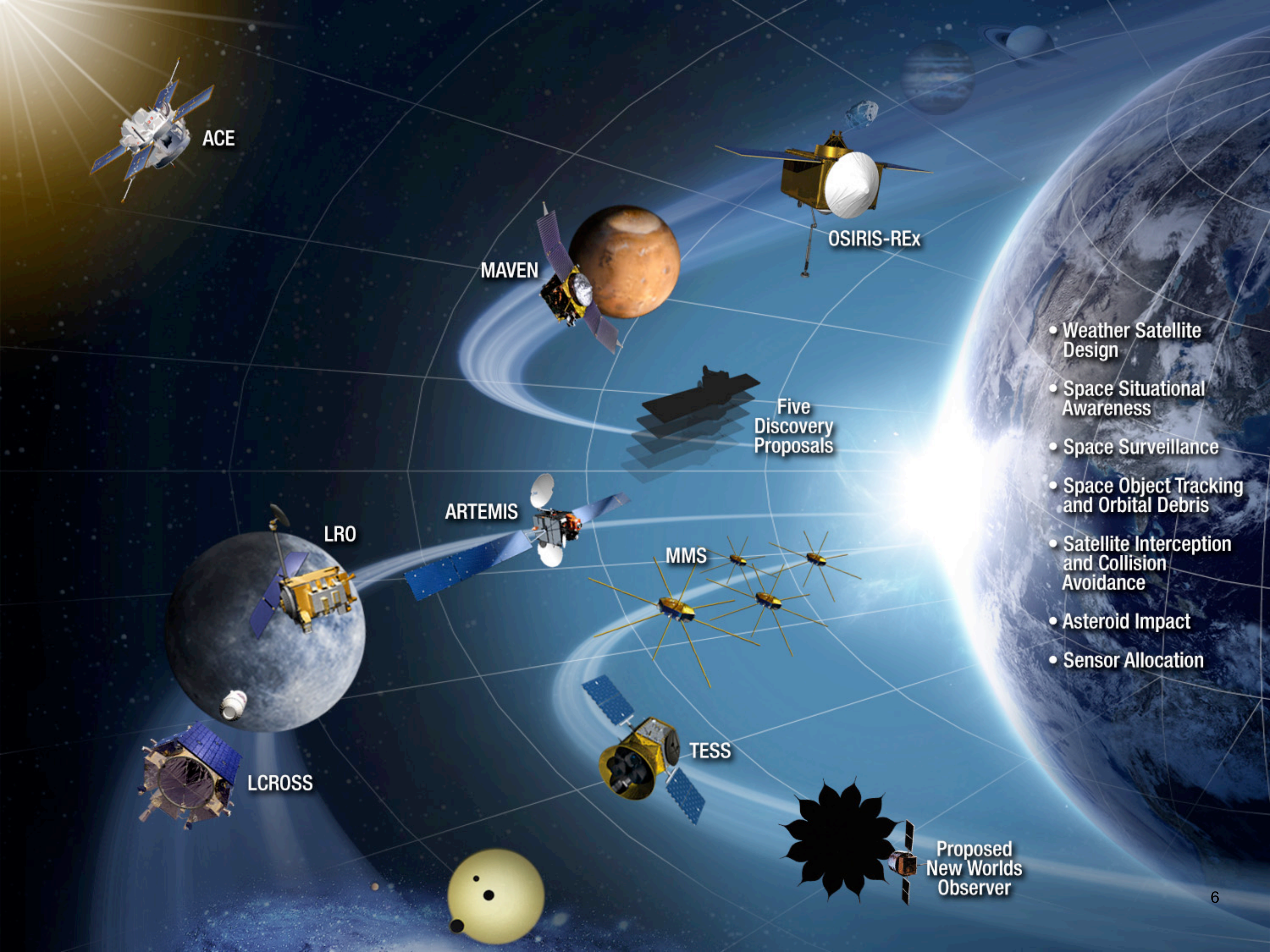
GMAT – Rocket Science for Everyone

The Model

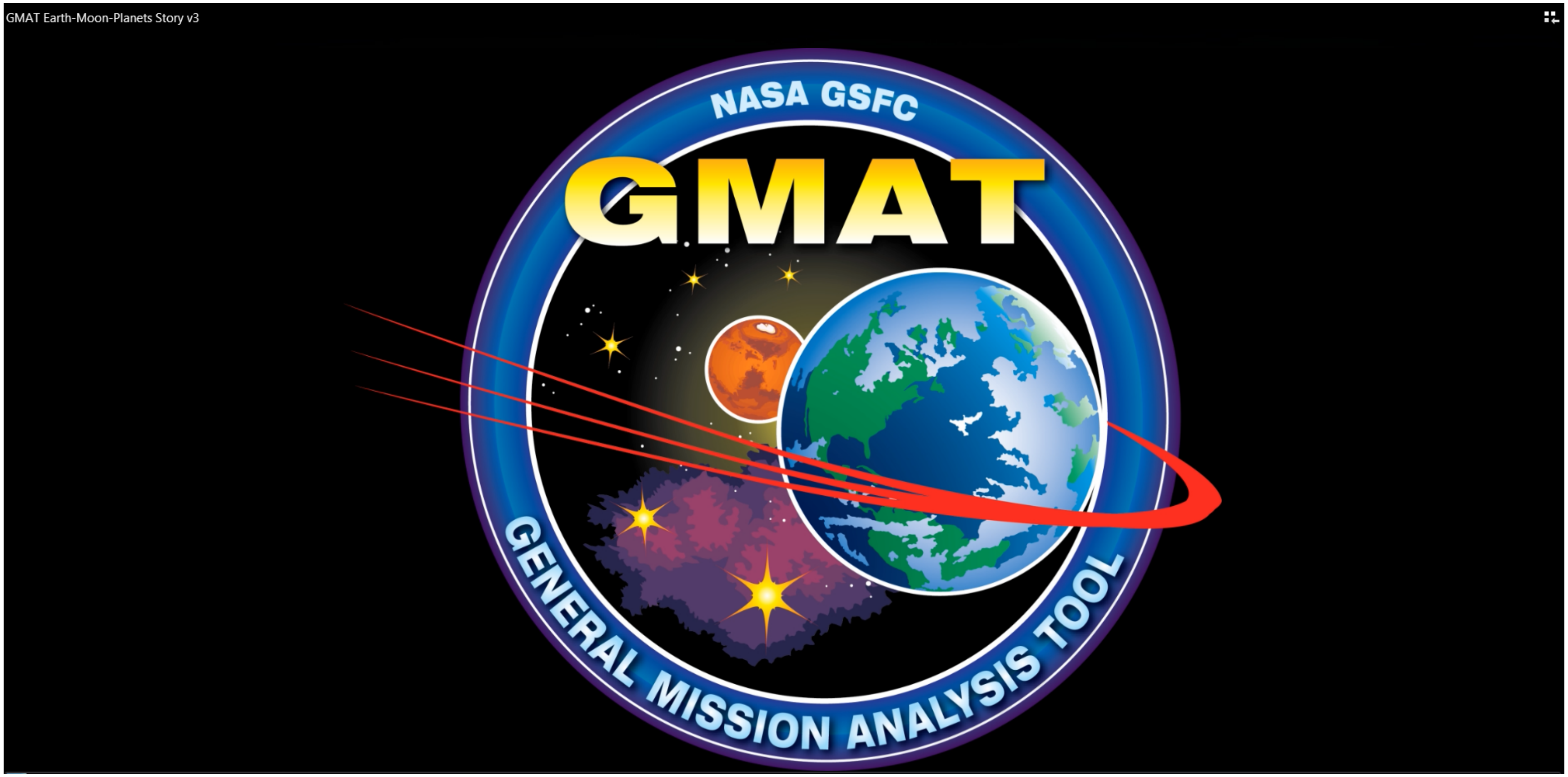


Benefits

- Maximal Tech. Transfer
- Visibility and Transparency
- Cost Effective


ACE
MAVEN
OSIRIS-REx
Five Discovery Proposals
ARTEMIS
LRO
MMS
TESS
LCROSS
Proposed New Worlds Observer
• Weather Satellite Design
• Space Situational Awareness
• Space Surveillance
• Space Object Tracking and Orbital Debris
• Satellite Interception and Collision Avoidance
• Asteroid Impact
• Sensor Allocation

GMAT In Action

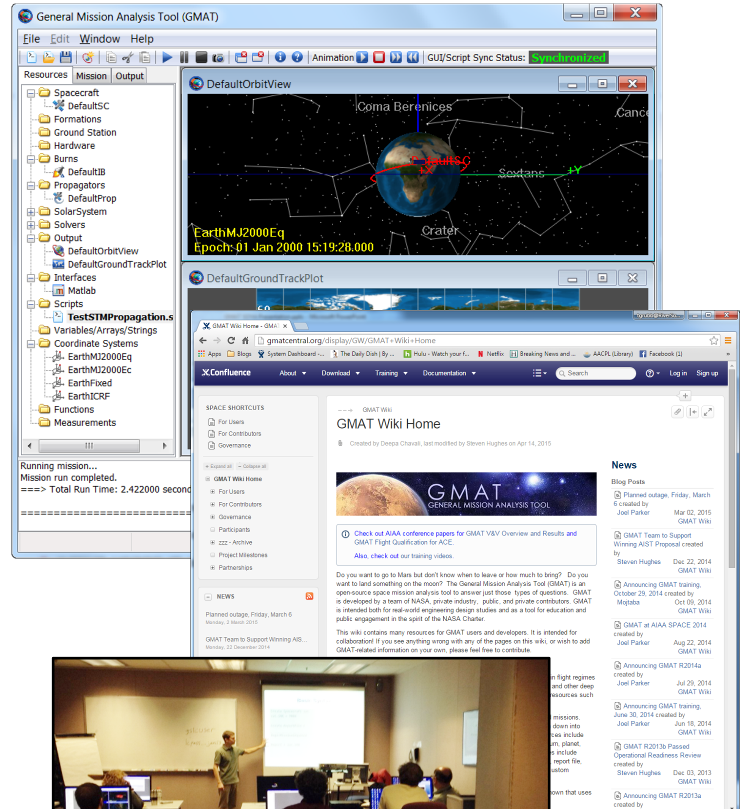


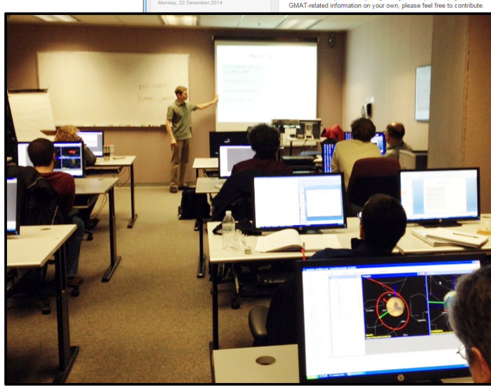
System Characteristics

- World-class quality software
  - TRL 9, Class B, (Part of Center-wide CMMI Accreditation)
  - Over 16,000+ automated script **and GUI** tests

- Large system with extensible design
  - 540k C++ LOC Core
  - Script, GUI, and plugin interfaces
  - 2 Interfaces to external systems (MATLAB and Python (under development)
  - 890k LOC from other libraries (SNOPT (Stanford Business Software). SPICE (JPL NAIF), Wx-Widgets, VF13ad (Harwell), TSPlot Plotting Package (Thinking Systems, Inc.), Mars-GRAM model (MSFC)

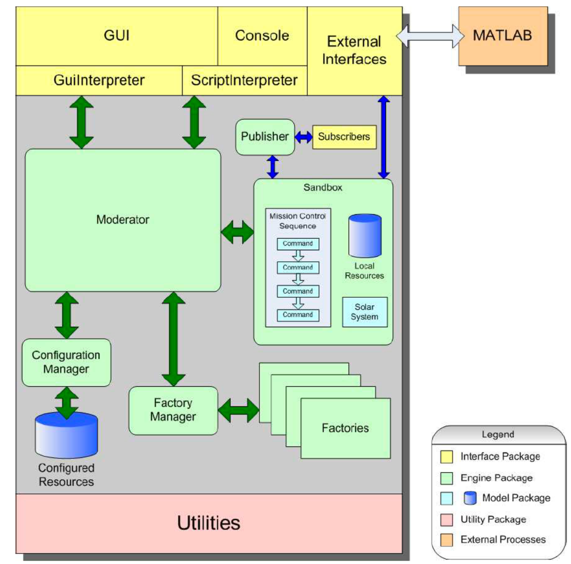
- Enterprise level support
  - Large online support site (wiki, forums, issue tracker, downloads, etc)
  - Extensive Documentation (~850 page User Guide and Reference Manual and ~100 pages of step-by-step tutorials)
  - Training (full-day live training courses and recorded training available via YouTube channel)

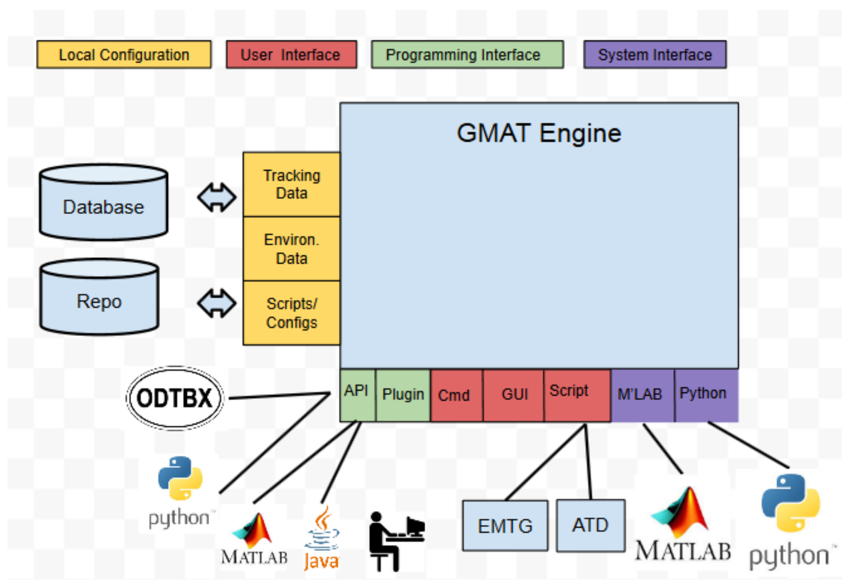




Extensibility

- GMAT's modern architecture was designed for extensibility
- Extensible System Interfaces
  - MATLAB
  - Python
  - API under development
  - Plugins
- Multiple User Interfaces
  - Script
  - GUI
  - Command line
  - API under development
- Extensible model subsystems
  - Dynamics Models
  - Environment Models
  - Estimators
  - Measurements
  - Propagators





Past Release Summary

- GMAT R2013a
  - First production (non-beta) release
  - Focused entirely on QA and documentation
  - Very few new features—but many improved
  - New support for ICRF coordinate systems
- GMAT R2013b (internal)
  - First operationally-certified release
  - Focused on ACE mission requirements
- GMAT R2014a
  - Public release of all R2013b features
  - State representations
  - Attitude models
  - Customizable orbit segment colors
  - Mars-GRAM 2005 atmosphere model
  - LHS parameter dependencies
  - New solver algorithms
- GMAT R2015a
  - GMAT Functions
  - Python Interface
  - Eclipse Location
  - Ground station contact location
  - SNOPT Optimizer
  - Space weather modelling
  - 3D models for celestial bodies
  - Solver status window

Ongoing Navigation Development

- 2009 - 2011
  - Began evaluation of GMAT as a possible navigation tool in 2009
  - Worked with AFRL and IRAD funding to design and implement a navigation subsystem and demonstrate feasibility.
  - Key Conclusion: GMAT could perform OD without significant design changes.
- 2012 – 2013
  - Interplanetary models dynamics models
  - DSN data types
- 2014 – 2015
  - Measurement model re-design based on GEODYN principles
  - User interface re-design for usability based on FDF feedback
  - Testing against flight data
  - Improved batch estimator
  - New data types
  - Measurement editing
  - Improved Reporting
  - Improved bias modelling
  - Improved inverse algorithms for normal equations
  - New Solve-fors
  - Low thrust navigation studies
- Major testing effort in FDF

GMAT was selected as the core tool for GSFC navigation and is preparing for operational use in fall of 2016

Preliminary Navigation Results

Mission	Regime
LRO	Lunar
STEREO-A	Deep Space
SOHO	Libration
DSCOVR	Libration
TDRS-10	GEO
AQUA	LEO
Aura	LEO

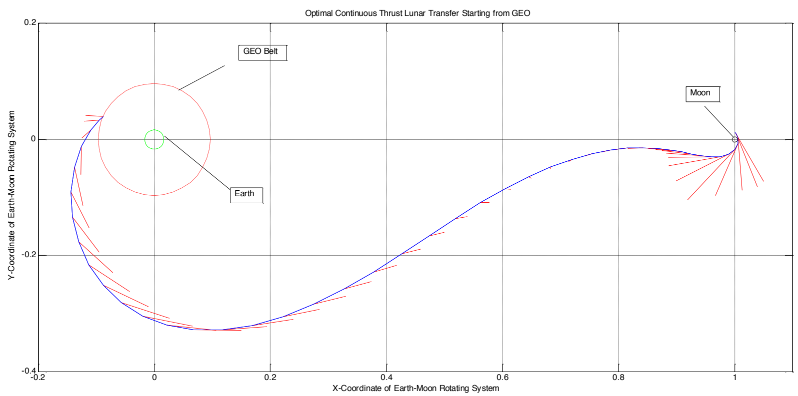
- FDF has used GMAT successfully for OD on a broad range of GSFC flight regimes.
- GMAT navigation solutions are at or near operational quality for those missions now.
- These mission span the GSFC portfolio
- These missions cover the major networks GN, SN, DSN.

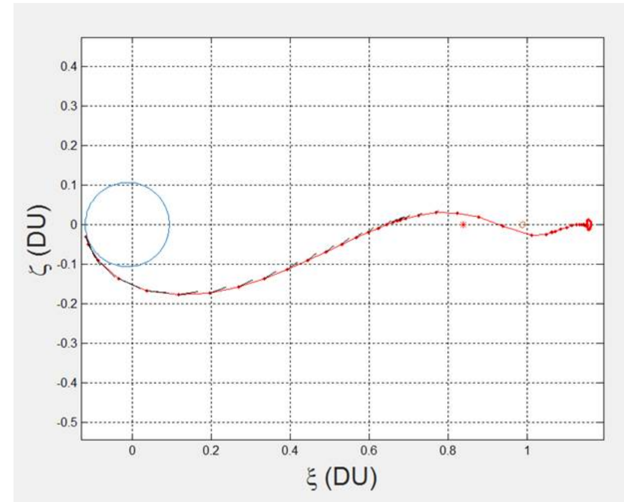
GMAT has not been used for operational navigation yet but planned for fall of 2016.

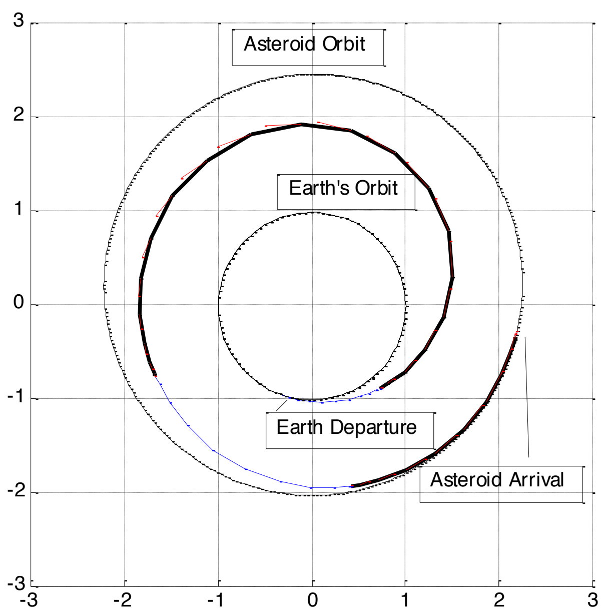
Optimal Control Development

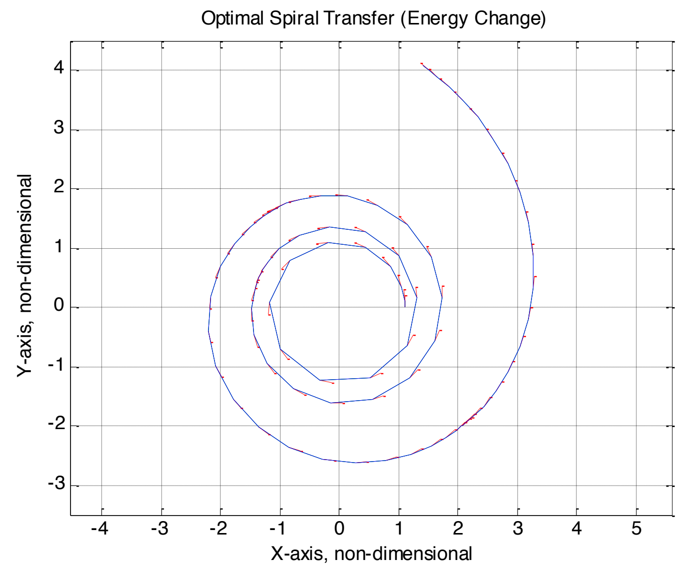
- Feature rich MATLAB prototype based on collocation
  - Prototyped Two "Transcriptions"
  - Algebraic Path Functions
  - Integral and/or Algebraic Cost
  - Boundary Conditions
  - Defect constraint functions
  - Analytic or finite differenced partials
  - Multiple phases and linkage constraints
- Solved 15+ test problems
- Currently migrating to C++ stand-alone
- Integrate into GMAT next year

Low Thrust Optimal Control Results









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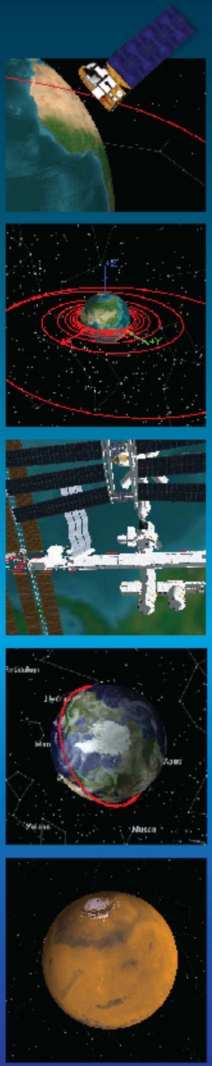

Past and Present Usage

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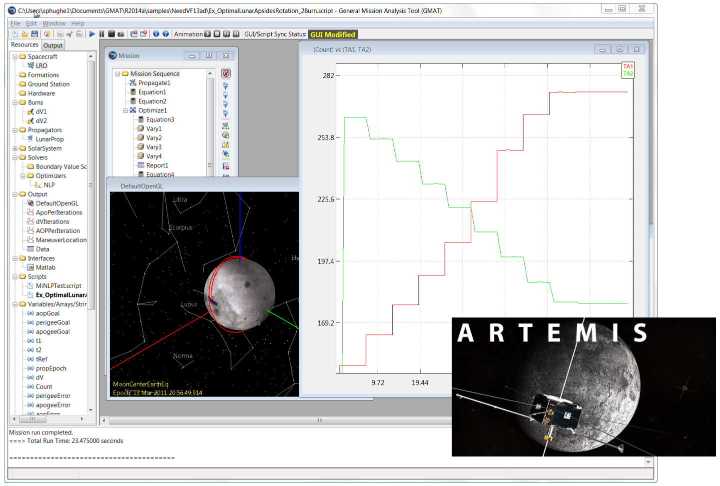
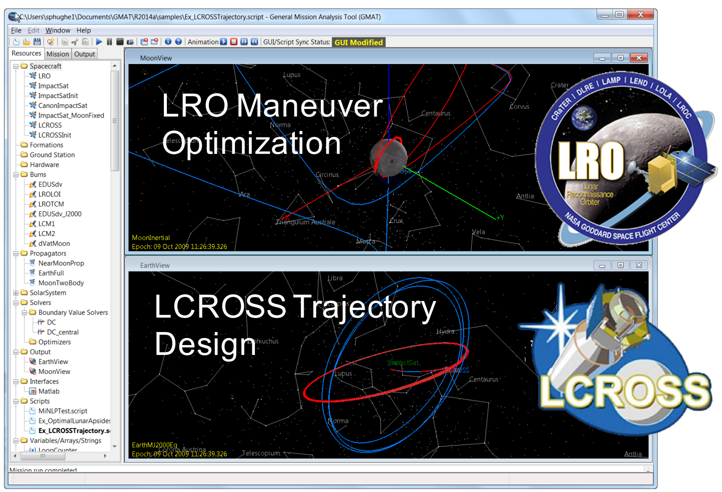
Usage: NASA Missions



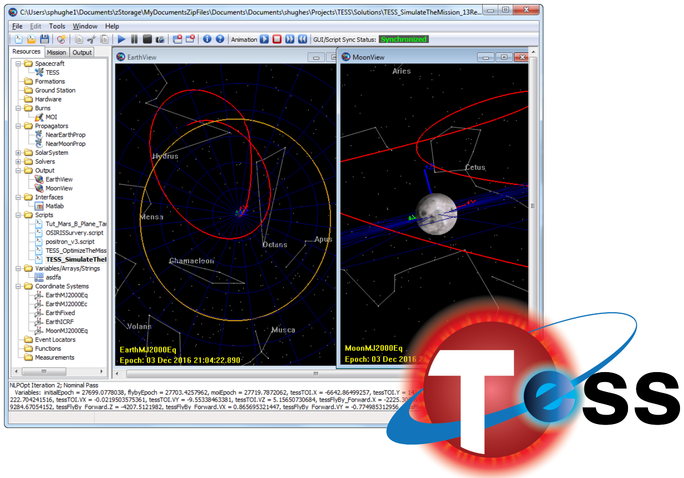


- ARTEMIS – Enabling Innovation
  - *Objective*: Studies acceleration, reconnection, turbulence and electrodynamics of the Moon's Interaction with the Sun,
  - *Application*: Resource-saving solutions have enabled the mission to fly to this day, possibly enabling synergistic science with MMS (March 2015 Launch)
- Lunar Reconnaissance Orbiter (LRO):
  - *Objective*: Mapping and lunar science, launched in June 2009
  - *Application*: Saved the mission 10-15% fuel cost (equivalent to additional year of station-keeping)
- LCROSS
  - *Objective*: Confirm the **presence** or absence of water ice in a permanently shadowed crater near a lunar polar region, June 2009.
  - *Application*: Optimize an entire launch period consisting of dozens of trajectories rapidly and in an automated way, saving weeks of analyst time and enabling larger-scale data analysis than would have been otherwise practical

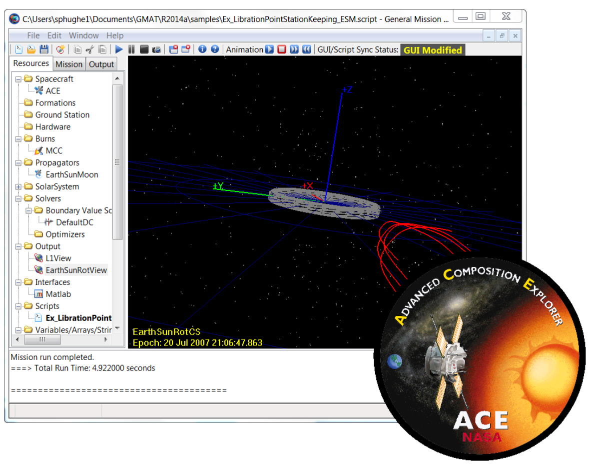
Usage: NASA Missions

- OSIRIS-REx
  - *Objective*: Return and analyze a sample of pristine carbonaceous asteroid regolith
  - *Application*: Used GMAT to optimize the entire 39-day launch period for OSIRIS-REx in a matter of minutes
- Multi-Scale (MMS)
  - *Objective*: Investigate three-dimensional structure and dynamics of the elusively thin and fast-moving electron diffusion region in key regions of re-connection.
  - *Application*: Used GMAT for end-to-end formation modeling and optimization for all phases of the mission, and now use it as the baseline tool for ground system testing
- MAVEN
  - *Objective*: Determine the role that loss of volatile compounds—such as carbon dioxide, nitrogen dioxide, and water—from Mars' atmosphere to space has played through time, giving insight into the history of Mars' atmosphere and climate, liquid water, and planetary habitability
  - *Application*: Used GMAT for mars transfer optimization analysis and to study strategies for Mars orbit maintenance which has unique mission constraints; passing through the atmosphere on each and every orbit

Usage: NASA Missions



- Transiting Exoplanet Survey Satellite (TESS)
  - *Objective*: Survey the brightest stars near the Earth for transiting exoplanets
  - *Application*: Primary mission design and operational maneuver planning tool. Found complete and valid solutions for TESS requirements in ONE week of analyst time compared to MONTHS of effort with other tools that did not find trajectories that met all requirements.



- Advanced Compositional Explorer (ACE)
  - *Objective*: To measure and compare the composition of several samples of matter, including the solar corona, the solar wind, and other interplanetary particle populations, the local interstellar medium (ISM), and galactic matter.
  - *Application*: Used GMAT to rapidly investigate alternative station keeping strategies.

Usage Summary

- 8 NASA missions
- 5+ Discovery proposal efforts
- 15 domestic and international universities
- 6 OGAs
- 12 contributing commercial firms
- 13 commercial firms using in open literature
- 30+ independent peer reviewed publications citing analysis performed using GMAT

GMAT is used world-wide

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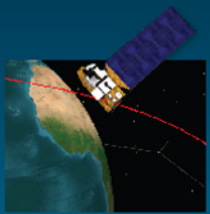

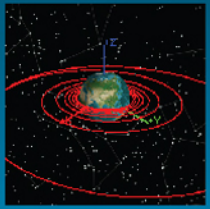

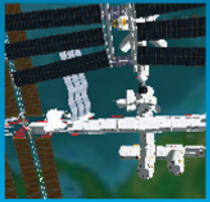

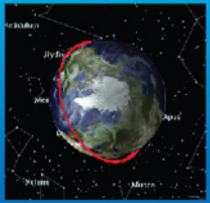

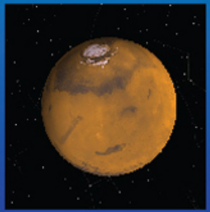

Application of GMAT to TESS Mission Critical Design

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general mission analysis tool

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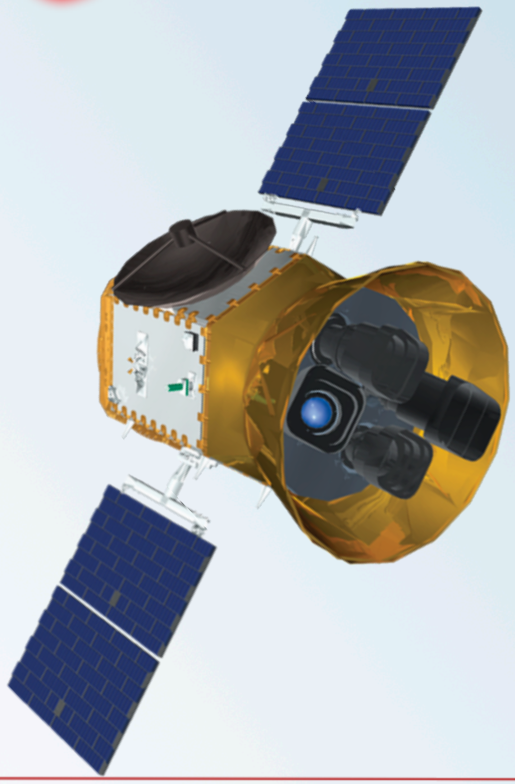
Transiting Exoplanet Survey Satellite



Orbit & Mission Design

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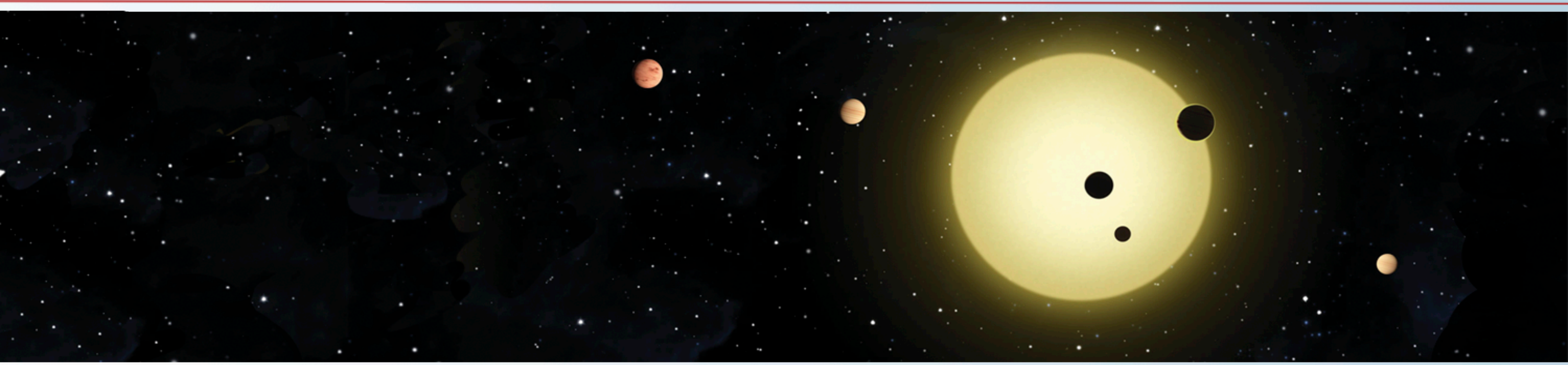
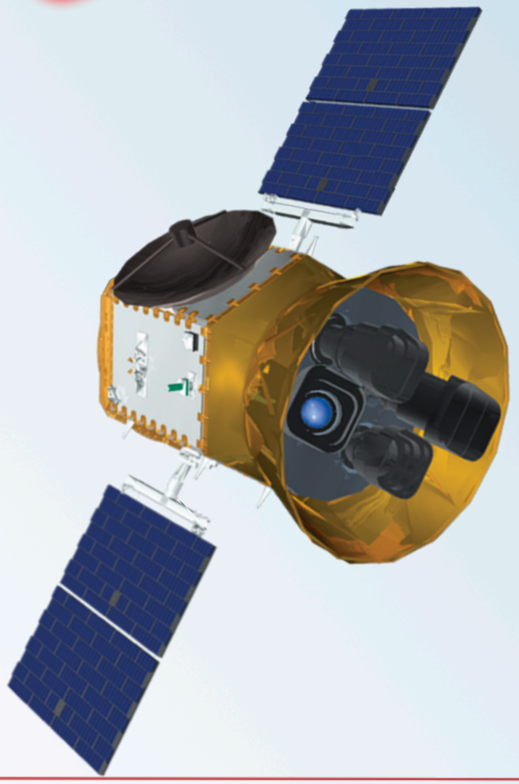
March 11, 2015



Agenda

1. Mission Overview
2. Requirements
3. Trajectory Design Process
4. Solution Generation Process
5. Finite Burn Modeling
6. Launch Vehicle Dispersion Analysis
7. Maneuver Planning
8. Launch Window Analysis
9. Conclusions

Transiting Exoplanet Survey Satellite



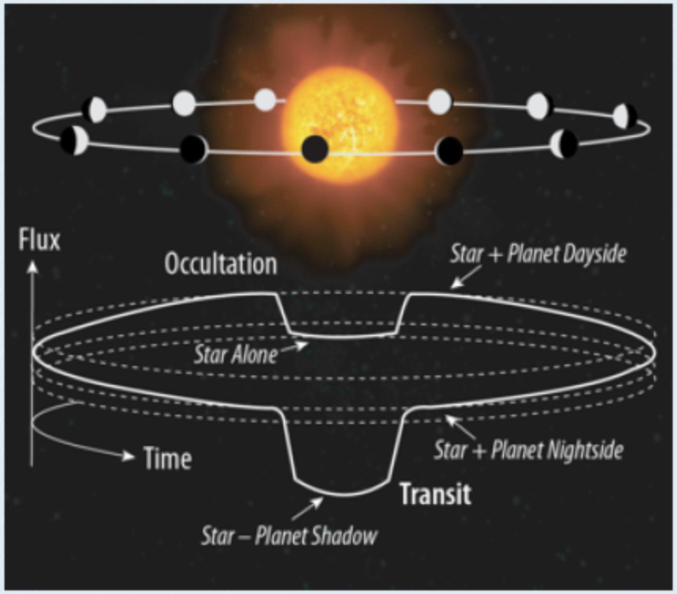
01: Mission Overview

TESS Mission Design Pre-CDR Peer Review

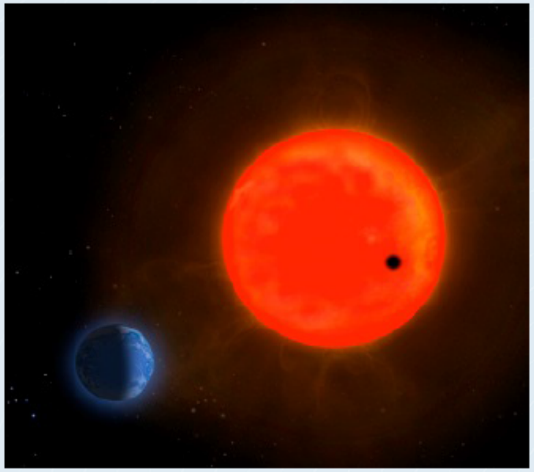
Joel Parker

March 11, 2015

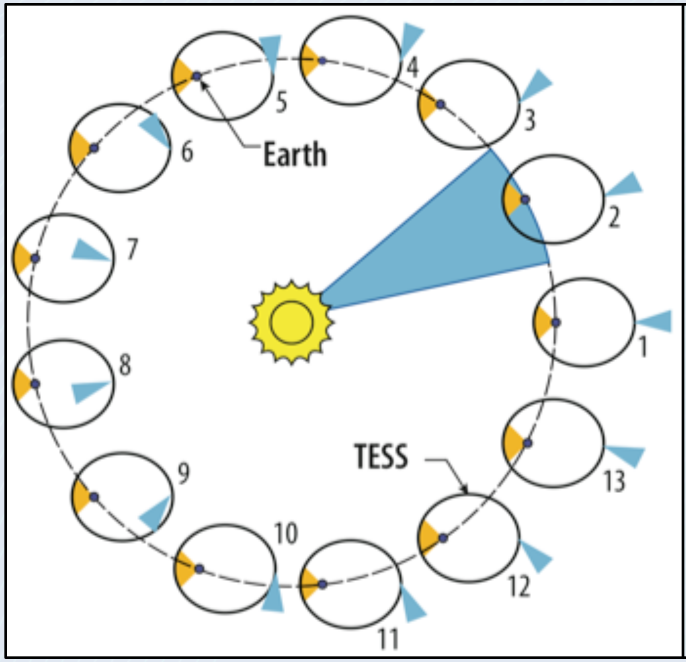
TESS Science Goals and Drivers

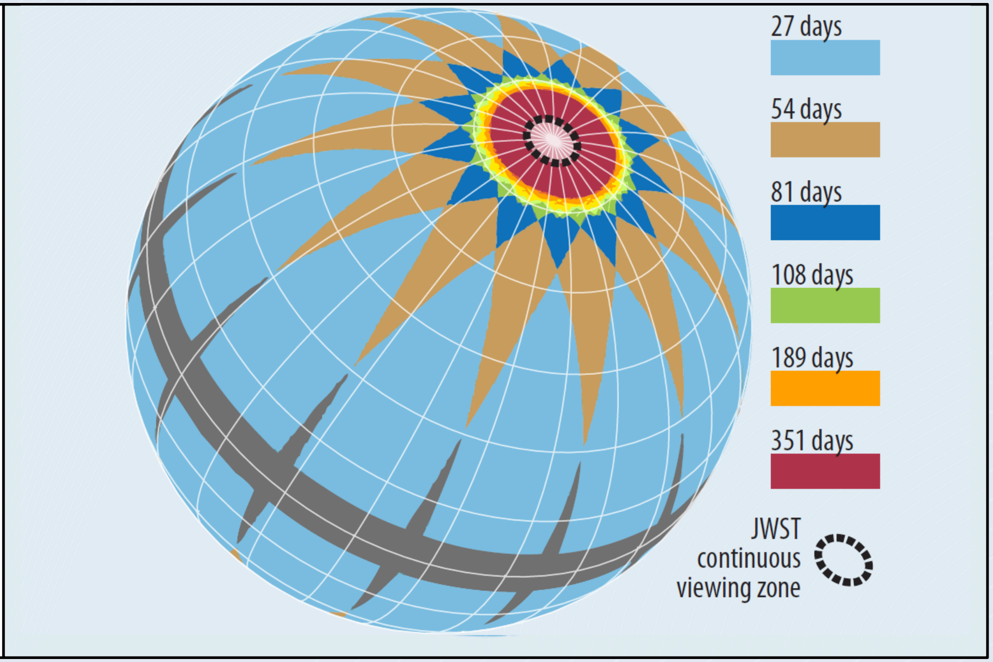


- **Primary Goal:** Discover Transiting Earths and Super-Earths Orbiting Bright, Nearby Stars
  - *Rocky Planets & Water Worlds*
  - *Habitable Planets*
- Discover the "Best" ~1000 **Small** Exoplanets
  - *"Best" Means "Readily Characterizable"*
    - *Bright Host Stars*
    - *Measurable Mass & Atmospheric 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



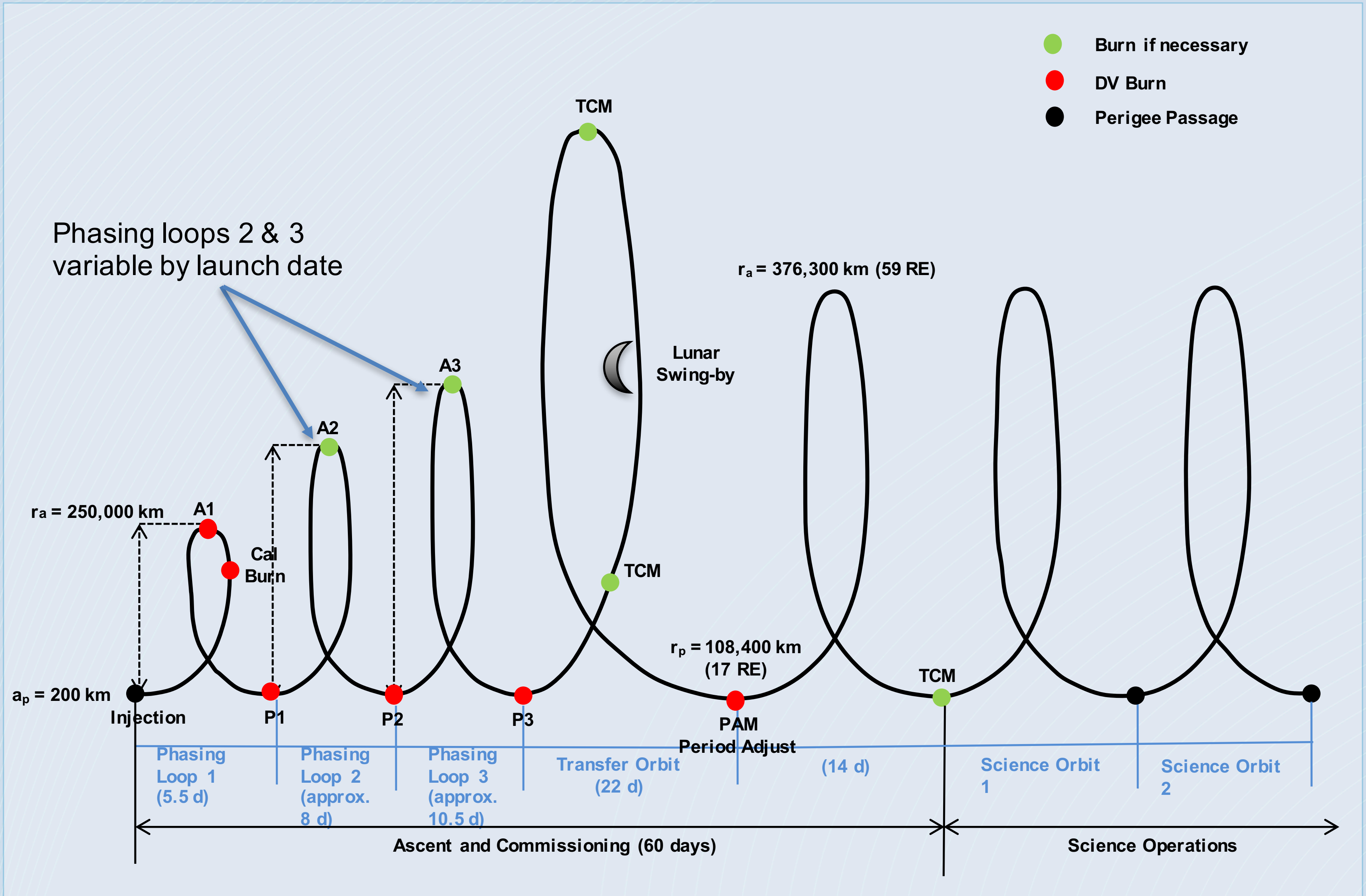


Anti-Solar segments drive +/- 15 deg

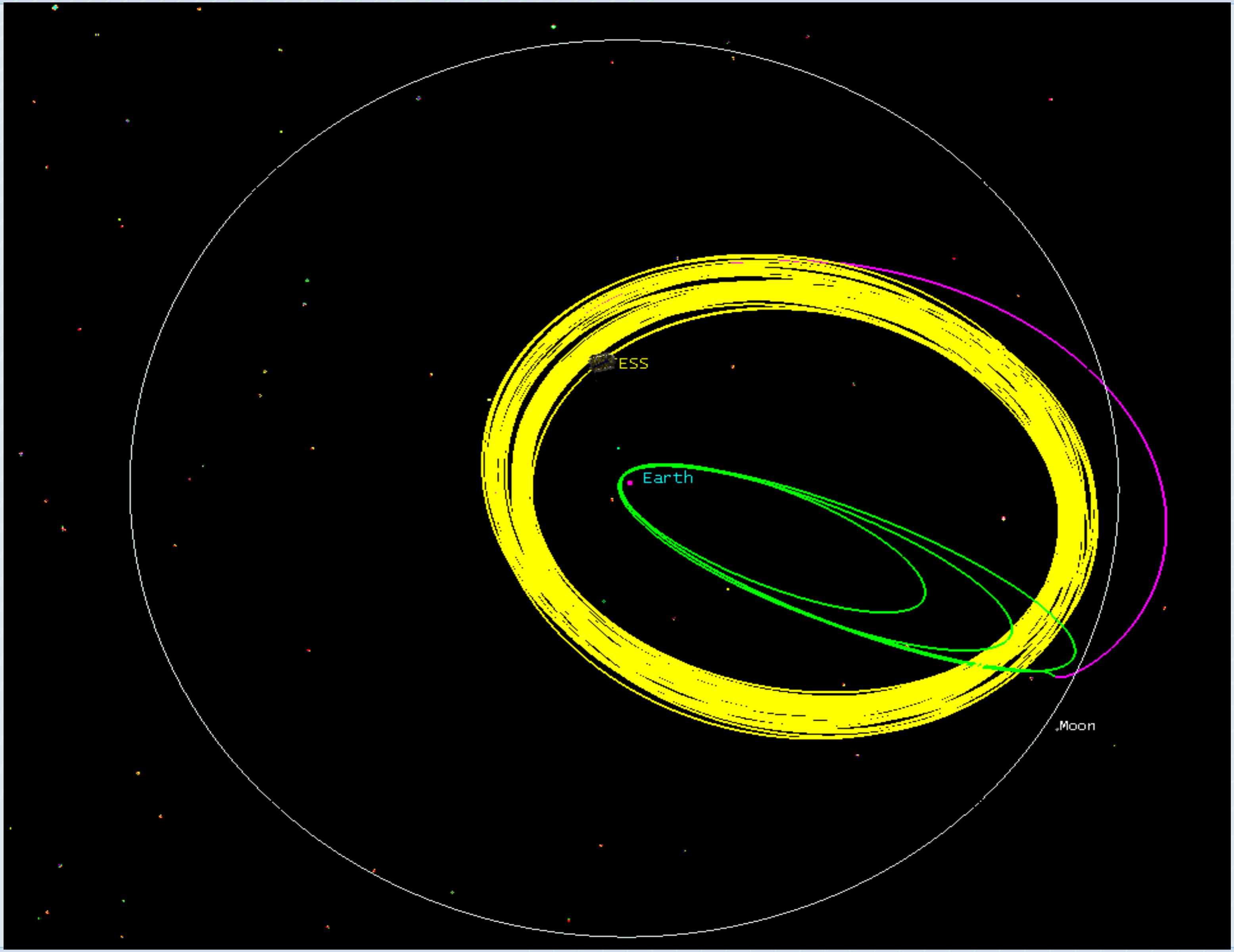
Coverage of ecliptic poles drives Pitch angle (nominally 54 deg)

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

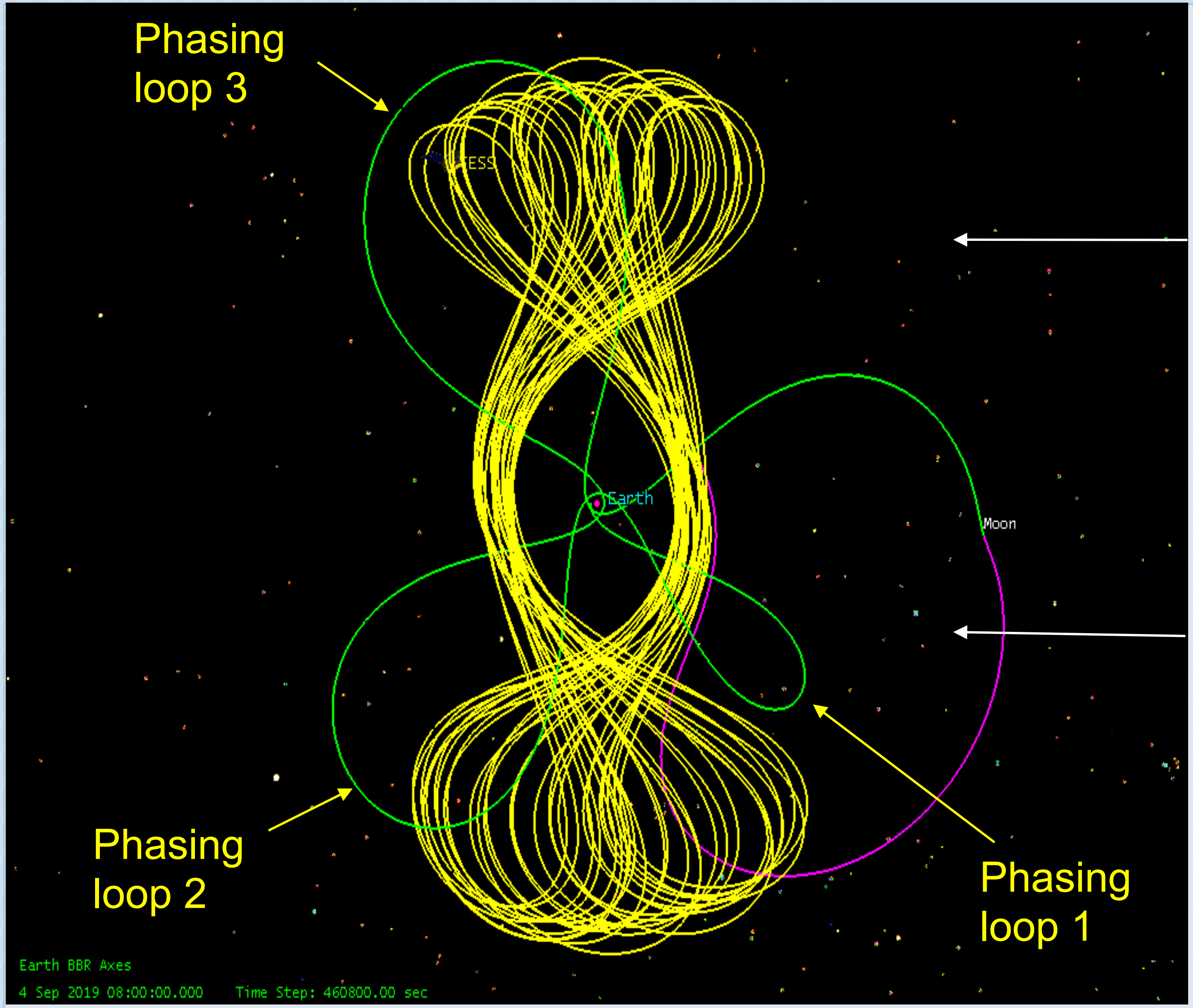
Launch to Science Orbit Timeline



Nominal Aug 10 solution: Inertial frame



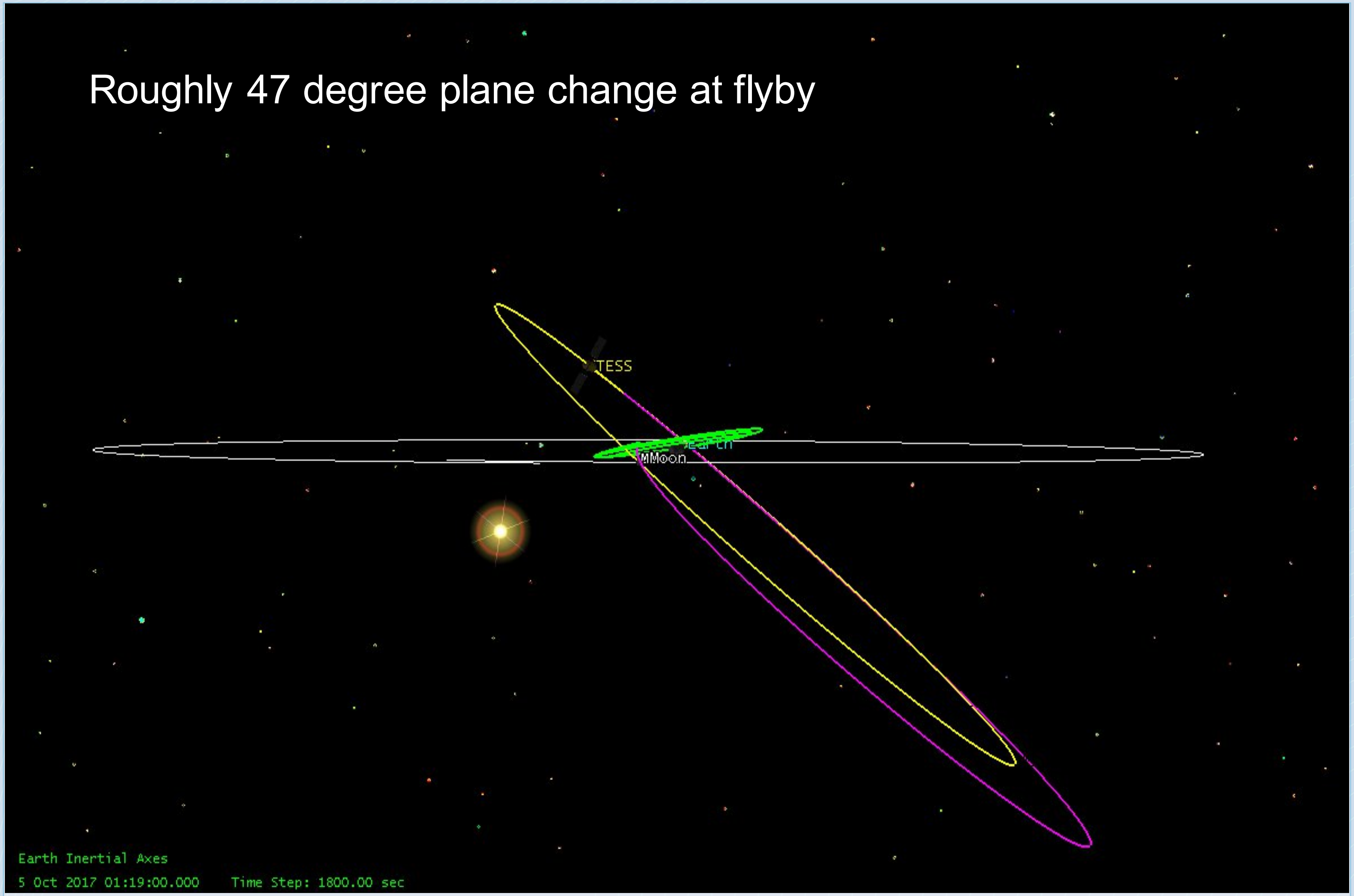
Nominal Aug 10 solution: Rotating frame

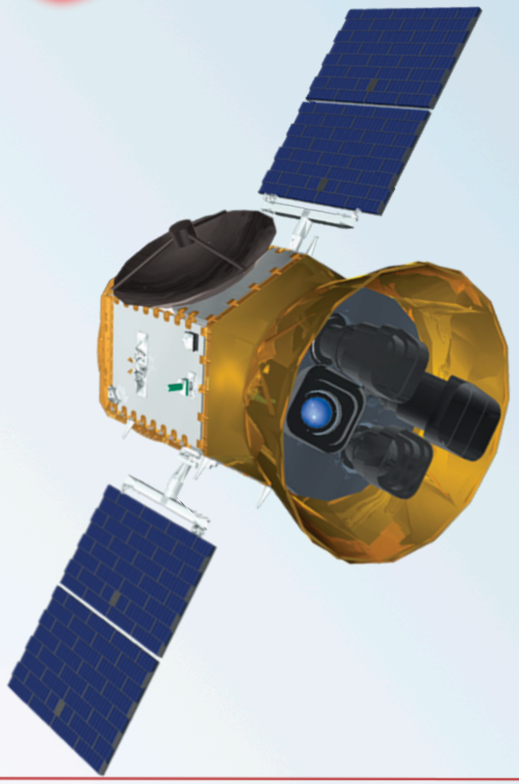


For a loop in the1st quadrant, the Moon is behind and lowers perigee

For a loop in the 4th quadrant, the Moon is ahead and raises perigee

Flyby Plane Change





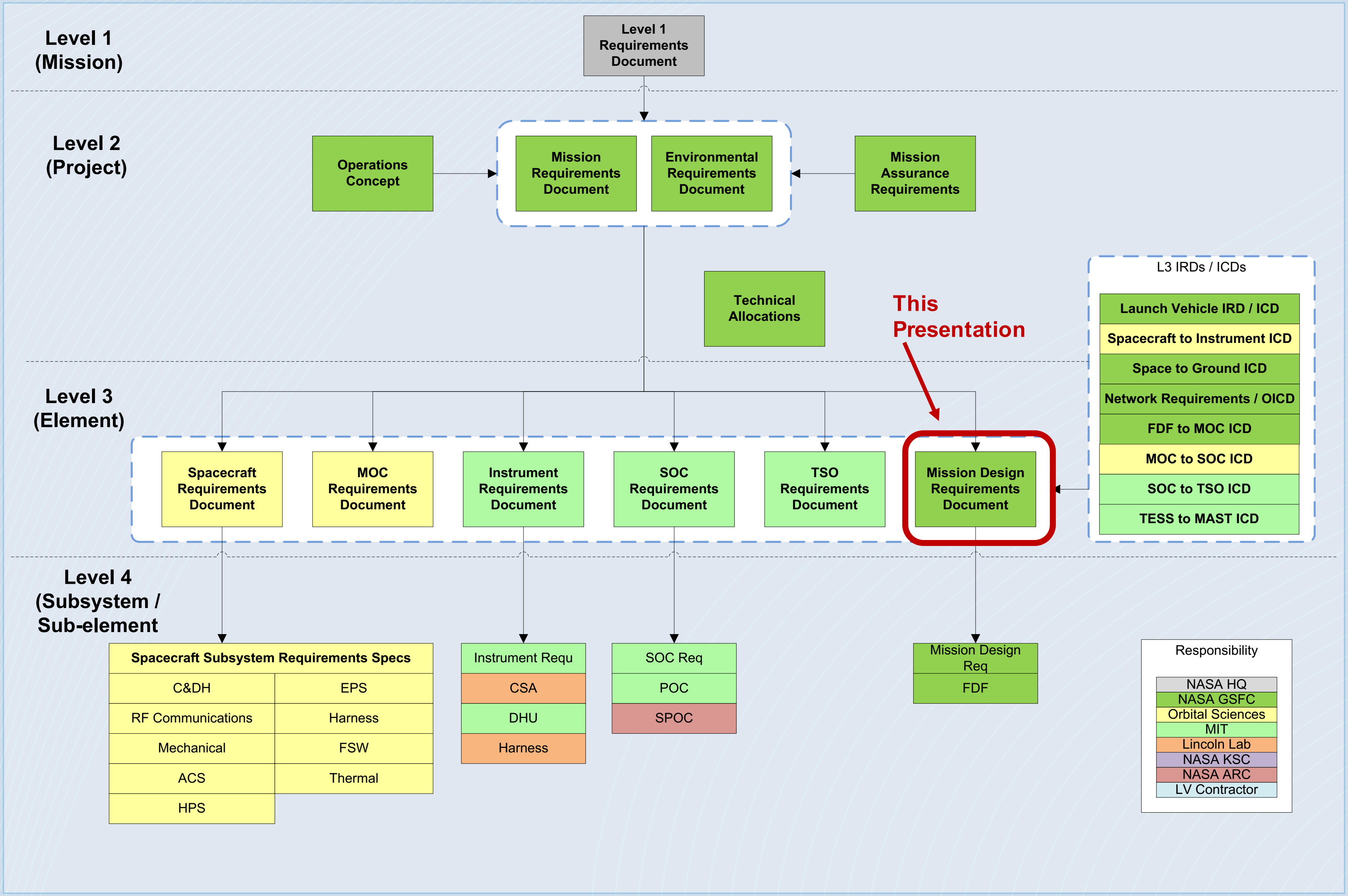
02: Requirements

TESS Mission Design Pre-CDR Peer Review

Joel Parker

March 11, 2015

Requirements Architecture



Key L2 Mission Design Requirements

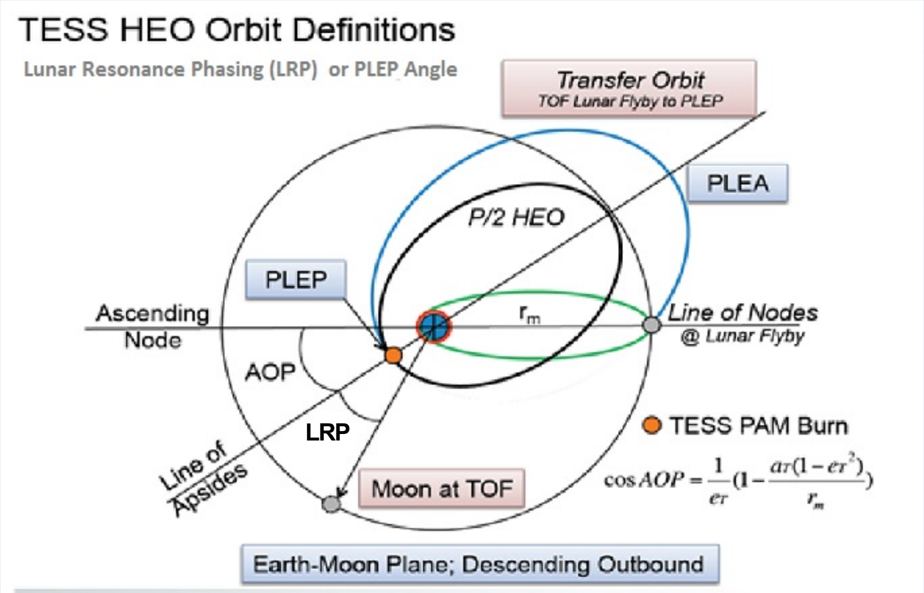
ID	Title	Requirement Summary
MRD_2	Mission Life	2-year mission + 2-month commissioning
MRD_10	Observation Period	HASO duration ≥ 12.5 days per orbit
MRD_54	Launch Period	Launch opportunities on at least 5 days days per lunar cycle
MRD_55	Launch Window	30-Second Launch window
MRD_42	Ascent and Commissioning Duration	Achieve mission orbit within 2 months after launch
MRD_51	Mission Orbit	2:1 lunar-resonant orbit
MRD_52	Maximum Range in LAHO	Perigee < 22 Re
MRD_101	Mission Maximum Range	Apogee < 90 Re
MRD_53	Avoidance of Geosynchronous Orbit	Orbit does not intersect GEO band for mission + 100 years (TBD)
MRD_56	Eclipse Frequency and Duration	No eclipses longer than 5 hours and not to exceed 14 in number (duration = umbra + 0.5*penumbra
MRD_104	Delta-V Allocation	Total ΔV ≤ 215 m/s (99% probability)
MRD_129	Longest Single Maneuver	Longest continuous maneuver ≤ 95 m/s
MRD_85	Sun in Instrument Boresight	FOV exclusion of 54°×126° (TBR) for 15 minutes (TBR)
~~**MRD_64**~~	~~Missed Maneuver~~	~~Achieve mission orbit w/ any single missed/aborted maneuver. (Deleted)~~

Change since PDR Peer Review

Consistent with EXP-TESS-GSFC-RQMT-0001 Rev B

L3 Mission Design Requirements

ID	Parent ID	Title	Requirement	Compliance
L3_FD_1	MRD_10, MRD_51	Mission Orbit SMA	The target mission orbit Semi-Major Axis (SMA) shall be 38 Re.	**Comply.** Design constraint.



Change since PDR Peer Review

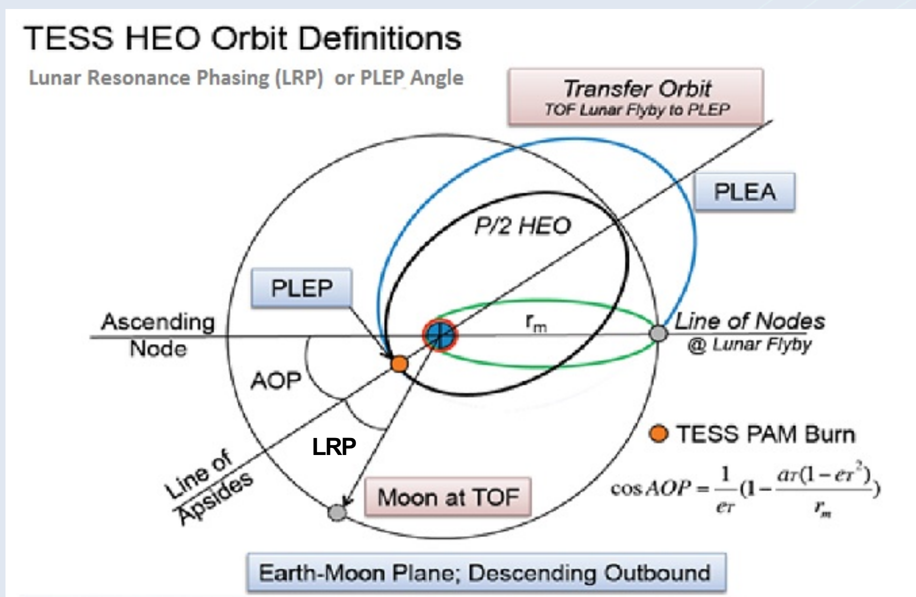
Consistent with EXP-TESS-GSFC-RQMT-0015 Rev (-)

L3 Mission Design Requirements

Waiver pending on orbital debris requirement

ID	Parent ID	Title	Requirement	Compliance
L3_FD_3	MRD_53	Mission Orbit Minimum Perigee	FD shall target a mission orbit with a minimum perigee that shall stay above GEO radius + 200 km.	**Comply.** Results shown to 100 years.
L3_FD_29	MRD_52	Mission Orbit Maximum Perigee	FD shall target a mission orbit with a maximum perigee that shall stay below 22 Re for the duration of the mission.	**Comply.** All <20.5 Re
L3_FD_30	MRD_101	Transfer Orbit Maximum Apogee	FD shall target a lunar flyby that results in a transfer orbit with a maximum apogee less than 90 Re.	**Comply.** All <80 Re

L3_FD_{29, 30, 33} replace old L3_FD_3 in terms of Kozai constant.



Change since PDR Peer Review

Consistent with EXP-TESS-GSFC-RQMT-0015 Rev (-)

L3 Mission Design Requirements

ID	Parent ID	Title	Requirement	Compliance
L3_FD_21	MRD_54	Launch Period	FD shall design for at least 5 launch days in any given Lunar cycle.	**Comply.** At least 9 sol'ns/mo for current period.
L3_FD_22	MRD_55	Launch Window	FD shall design for launch windows of at least 5 minutes during each day of the launch period.	**Comply.** Current strategy meets req.
L3_FD_27	MRD_42	Commissioning Duration	FD shall design the phasing loops and post lunar encounter transfer orbit to achieve mission orbit within 2 months after launch.	**Comply.** PAM at < 43 days.
L3_FD_24	MRD_85	Sun in Instrument Boresight	FD shall design the PAM to occur when the sun is not within a FOV of 54°×126° centered on the camera boresight axis (X-Z plane) for ≥15 minutes.	**Comply.** Basis for sol'n selection.
L3_FD_28	MRD_104	Delta-V Budget	FD shall design ascent-to-mission orbit to require no more than 215 m/s delta-V with 99% probability of success.	**Comply.** See detailed analysis.
L3_FD_25	MRD_129	Maneuver Magnitude	The largest maneuver magnitude shall be <95m/s.	**Comply.** PAM < 75 m/s
L3_FD_4	MRD_56	Eclipse Frequency and Duration	FD shall target a mission sequence that limits the total number of eclipses from LV separation through the end of the prime mission to 2 eclipses with a maximum eclipse duration of 5 hours, and 14 additional eclipses with a maximum eclipse duration of 4 hours.	**Comply.** ~~No more than 11 < 4hr + 1 < 6hr~~ **Needs updating**

Requirements added to flow from L2

Change since PDR Peer Review

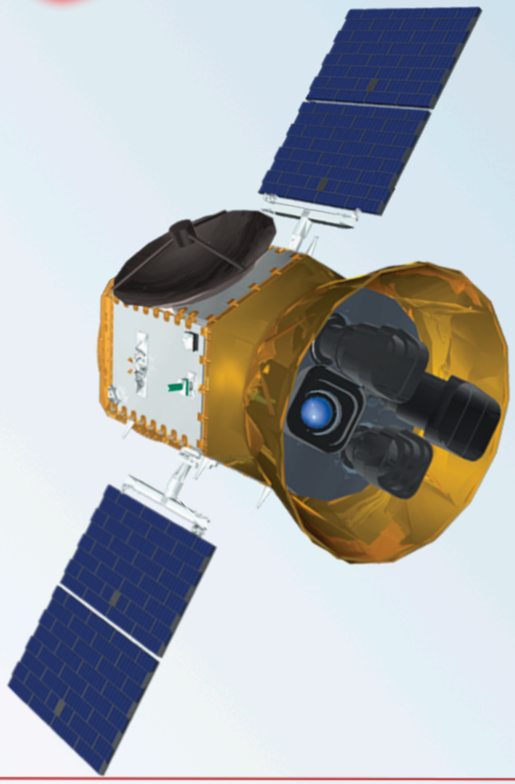
Consistent with EXP-TESS-GSFC-RQMT-0015 Rev (-)

Delta-V Budget

Event	Planned ΔV (m/s)	Current ΔV (m/s) Aug 2017
A1	20	0–17
P1	35	31–50
P2	20	0-20
P3	5	0–8
Period Adjust Maneuver (PAM)	70	56–68
Deterministic Total	**150**	**109–131**
Launch Window Allowance	10	10
Launch Vehicle Dispersion	25	25–31
Trajectory Correction Maneuvers	10	15-26
Margin	20	22-55
Total	**215**	**215**

Consistent with EXP-TESS-GSFC-SER-0001 Rev B

Transiting Exoplanet Survey Satellite

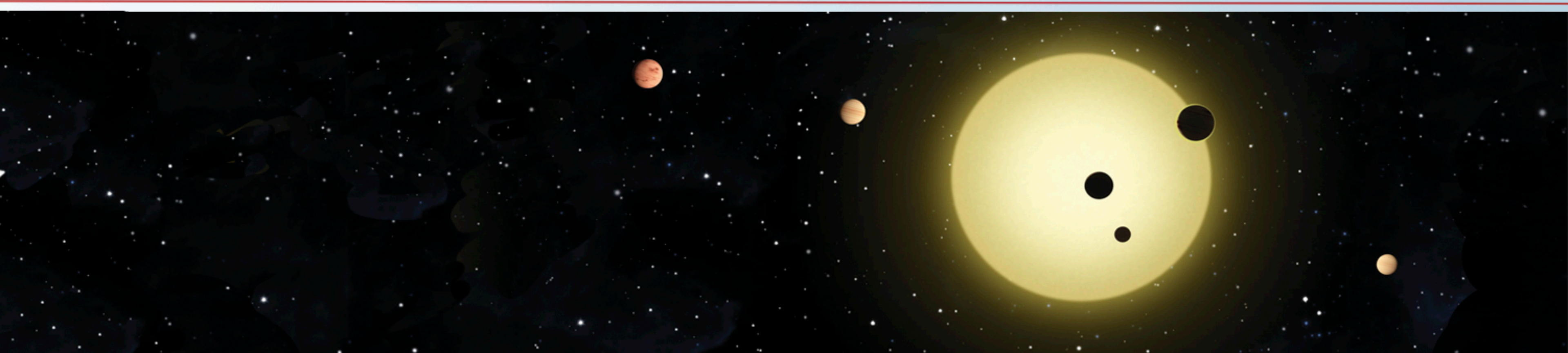


03: Trajectory Design Process

TESS Mission Design Pre-CDR Peer Review

Joel Parker

March 11, 2015



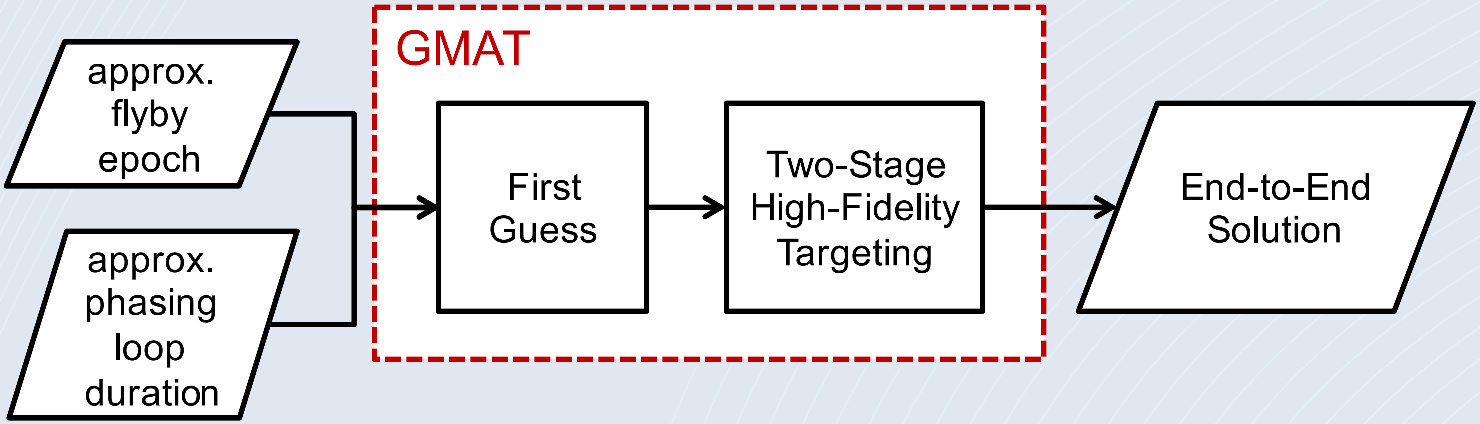
Overview

The TESS trajectory design process is based on three components:

- Theoretical basis
  - *Kozai constant*
  - *Tisserand condition*
- Two-body patched-conic first guess
  - *Implementation of theory to approximate final trajectory*
- High-fidelity targeting
  - *Transitions approximate first guess to realistic final solution*

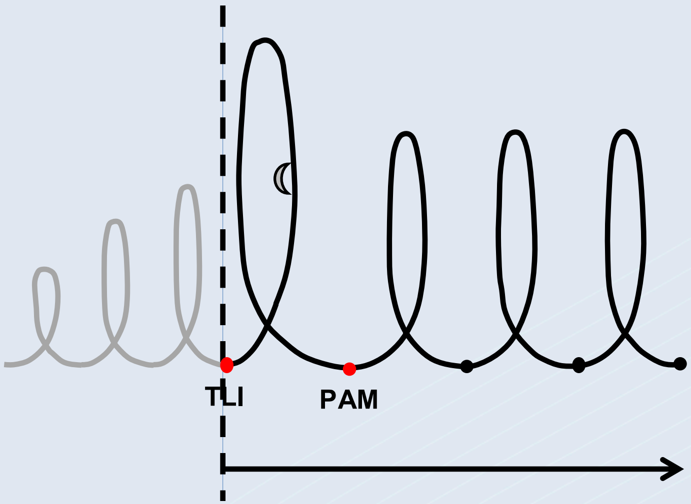
Implementation Overview

- General Mission Analysis Tool (GMAT) used for implementation of design
  - *GSFC's in-house high-fidelity trajectory design software*
- Uses first guess to seed numerical targeting algorithm

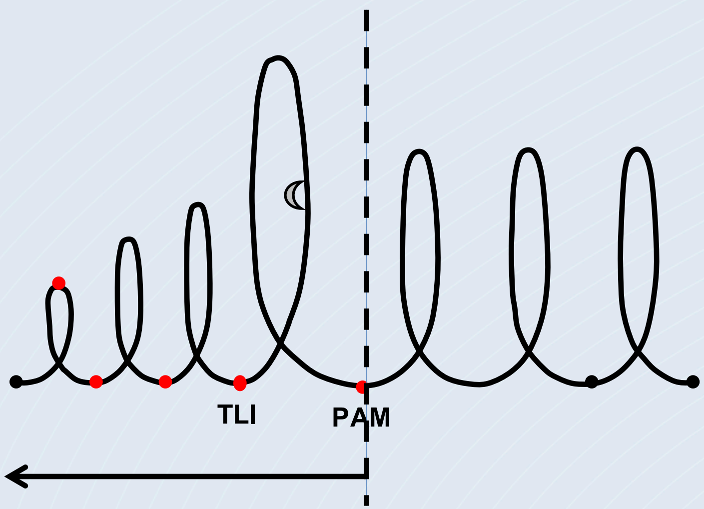


GMAT Design Approach

- Two targeting stages
- Stage 1: Design from Translunar Injection (TLI) through flyby to Science Orbit
  - *Multiple-shooting process*
  - *Starts with patched-conic first guess*

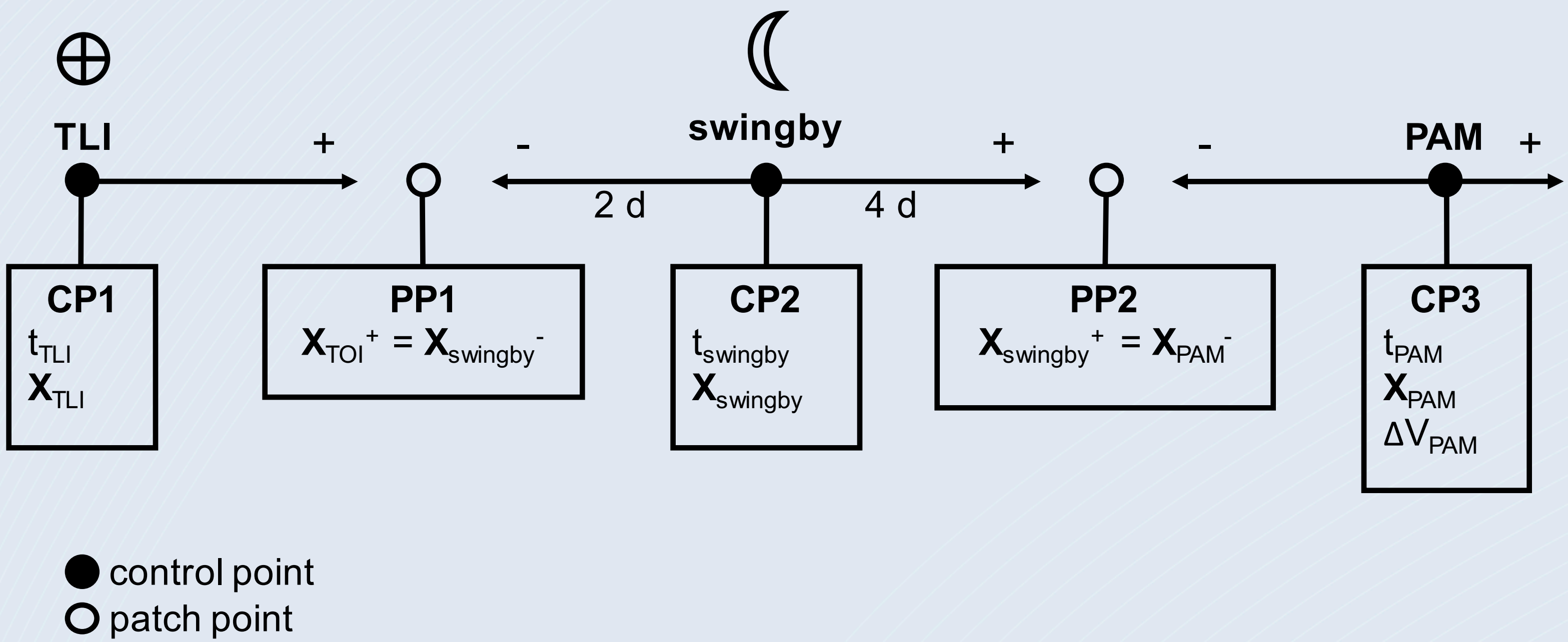


- Stage 2: Backwards design from converged mission orbit to launch vehicle separation (adding phasing loops)
  - *Single-shooting process*
  - *Starts with converged outbound solution + 2-body phasing loops guess*



Outbound Sequence Overview

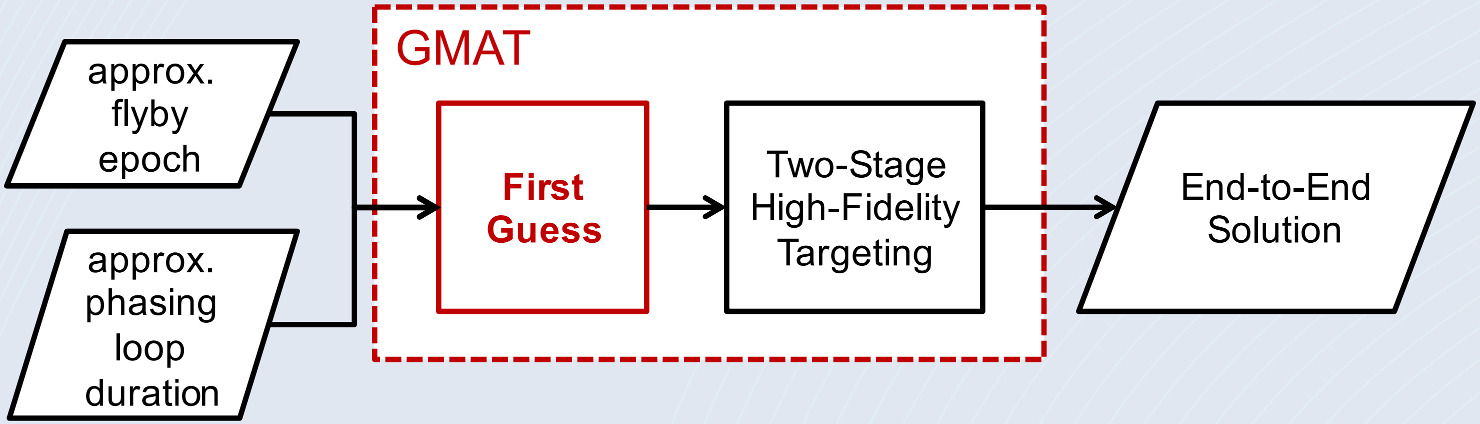
- Multiple-shooting approach w/ 5 segments



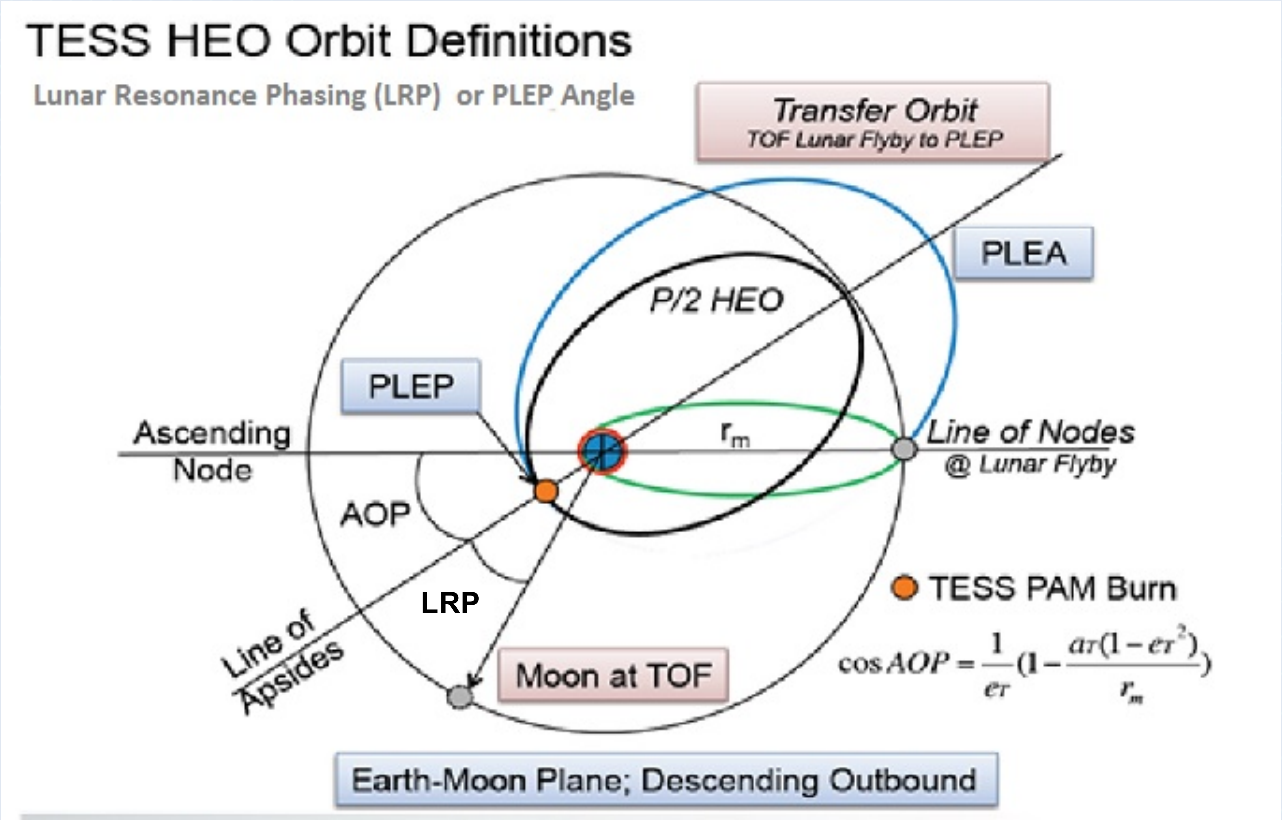
- Start with patched-conic initial guess for each segment
- GMAT targeting sequence used to find smooth solution from segmented initial guess

Implementation Overview

- General Mission Analysis Tool (GMAT) used for implementation of design
  - *GSFC's in-house high-fidelity trajectory design software*
- Uses first guess to seed numerical targeting algorithm



- The TESS trajectory has two critical features:
  - *Transfer orbit (result of lunar flyby)*
  - *2:1 lunar resonant mission orbit*



Tisserand Criterion

- The Tisserand criterion holds that a quantity T is constant before and after a flyby:

$$T = \frac{1}{2a} + cos(i)\sqrt{a(1-e^2)}$$

- Here $\boldsymbol{a}$ is semimajor axis (scaled by distance between the primary bodies), $\boldsymbol{e}$ is eccentricity and $\boldsymbol{i}$ is inclination to the orbit plane of the primaries
- The Tisserand criterion is used for TESS to design the lunar flyby.
  - *We choose the value of $\boldsymbol{T}$ to obtain the desired orbit properties of the transfer orbit after flyby to mission orbit.*
  - *The transfer orbit shape is driven by a timing condition: the need for the spacecraft at Post Lunar Encounter Perigee (PLEP) to nearly line up with the Moon. The spacecraft-Earth-Moon angle at perigee is called PLEP misalignment or the Lunar Resonant Phase Angle.*
  - *We then use the value of $\boldsymbol{T}$ to infer the shape of the orbit before flyby*

Kozai Mechanism

- The Kozai Mechanism describes the long-term evolution of a highly eccentric, highly inclined orbit due to a third body (Moon).
- The Kozai model implies that:
  - *Orbit semimajor axis is conserved*
  - *Kozai parameter* $\boldsymbol{K = cos(i)\sqrt{1 - e^2}}$ *is constant, where* $\boldsymbol{e}$ *is eccentricity and* $\boldsymbol{i}$ *is inclination to the Moon orbit plane*
- Kozai mechanism predicts
  - *Eccentricity and inclination oscillate in unison, with a period of about 8 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, if the initial inclination is higher than critical inclination 39.2 deg*

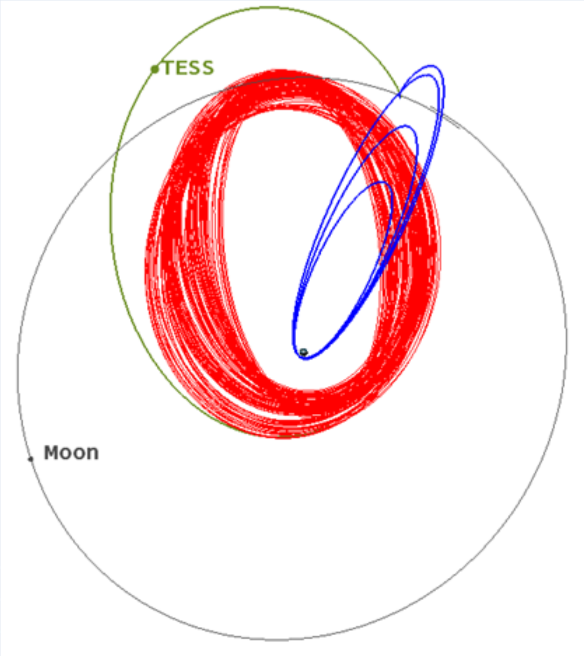
Kozai Mechanism (cont'd)

- 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 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

Methods described by Aerospace Corp in CSR and flight dynamics paper "A High Earth, Lunar Resonant Orbit For Lower Cost Space Science Missions" by Gangestad, Henning, Persinger and Ricker (AAS 13-810)

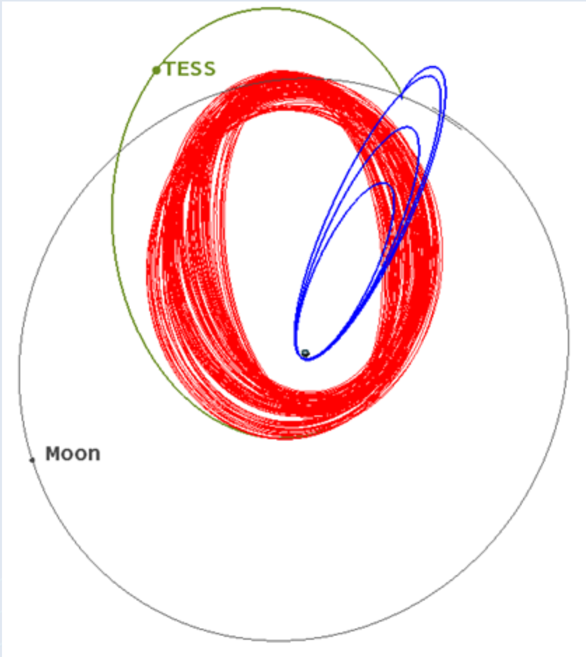
1^{st} Guess 2- and 3-Body Approximations

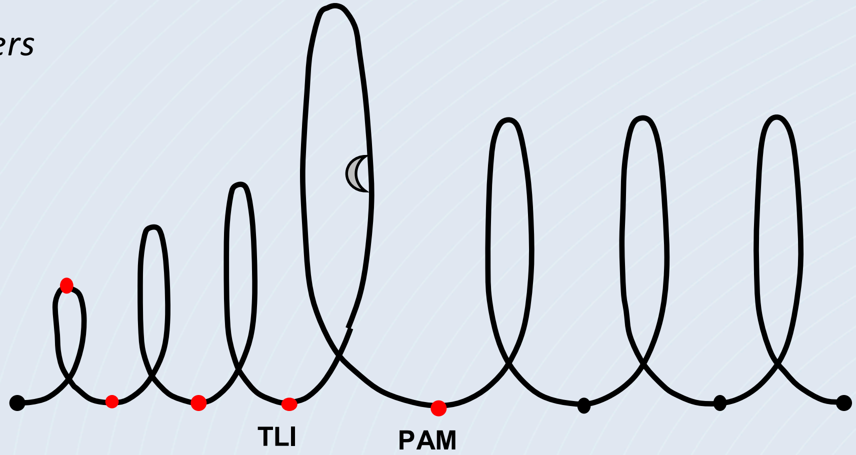
- Start with approximate flyby epoch
  - *Fixes RAAN and AOP of pre- and post-flyby arcs*
  - *Fixes Moon distance at flyby*
- Mission orbit
  - *2:1 lunar resonance → SMA = 38 Re (NOTE: the mission does not require exact resonance)*
  - *Set PLEP = 17 Re →* $e = 0.55$
  - *Choose K = 0.64 →* $i \cong 39°$ *w.r.t. Moon orbit*
- Transfer orbit (post-flyby)
  - *Match mission orbit* r_p*, orbit plane, line of apsides*
  - *Choose Tisserand value = 1.15 →* $r_a \cong 1.3\times$ *flyby Moon radius*
  - *Choose inbound/outbound flyby → TA at flyby*
  - *Argument of latitude 0 (asc.) or 180 (desc.) → argument of perigee*
  - *Ascending/descending choice & inclination w.r.t. Moon orbit → J2000 inclination*
- Pre-flyby:
  - *Ascending/descending choice & J2000 inc. at TLI → inclination w.r.t Moon orbit*
  - *Tisserand value & Rp at TLI →* $r_a \cong 1.03\times$ *flyby Moon radius*
  - *Choose inbound/outbound flyby → TA at flyby*
  - *Argument of latitude 0 (asc.) or 180 (desc.) → argument of perigee*



1^{st} Guess 2- and 3-Body Approximations

- Flyby
  - *Pre- and post-flyby velocity directions → bend angle + orbit plane*
  - *Bend angle → eccentricity*
  - *v_∞ (at Moon's Sphere of Influence) → SMA*
- Phasing loops
  - *Guess a total phasing loop duration*
  - *J2000 inclination = 28.5° typically*
  - *LV separation altitude = 200 km*
  - *P1, P2, P3 altitude = 600 km*
  - *Same orbit plane and line of apsides as pre-flyby orbit*
  - *Pre-flyby radius from Tisserand criterion*
  - *A3 radius = pre-flyby radius*
  - *Phasing loop duration guess → A2 radius*
  - *Apogee radii A1, A2, A3, A4 → P1, P2, P3 maneuvers*
- Connection to launch site
  - *Separation AOP → coast duration (AOP - 90°)*
  - *RA at coast injection = RA at KSC*





Lunar Flyby Orbit Geometry Options

- Data shows generally best results for:
  - *Pre-flyby inbound*
  - *Post-flyby descending*
  - *Post-flyby outbound*
- Pre-flyby ascending / descending can be selected
- For operational simplicity, we currently use ascending case only.
- Implies short-coast solution at Earth departure

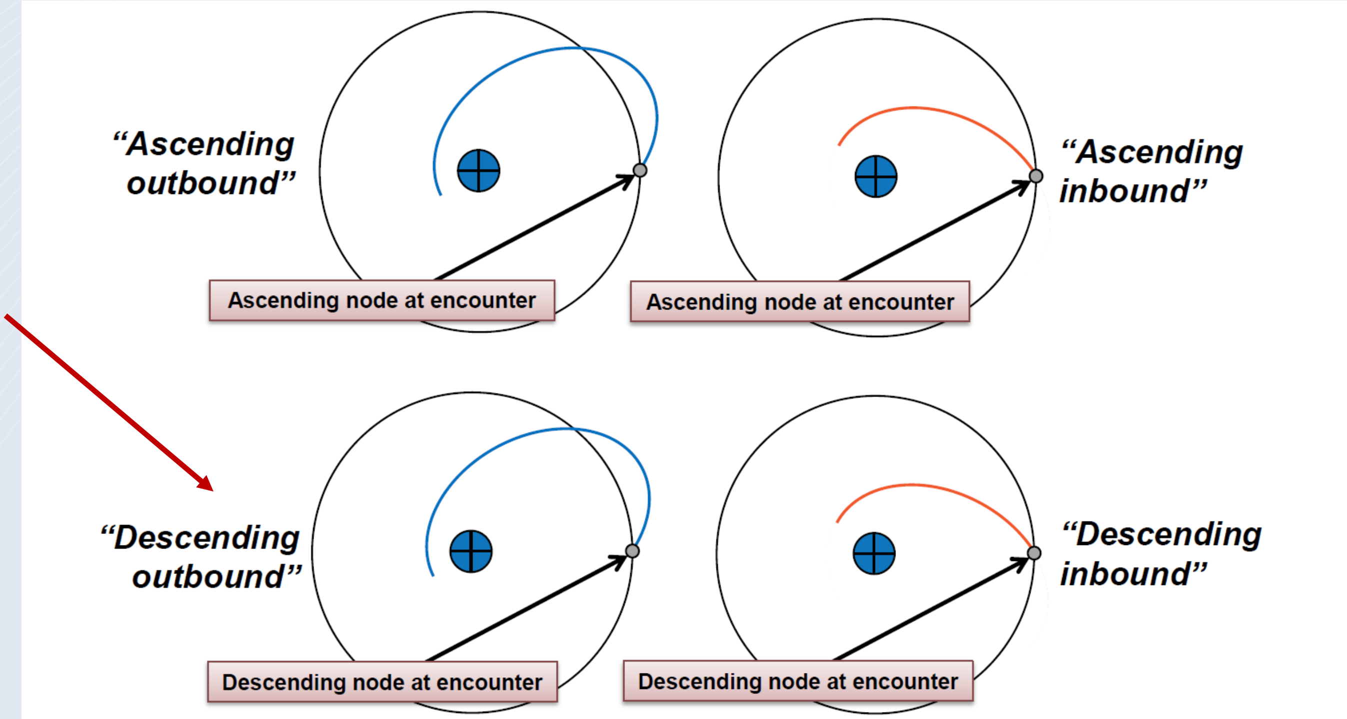
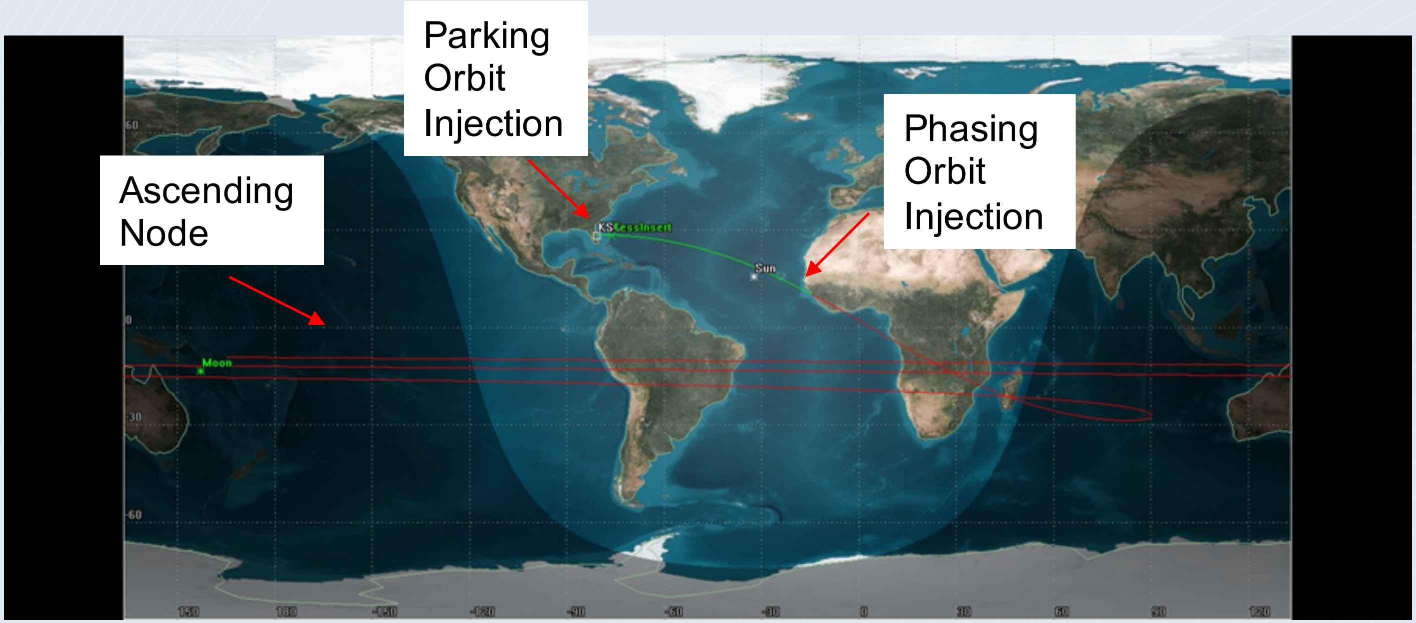


Figure 6: The four possible paths of the spacecraft following the lunar flyby.

From Gangestad, J. et al. "A High Lunar Resonant Orbit for Lower Cost Space Science Missions, AAS 13-810

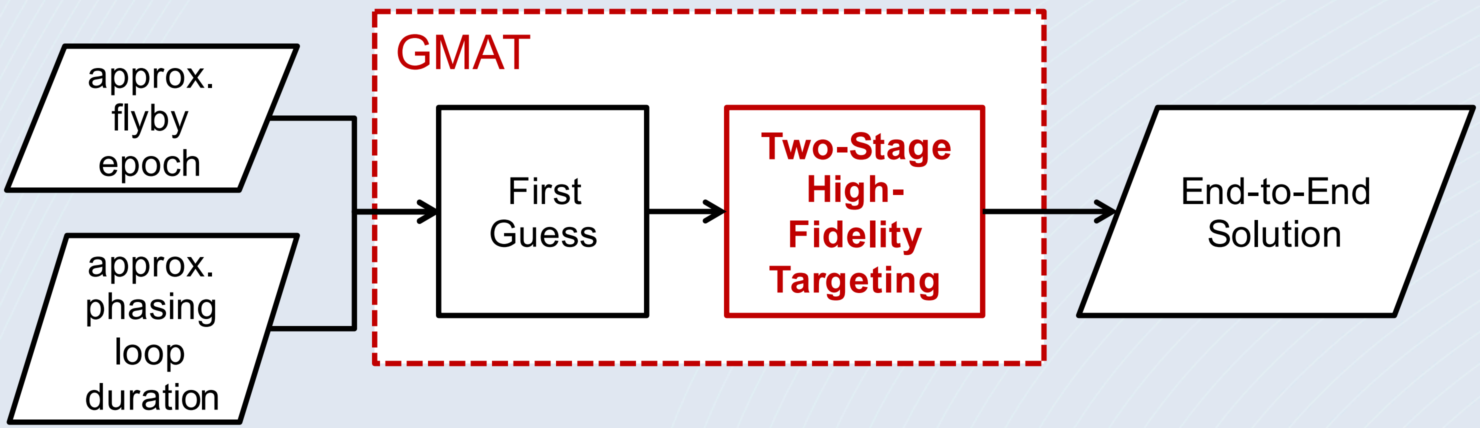
Parking Orbit

- Two constraints connect our separation state back to launch:
  - *Approx. parking orbit duration (AOP - 90°)*
  - *RA at parking orbit injection (matches RA of KSC)*



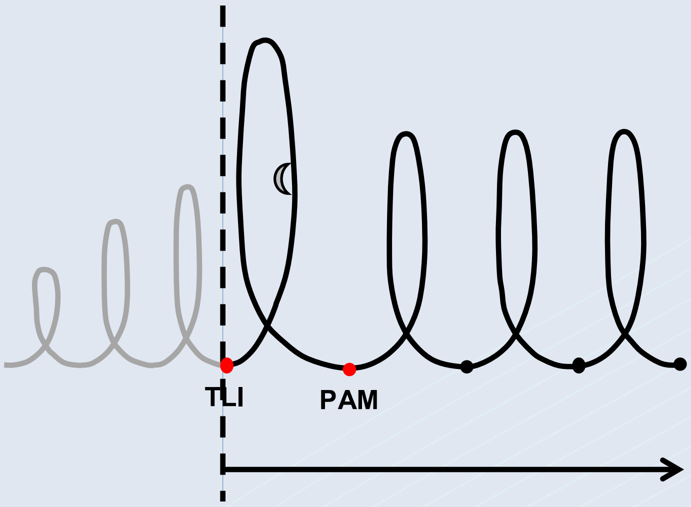
Implementation Overview

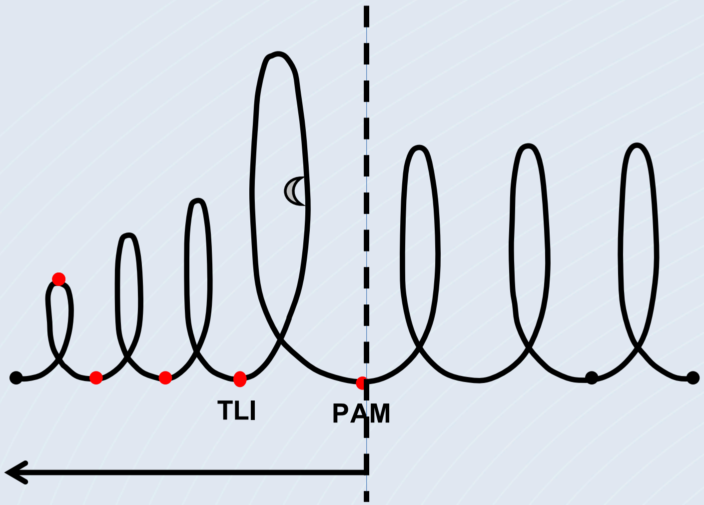
- General Mission Analysis Tool (GMAT) used for implementation of design
  - *GSFC's in-house high-fidelity trajectory design software*
- Uses first guess to seed numerical targeting algorithm



GMAT Design Approach

- Two targeting sequences
- Stage 1: Design from Translunar Injection (TLI) through flyby to Science Orbit
  - *Multiple-shooting process*
- Stage 2: Backwards design from converged mission orbit to launch vehicle separation (adding phasing loops)
  - *Single-shooting process*
  - *Starts with converged outbound solution + 2-body phasing loops guess*
- Both stages use VF13 NLP solver as robust targeter
  - *Seeks feasible solution only; not optimizing*
- Final 3rd stage: forward-propagation from SEP to check constraints





Modeling Assumptions

- All analyses share common force models, spacecraft parameters, solar system models, to the extent practical.

Spacecraft model	
Mass*	201.9 kg
Coeff. of reflectivity (SRP)	1.5
SRP area	3.5 m^2

*Low dry mass estimate, used to model worst-case SRP effect & kept for continuity. Current mass estimate is used in finite burn analysis.

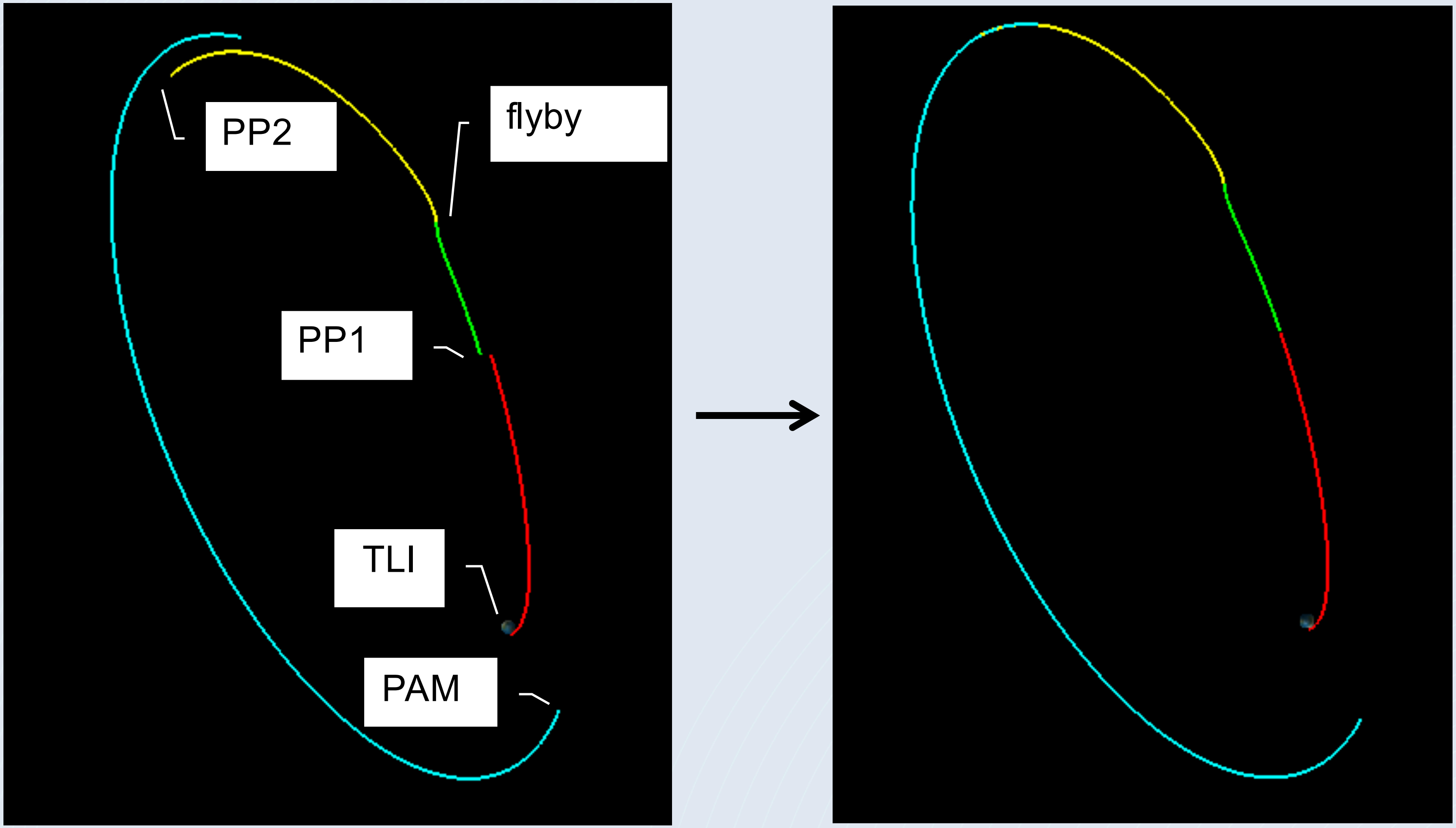
Force modeling	Phasing loops	Flyby	Mission orbit
Central-body gravity	JGM-2 40×40	Moon point mass	JGM-2 8×8
Third-body gravity	Sun, Moon	Sun, Earth	Sun, Moon
SRP	Enabled	Enabled	Enabled
Drag	Disabled	Disabled	Disabled

Solar system ephem
DE421

Stage 1: Outbound Sequence Constraints

Parameter	Value	Description
TLI inclination	28.5°	Fixes TLI at approximate LV insertion inclination
TLI perigee altitude	600 km	Phasing loop perigee altitude
TLI R·V	0	Fixes TLI at perigee
Mission orbit perigee radius	17 RE	Design value for min/max perigee behavior
PAM R·V	0	Fixes PAM at perigee
Mission orbit LRP angle	≤ 36°	Maximum misalignment from resonant condition
Mission orbit energy	2:1 resonance	Energy from SMA consistent with 2:1 resonant condition
Mission orbit Kozai parameter	0.60 ≤ K ≤ 0.80	Controls long-term perigee behavior
Mission orbit ecliptic AOP	≥ 30°	Controls maximum eclipse behavior
Position/velocity continuity	-	Position/velocity continuity between all segments

Outbound Sequence Overview



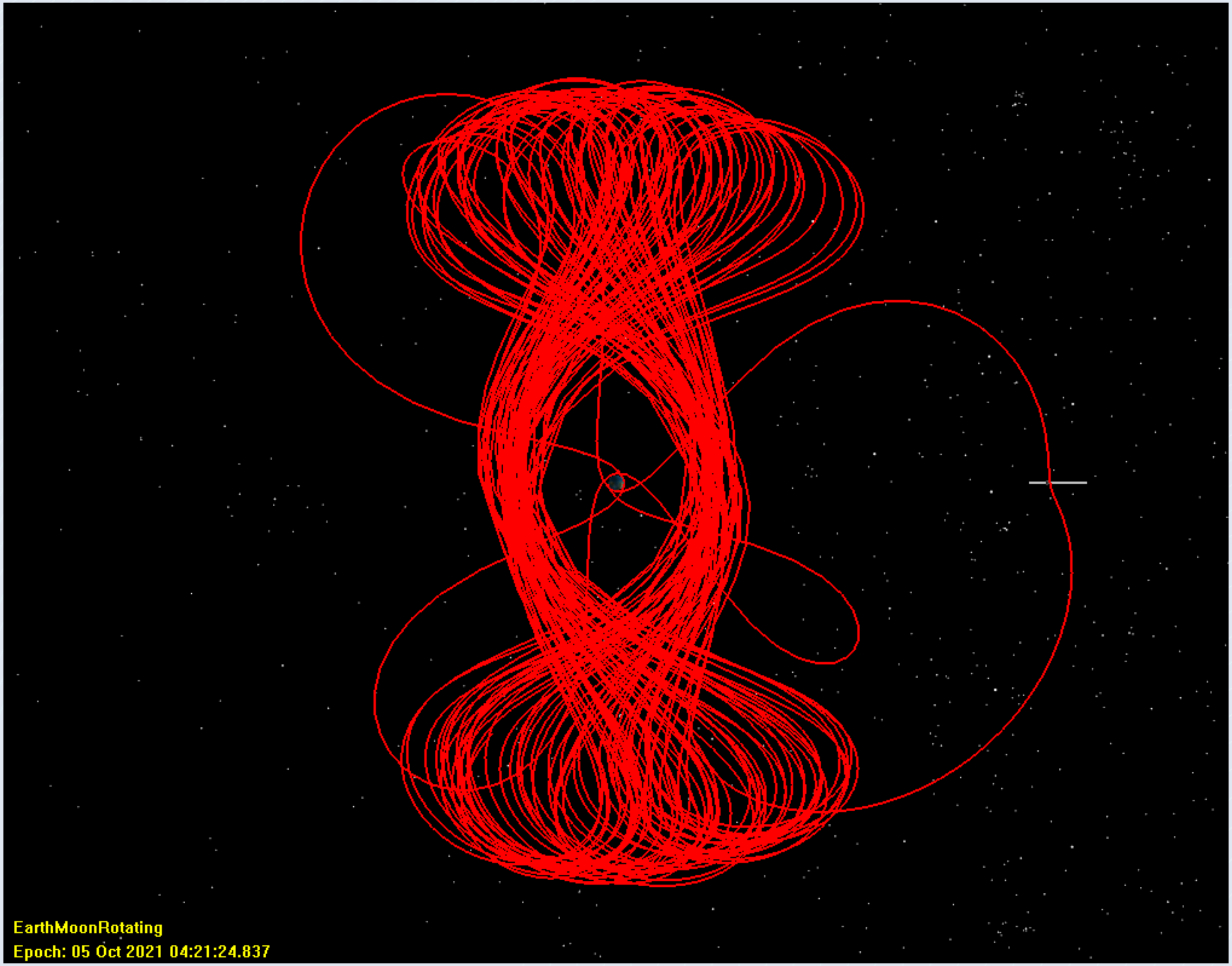
Phasing Loops Sequence Overview

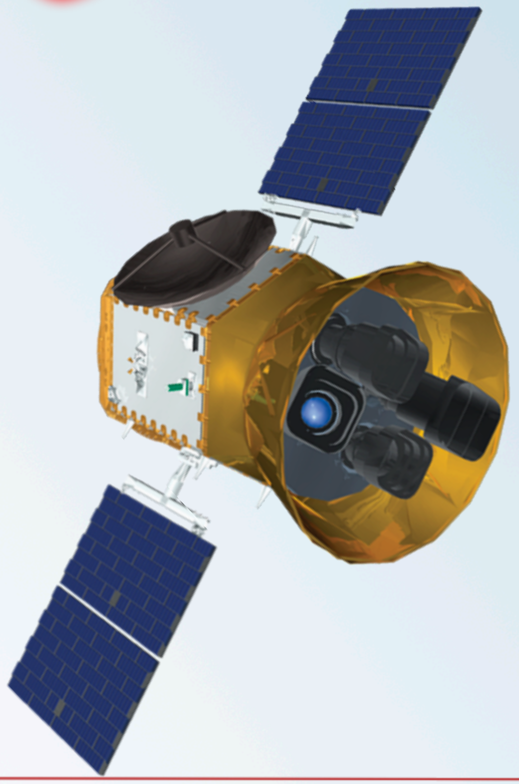
- Starts with converged outbound solution
- Back-propagates from PAM through flyby to TLI
- Uses targeting sequence to add on phasing loops
  - *Two-body initial guess for A1–A3, P1–P3 burns*
- Insertion constraint is now enforced at insertion, not at TLI
  - *Small out-of-plane components are added to PAM to correct inclination at TLI*
  - *This is a side effect of the two-stage approach; would go away in an end-to-end solution*

Stage 2: Phasing Loops Constraints

Parameter	Value	Description
P1–P3 altitude	≥600 km	Phasing loop perigee altitude
A3 radius	≤ pre-flyby radius	
A2 radius	A1 ≤ A2 ≤ A3	
A1 radius	275,000 km	A1 design radius
Separation altitude	200 km	LV requirement
Separation inclination	28.5° TOD	LV requirement
Separation epoch	match launch modeling & desired phasing loop duration	Analytical model based on launch site
Launch RA	Consistent w/ KSC launch	

Final Converged Solution



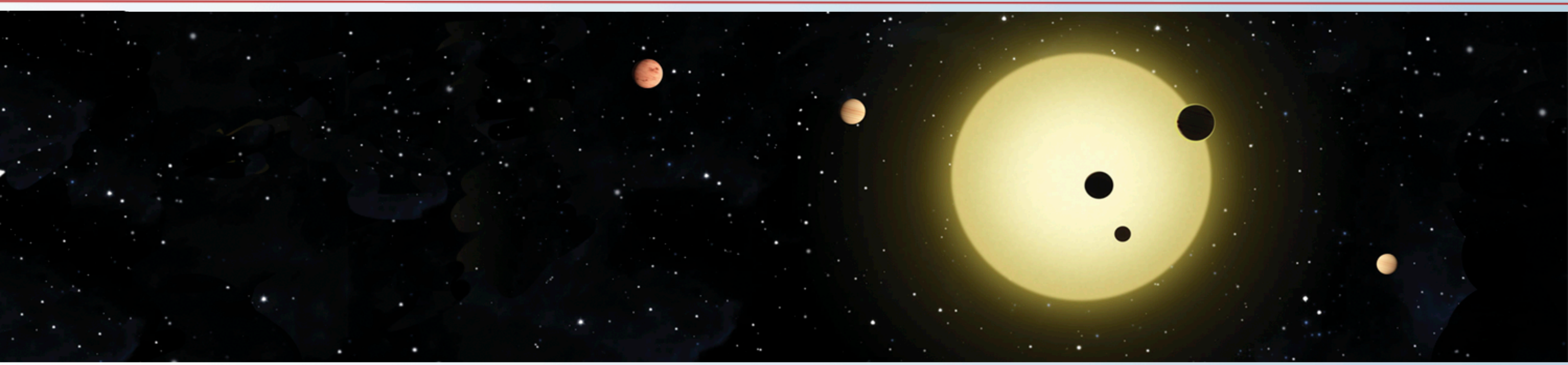


04: Solution Generation Process

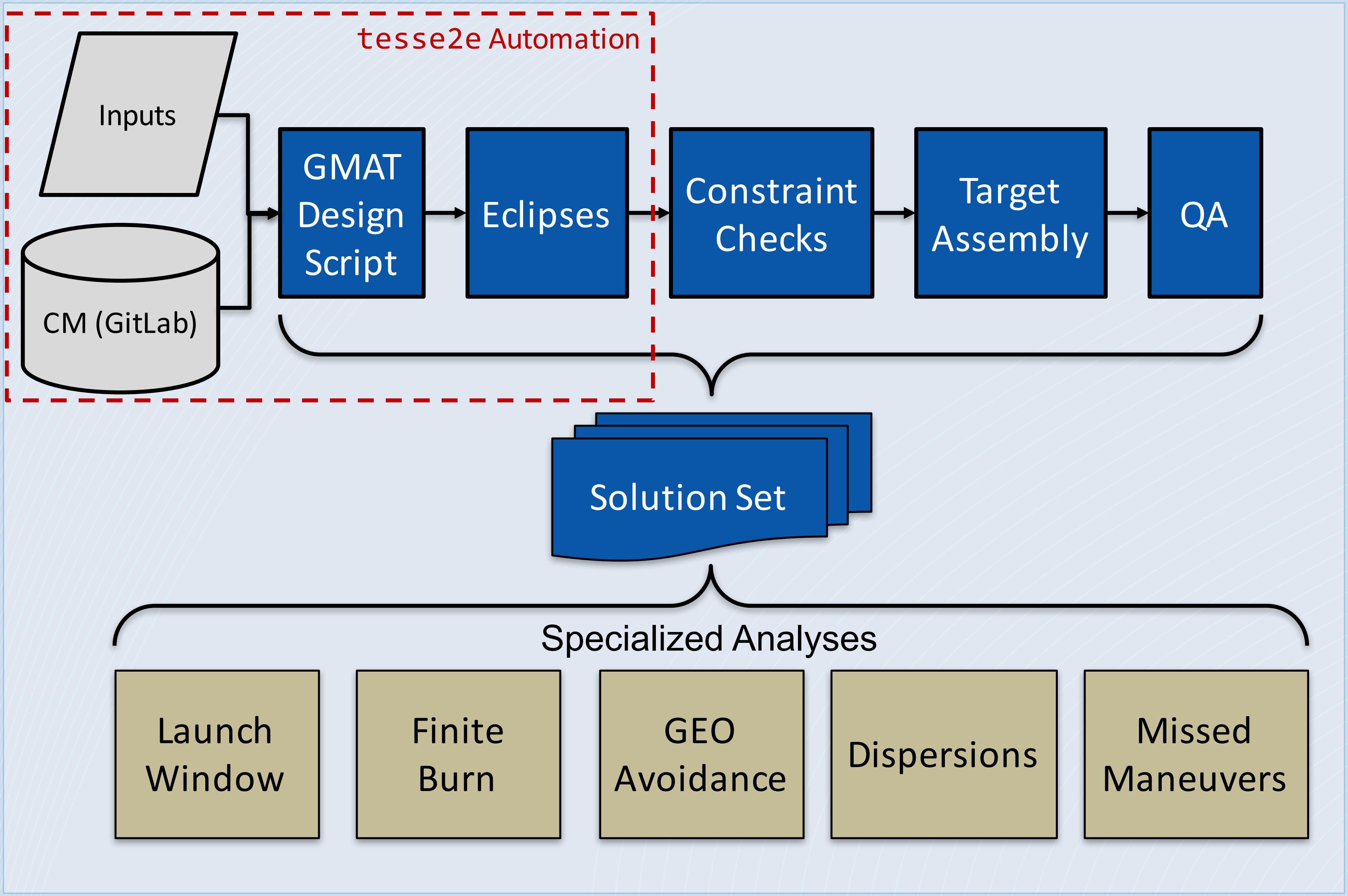
TESS Mission Design Pre-CDR Peer Review

Joel Parker

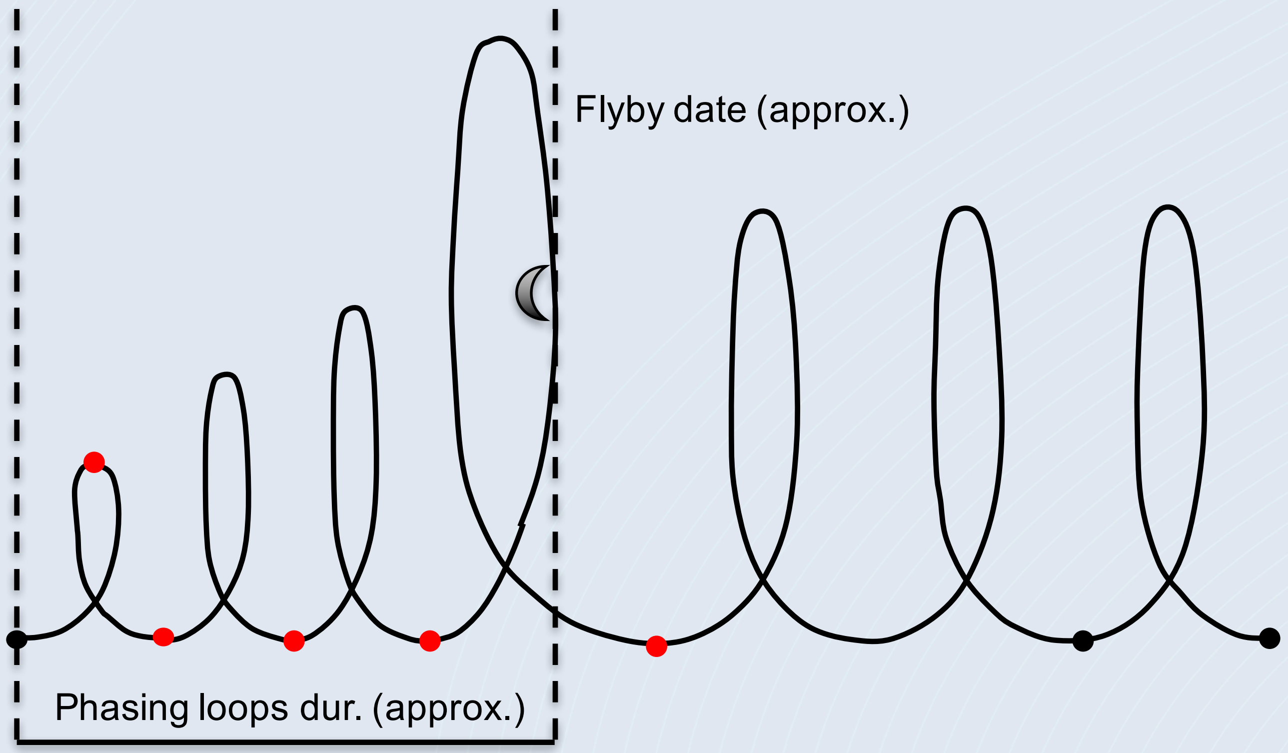
March 11, 2015



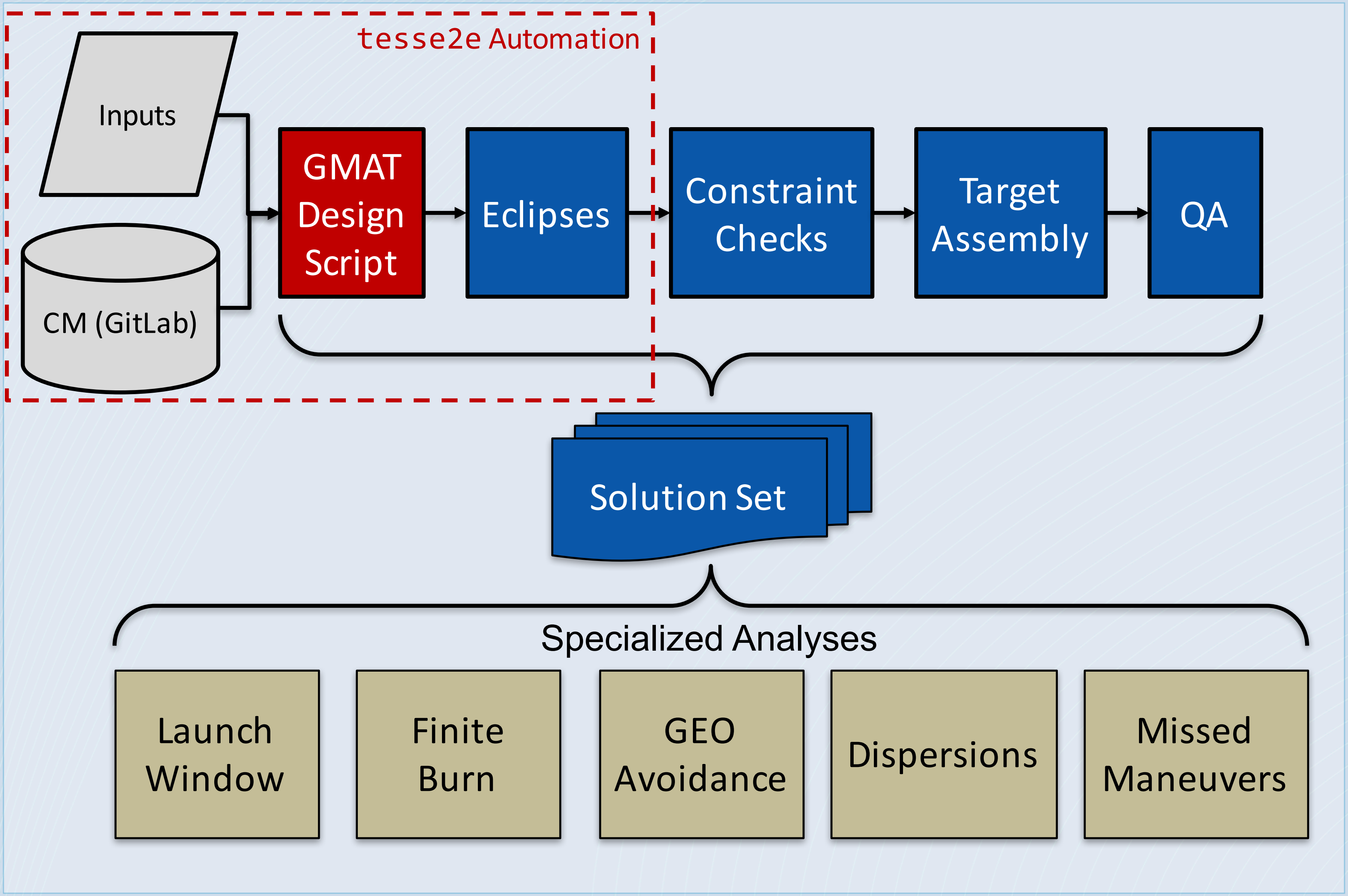
Trajectory Design Process Overview



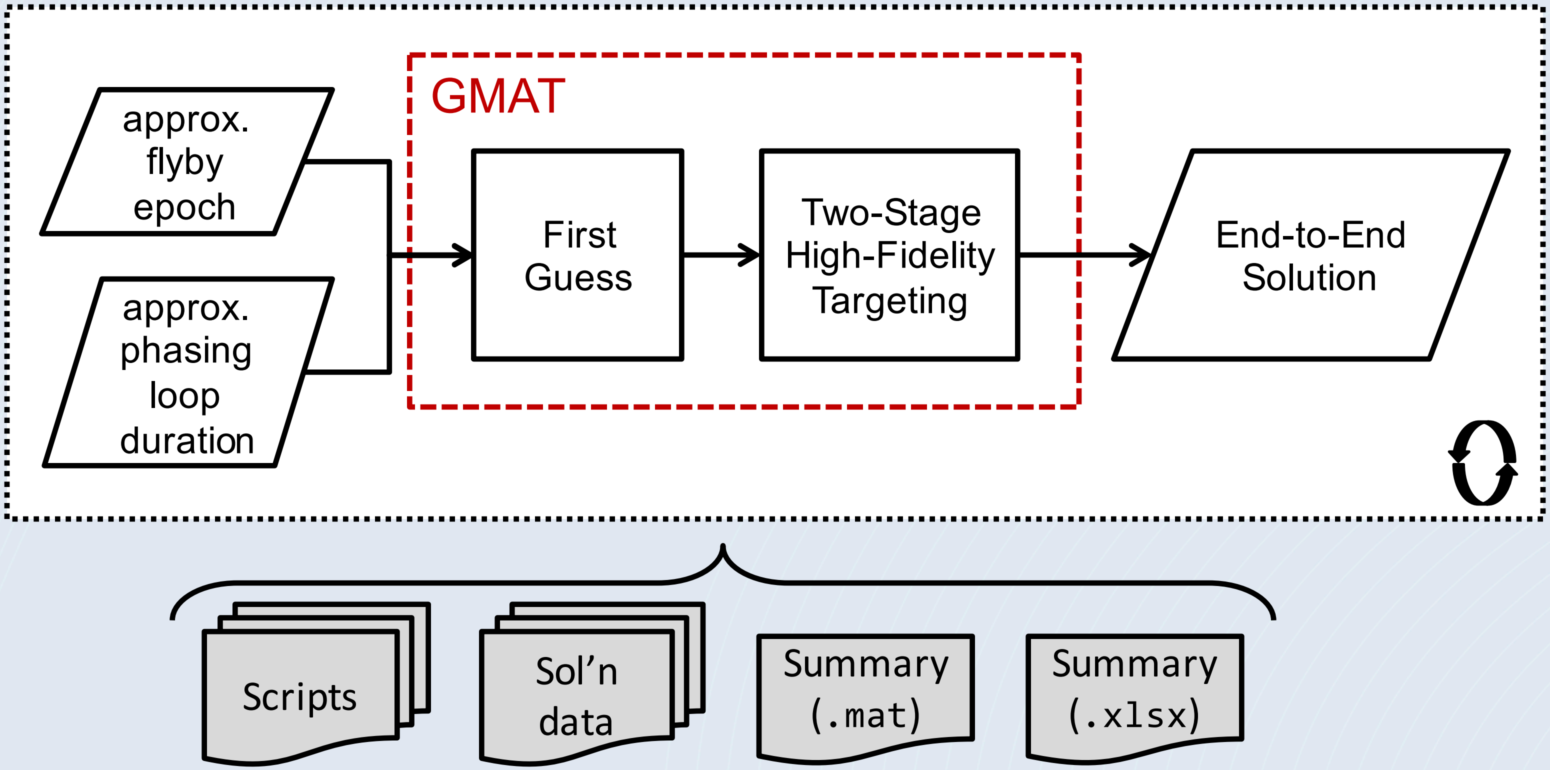
- Solutions are parameterized by two variables:
  - *Approximate flyby epochs*
    - *(# of lunar cycles, # days per lunar cycle, # epochs per day)*
  - *Approximate phasing loop duration*



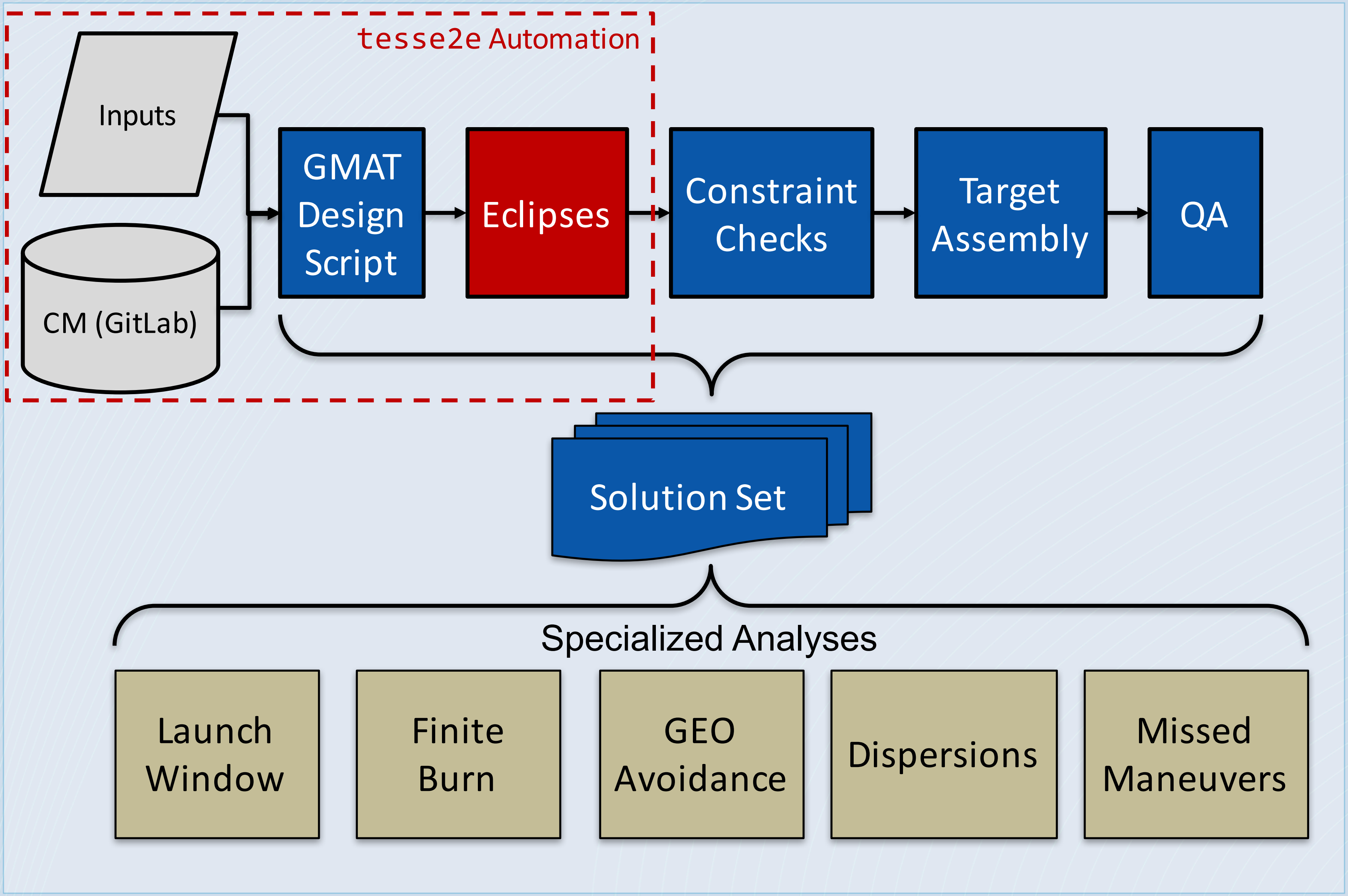
Trajectory Design Process Overview



- Template script implements trajectory design process.
- For each flyby epoch/phasing loop duration pair:
  - `tesse2e` *driver fills current values*
  - *Runs GMAT to generate converged solution*
  - *Stores output for next step*

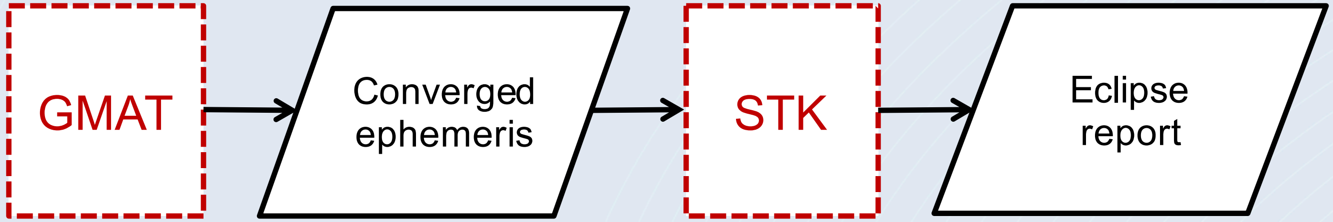


Trajectory Design Process Overview

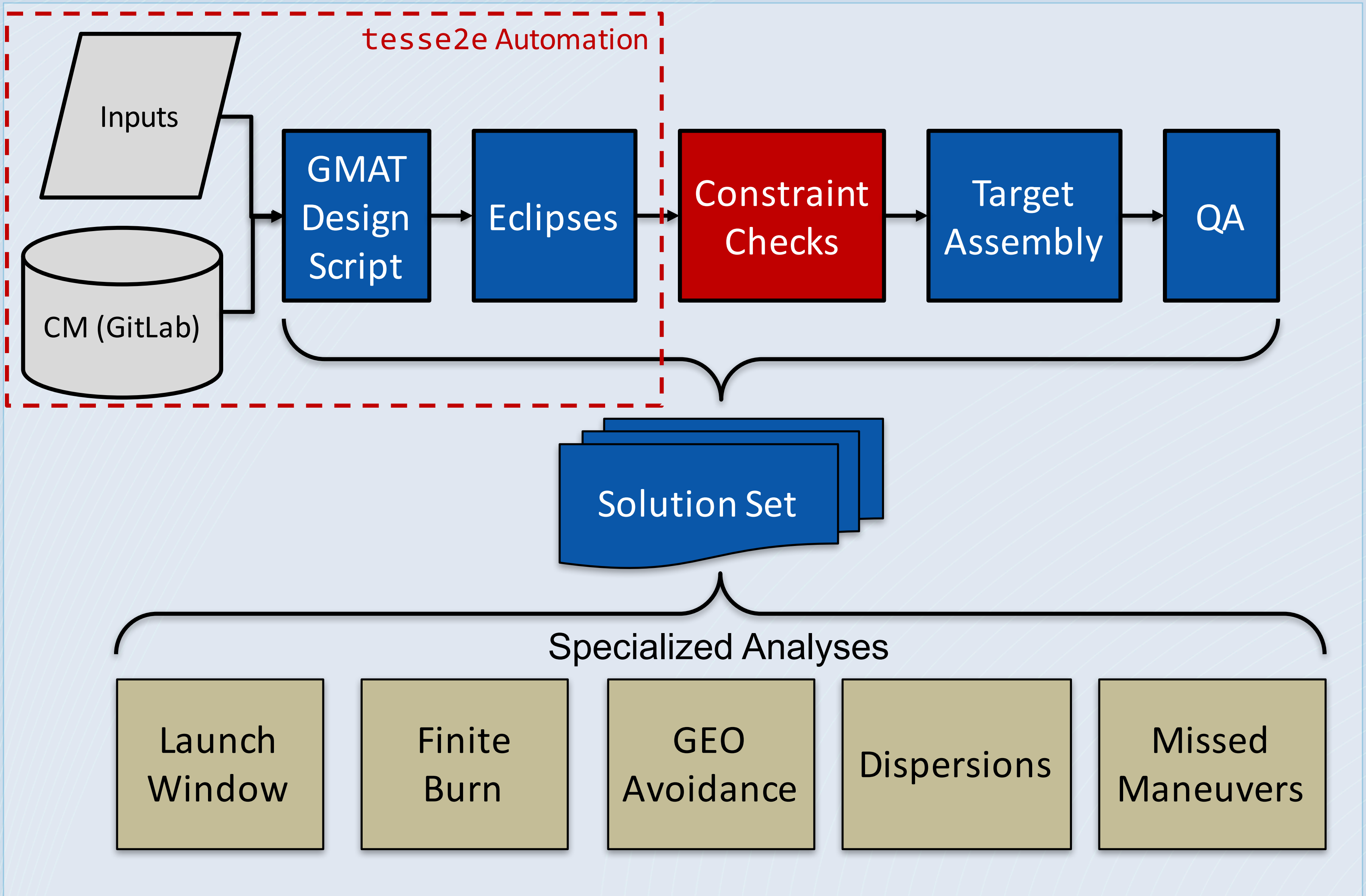


Eclipses

- STK/COM used to post-process GMAT-produced CCSDS ephemeris
  - *Until native GMAT eclipse detection is available*
- Simple Earth/Moon eclipse search

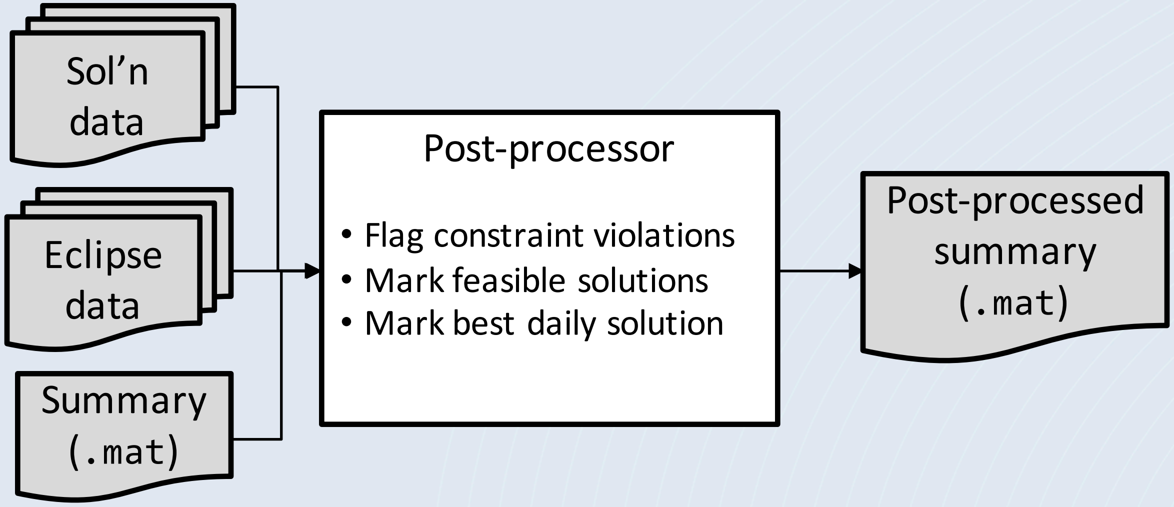


Trajectory Design Process Overview

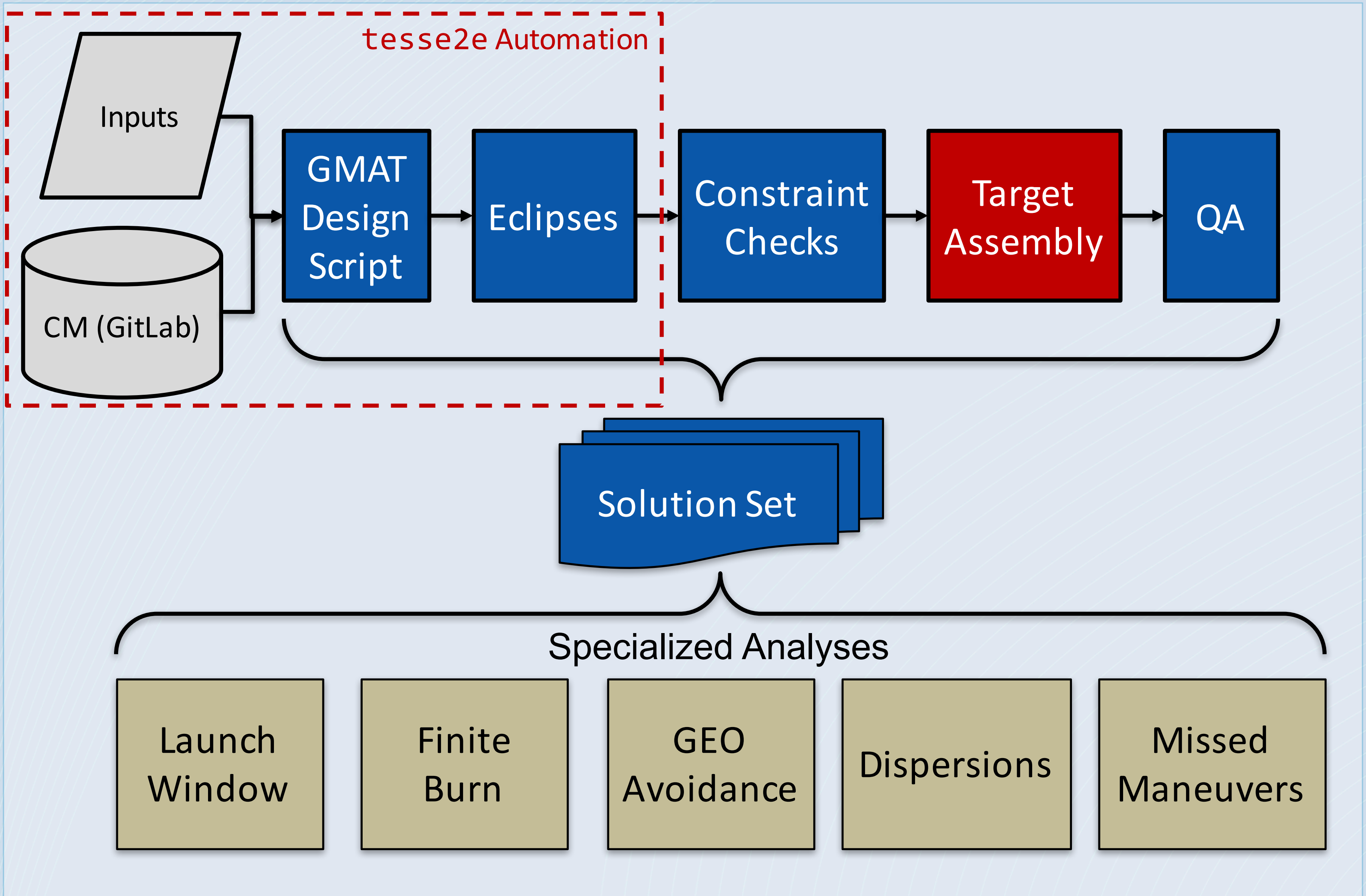


Constraint Checks

- MATLAB-based post-processing code
- Collects all available data
- Checks against remaining constraints
  - *Minimum perigee*
  - *Eclipses*
  - *FOV sun angle (during PAM)*
- Marks feasible solutions
- Marks best daily solution
  - *Currently “best” = feasible w/ lowest ΔV*

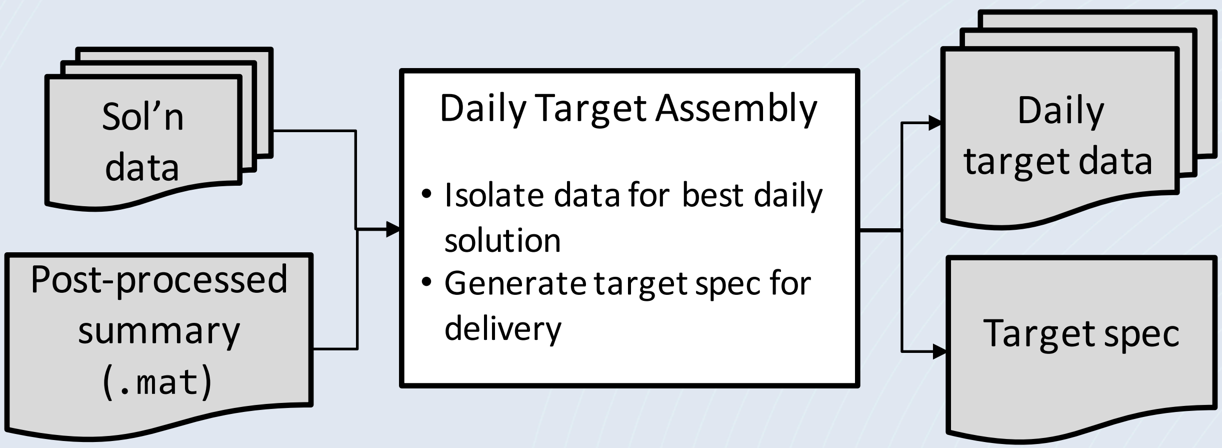


Trajectory Design Process Overview



Daily Target Assembly

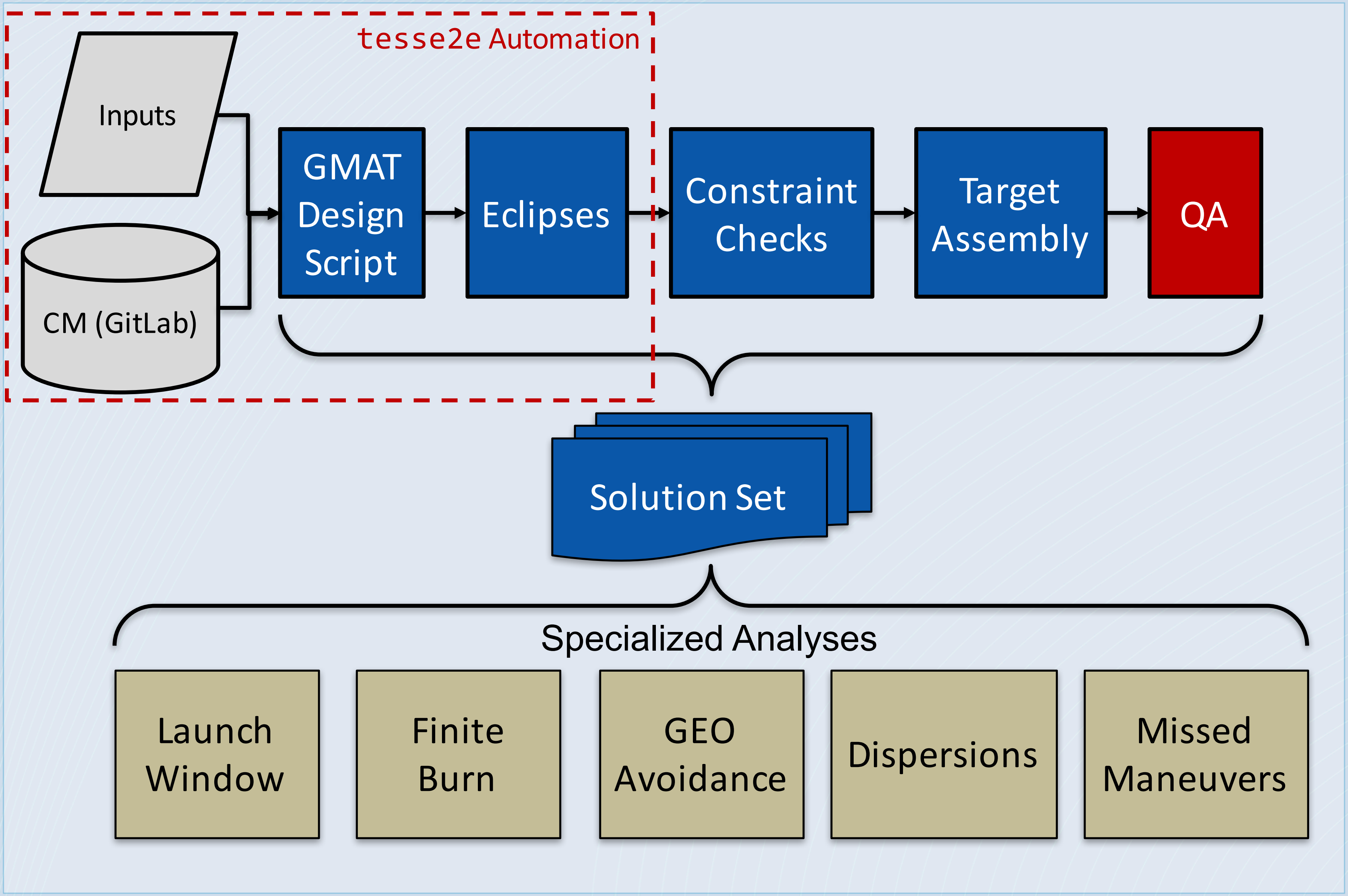
- Purpose:
  - *Isolate solution data related to best daily solutions*
  - *Process data to generate target spec for LV delivery*
- Targets specified at SECO-2
  - *Currently modeled as perigee separation state*
- Flyby B-plane parameters provided as well, as reference



Sample Target Spec

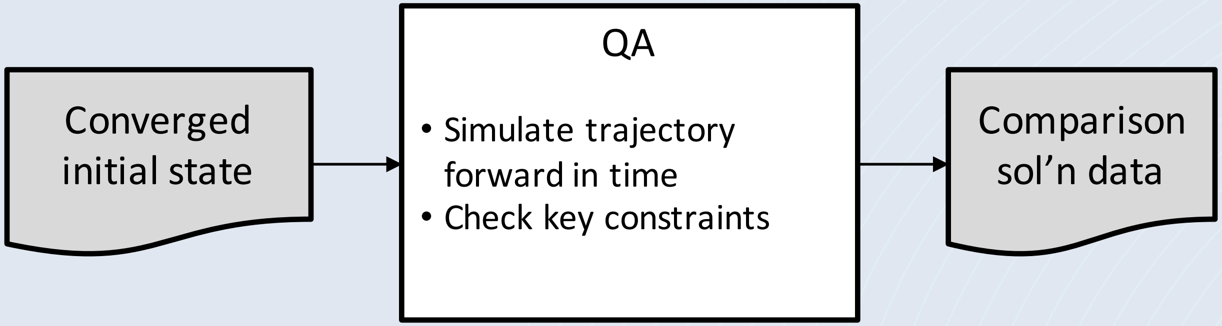
```
================================================================================
Launch date: 10 Aug 2017


TOD Keplerian elements at SECO-2:

Epoch (TAI)     = 10 Aug 2017 14:36:52.083
RadPer          = 6578.137720724262
RadApo          = 253436.6018892931
INC             = 28.50000147741938
RAAN (EME2000) = 2.077884859568627
AOP             = 176.7206400417074
TA              = 1.478779333471098e-06


Moon-centered inertial parameters at flyby:

Epoch (TAI)   = 07 Sep 2017 05:36:47.813
B-vector mag = 16941.87565965261
BdotR         = -12952.64867919611
BdotT         = 10920.44152314588
C3            = 0.6857082349237472


================================================================================
...
```

Trajectory Design Process Overview



QA

- Purpose:
  - *Independently simulate converged trajectory forward in time*
  - *Check key constraints and provide achieved values*
    - *Launch trajectory*
    - *Flyby B-plane parameters*
    - *Mission orbit LRP angle, energy, ecliptic AOP*
    - *Velocity-Sun angle at PAM*
    - *Minimum perigee (25 years)*



QA Results

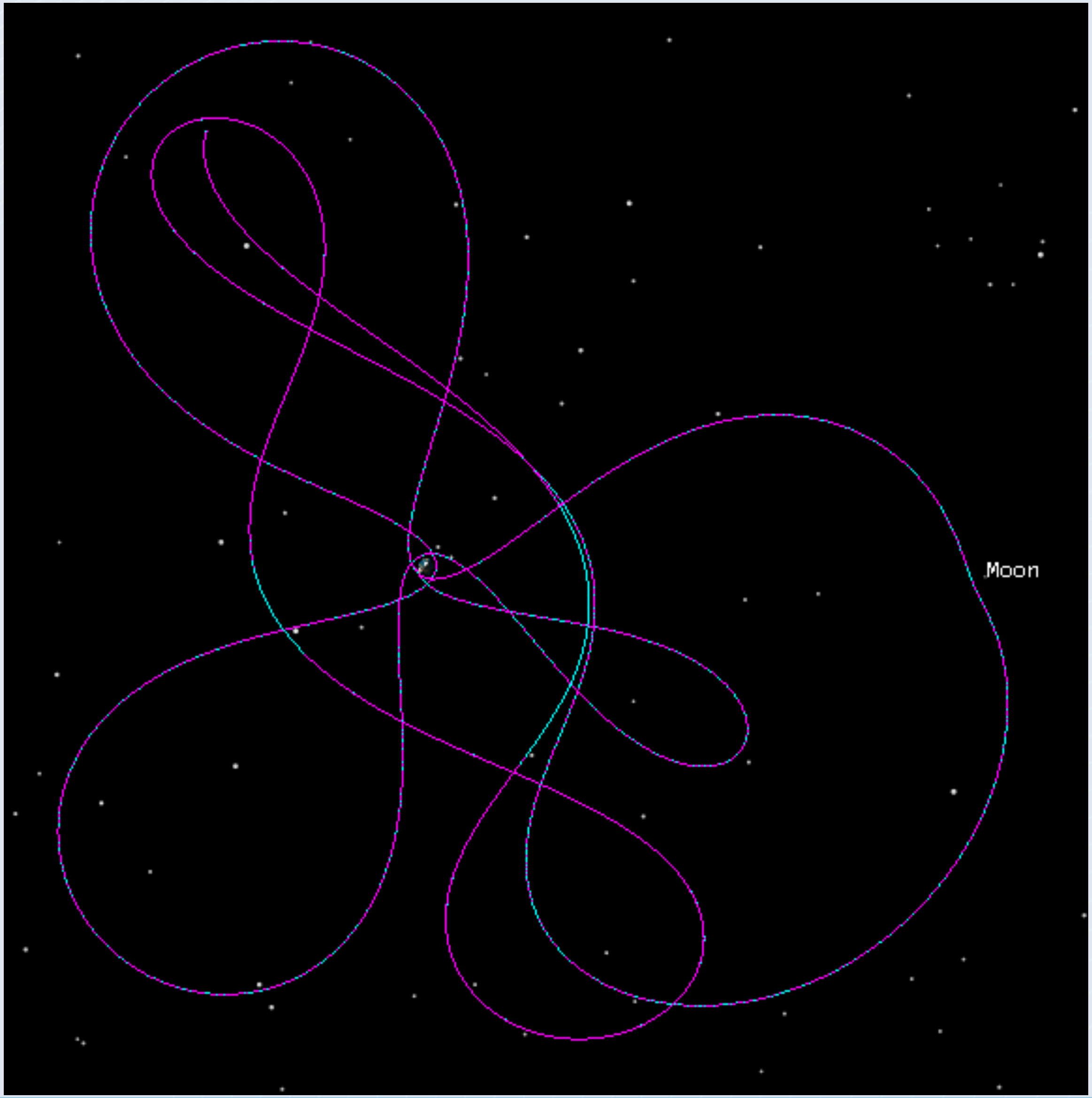
- Summary of matching between converged solution and independent QA

Value	Average absolute difference
Flyby periapsis epoch (s)	8e-5
Flyby B-vector mag. (km)	3e-5
Flyby B·R (km)	5e-5
Flyby B·T (km)	5e-5
Velocity-Sun angle at PAM (deg)	7e-2
LRP angle at PLEP (deg)	2e-1
Min. perigee violations (25 years)	No change

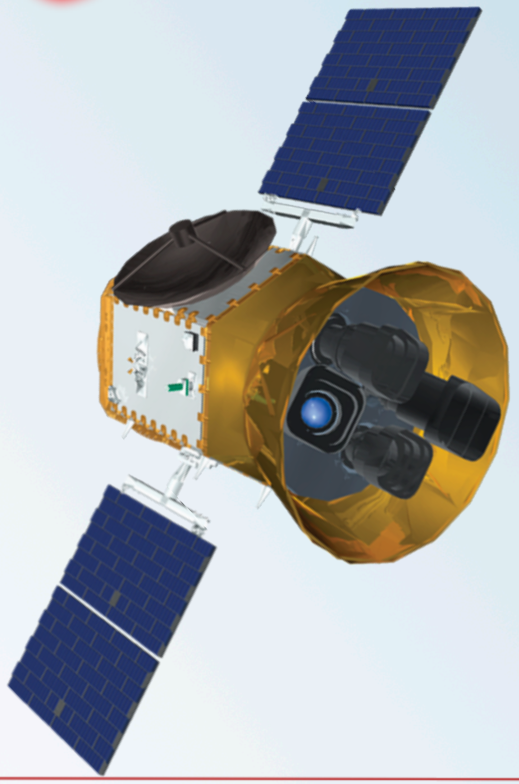
- Post-flyby parameters are sensitive to changes in flyby
- Results confirm that trajectory being designed & independently resimulated are equivalent

QA Results

Overlaid solution & QA ephemerides (phasing loops + 1 mission orbit)



Transiting Exoplanet Survey Satellite



05: Finite Burn Modeling

Donald Dichmann

March 11, 2015

Finite Burn Modeling

- Leostar-2/750 bus from Orbital Sciences
- Monopropellant hydrazine, blow-down system so thrust & Isp vary over mission
- Same propellant tank used for 22-N main thruster and 5-N ACS thrusters
- 5-N thruster would be ineffective for orbit maneuvers
- Orbital has provided functions to describe thrust & Isp as a function of feed pressure.
- Orbital also provided a data table to propulsion parameters at start, middle and end of mission
- From the data table we identified a linear relationship between tank pressure and feed pressure, so we can express thrust & Isp as a function of tank pressure in GMAT

Design Process

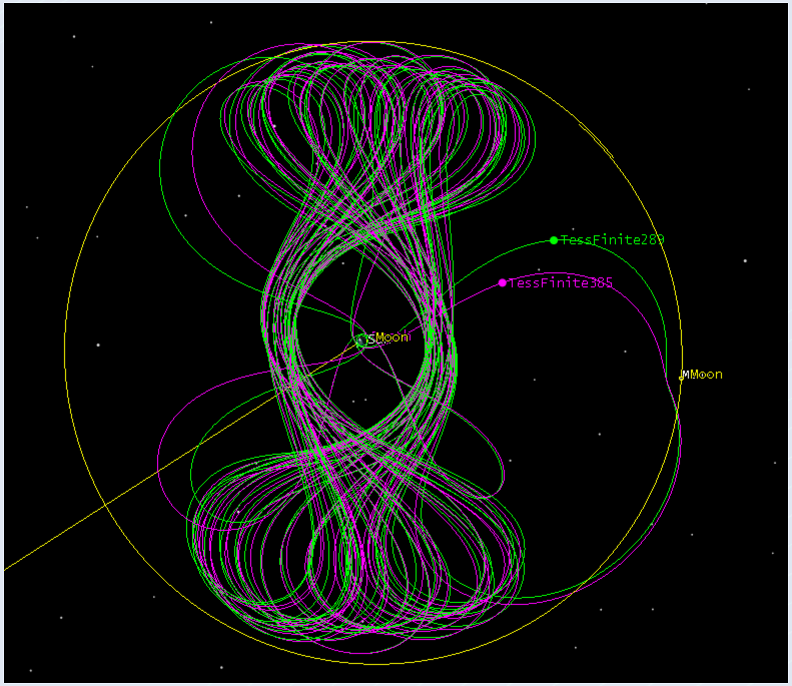
- As noted earlier, we compute the impulsive maneuvers in 2nd solver sequence, then use these in the 3rd sequence (final propagation) to determine equivalent finite burns
- At the time of each maneuver we determine the initial thrust and mass to estimate burn duration from DV:
  - *estimated duration = DV / accel, where accel = thrust / mass at start of burn*
- As a 1st guess we center the burn on the impulsive burn epoch
- For each burn we then retarget to solve for burn start epoch and burn duration:
  - *For P1 we target on AOP and the epoch of P2*
  - *For P2 we target on AOP and the epoch of P3*
  - *For P3 we use the centered burn based on impulsive maneuver*
    - *we could target on B-plane parameters of flyby*
  - *For PAM we target on AOP and mission orbit energy (equivalent to orbit period)*
- To compute finite-burn DV we use two methods
  - *DV = accel * duration, where accel = average of acceleration before burn and acceleration after burn (primary method)*
  - *DV = magnitude of difference between (velocity after burn duration with burn applied) - (velocity after burn duration with no burn applied)*

Observatory Mass

- During the design process there have been different observatory masses:
  - *Propellant mass is 45 kg*
  - *Not To Exceed (NTE) dry mass is 385 kg*
  - *On 2015/01/6, the Current Best Estimate (CBE) of dry mass was 268 kg On 2015/02/24, the CBE of dry mass was 289 kg*
    - *21 kg or 8% increase*
- In this review for finite burn analysis, we primarily use dry mass 289 kg, but we also look at the NTE mass for comparison

August 10 solutions

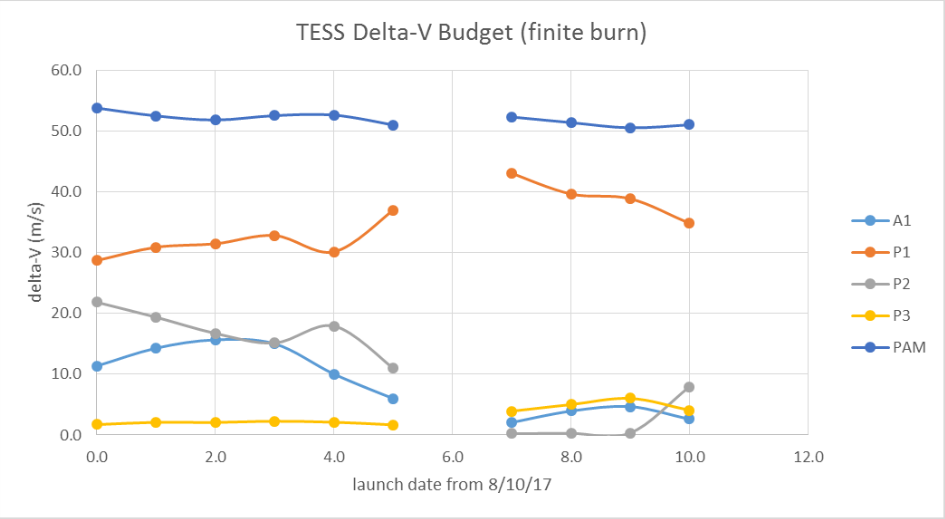
8/10/2017				
	dry mass (kg)	289.00		
	DV impulsive (m/s)	DV finite (m/s)	mass used (kg)	burn dur (sec)
A1	11.8	11.4	1.64	134.87
P1	28.4	28.7	4.09	379.61
P2	21.7	21.9	3.10	327.19
P3	1.7	1.7	0.24	27.26
PAM	53.5	53.8	7.48	928.64
total	117.1	117.5	16.55	1797.58
	dry mass (kg)	385.0		
	DV impulsive (m/s)	DV finite (m/s)	mass used (kg)	burn dur (sec)
A1	9.1	8.7	1.63	133.66
P1	17.4	17.6	3.24	295.66
P2	31.9	32.9	5.99	644.59
P3	1.4	1.4	0.25	29.85
PAM	52.1	53.5	9.58	1285.73
total	111.8	114.2	20.69	2389.51



- For different masses we get different SRP & different individual phasing loop sizes, though we design with the same phasing loop total duration
- We see only small DV penalty for finite burns:
  - 0.3% for mass 289 kg, 2% for mass 385 kg
- For different masses we get nearly the same total DV, but the burn durations and masses used are proportionally higher

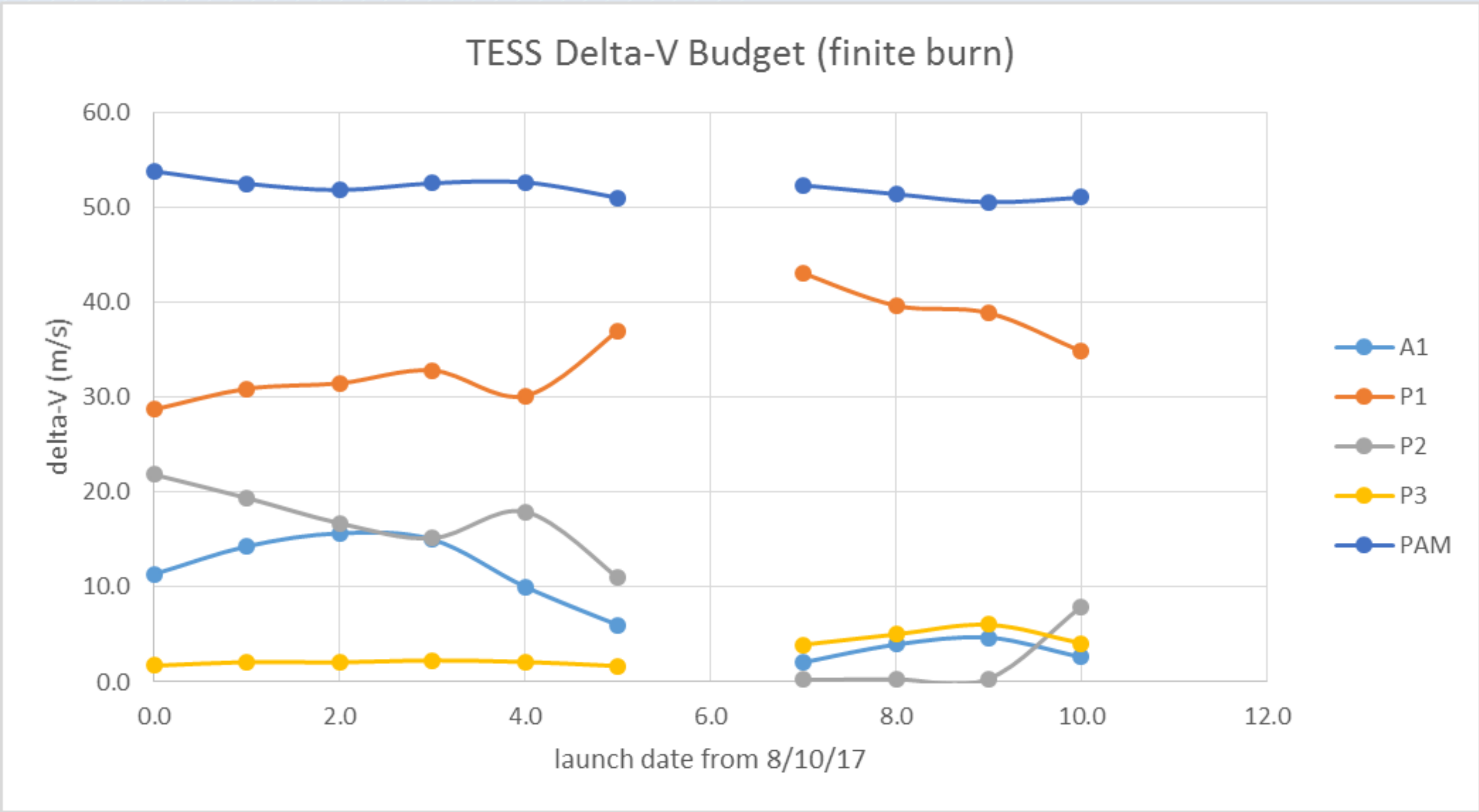
Finite Burn Deterministic DV

					Launch Date						
Maneuver (m/s)	**8/10/17**	**8/11/17**	**8/12/17**	**8/13/17**	**8/14/17**	**8/15/17**	**8/16/17**	**8/17/17**	**8/18/17**	**8/19/17**	**8/20/17**
A1	11.4	14.3	15.7	15.0	10.0	6.0		2.1	4.0	4.7	2.6
P1	28.7	30.9	31.5	32.8	30.1	37.0		43.1	39.6	38.9	34.8
P2	21.9	19.4	16.7	15.2	17.9	11.0		0.3	0.3	0.4	8.0
P3	1.7	2.1	2.1	2.3	2.1	1.7		3.9	5.0	6.0	4.1
PAM	53.8	52.5	51.8	52.5	52.6	51.0		52.3	51.4	50.5	51.0
total	117.5	119.2	117.8	117.8	112.9	106.7		101.6	100.3	100.5	100.5

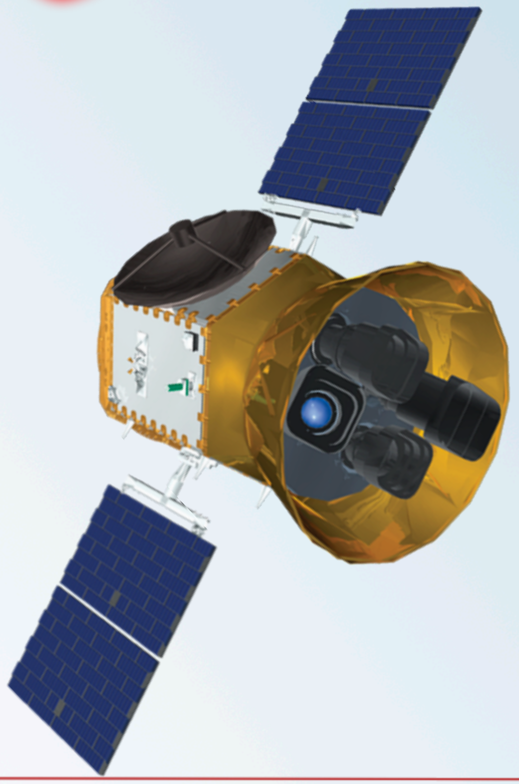


- These solutions all used phasing loop duration of 27.3 days, and dry mass of 289 kg (CBE)
- Finite Burn DV is close to impulsive values, with only a few percent difference
- Ten launch dates found in Aug 2017, exceeding requirement of five days
- Trajectory design did not converge for 8/16/17, near lunar perigee.
  - The cause is being investigated
  - Note there is a significant drop in P2 after 8/16/17. This is when the 3rd loops reaches its max allowed apogee equal to 4th loop (flyby) apogee radius

Finite Burn Deterministic DV (cont'd)



Transiting Exoplanet Survey Satellite



06: Launch Vehicle Dispersion Analysis

TESS Mission Design Pre-CDR Peer Review

Craig Nickel
March 11, 2015

Overview

- Launch Dispersion analysis is based on expected dispersions, documented in the "TESS Trajectory Analysis Input Specifications" (the "Target Spec")
- Currently we do not have a full covariance matrix for launch dispersion
  - *Full launch dispersion covariance expected in April 2015*
- This first attempt appears to have produced an algorithm that can successfully retarget for any expected launch injection error
  - *What we show here are upper bounds on the DV penalty*
- Performed a 'hypercube' analysis based on the Target Spec
- Implemented a Monte Carlo simulation, assuming standard deviations from Target Spec
  - *Focus on 3-sigma level DV bounds to meet 99% probability to meet the DV budget*

ID	Parent ID	Title	Requirement	Compliance
L3_FD_28	MRD_104	Delta-V Budget	FD shall design ascent-to-mission orbit to require no more than 215 m/s delta-V with 99% probability of success.	**Comply.** See detailed analysis.

- Ultimate goal of the algorithm and this analysis, is to demonstrate that the phasing loop design is robust enough to still achieve the required nominal mission orbit, within the DV budget required

Delta-V Budget

Event	Planned ΔV (m/s)	Current ΔV (m/s) Aug 2017
A1	20	0–17
P1	35	31–50
P2	20	0-20
P3	5	0–8
Period Adjust Maneuver (PAM)	70	56–68
Deterministic Total	**150**	**109–131**
Launch Window Allowance	10	10
Launch Vehicle Dispersion	25	25–31
Trajectory Correction Maneuvers	10	15-26
Margin	20	22-55
Total	**215**	**215**

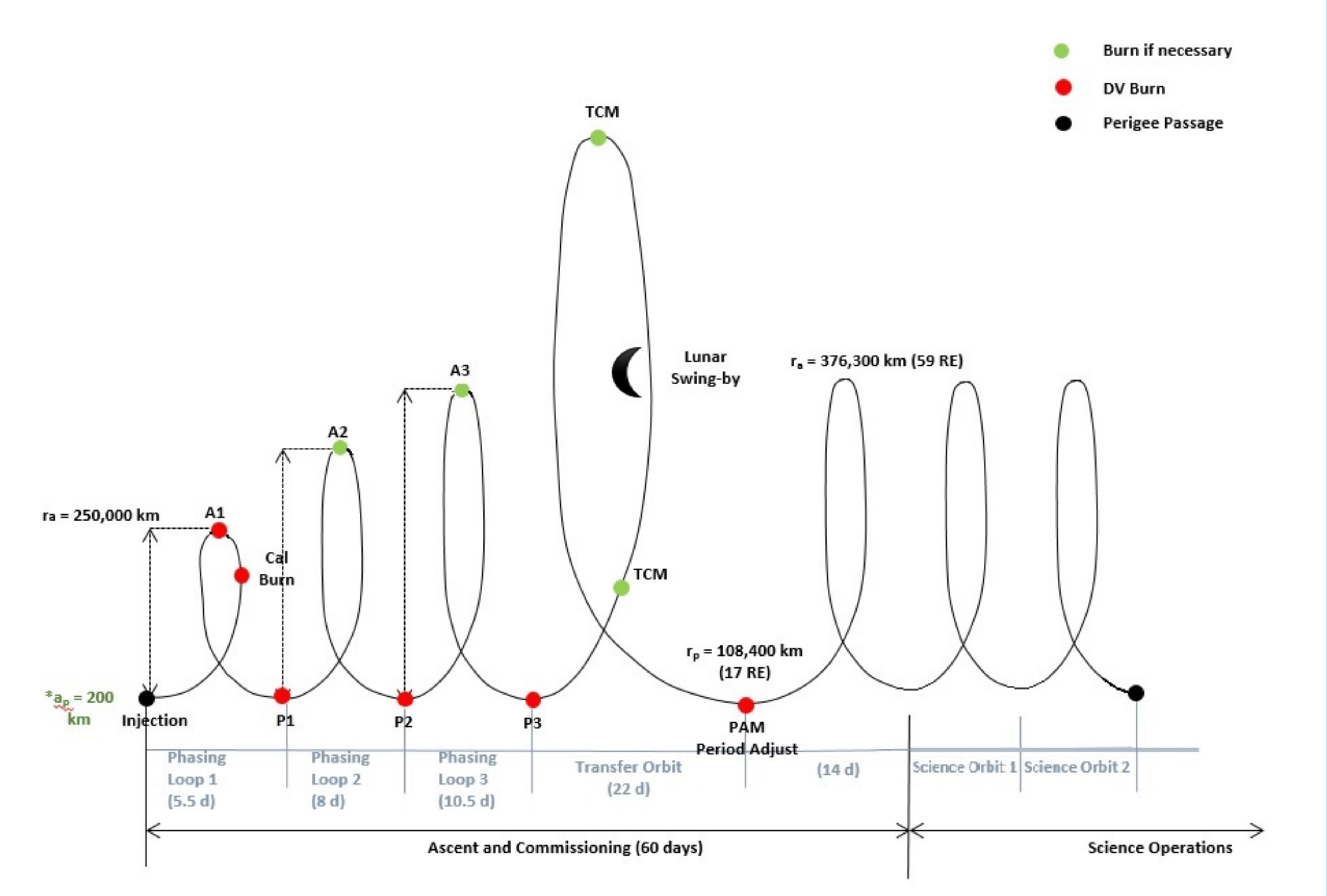
Process

- Hypercube Analysis
  - *Hypercube analysis in this instance means taking the target specification launch vehicle performance 3-sigma tolerances, and applying those dispersions to the nominal launch insertion Keplerian elements for that launch day*
  - *Apply the min and max expected error, and retarget the trajectory to determine the DV penalty*
- Apply phasing loop algorithm (details on the next slide)
  - *The goal is to still achieve the required nominal mission orbit*
  - *Consistent with the Launch Window and Missed/Partial Burn algorithms*
- Assess DV penalty for each individual dispersion
  - *Will provide upper bounds of DV required to correct*
- Simple Monte Carlo analysis also performed
  - *Assumes standard deviations from the Target Spec*
  - *100 random draws across all elements*

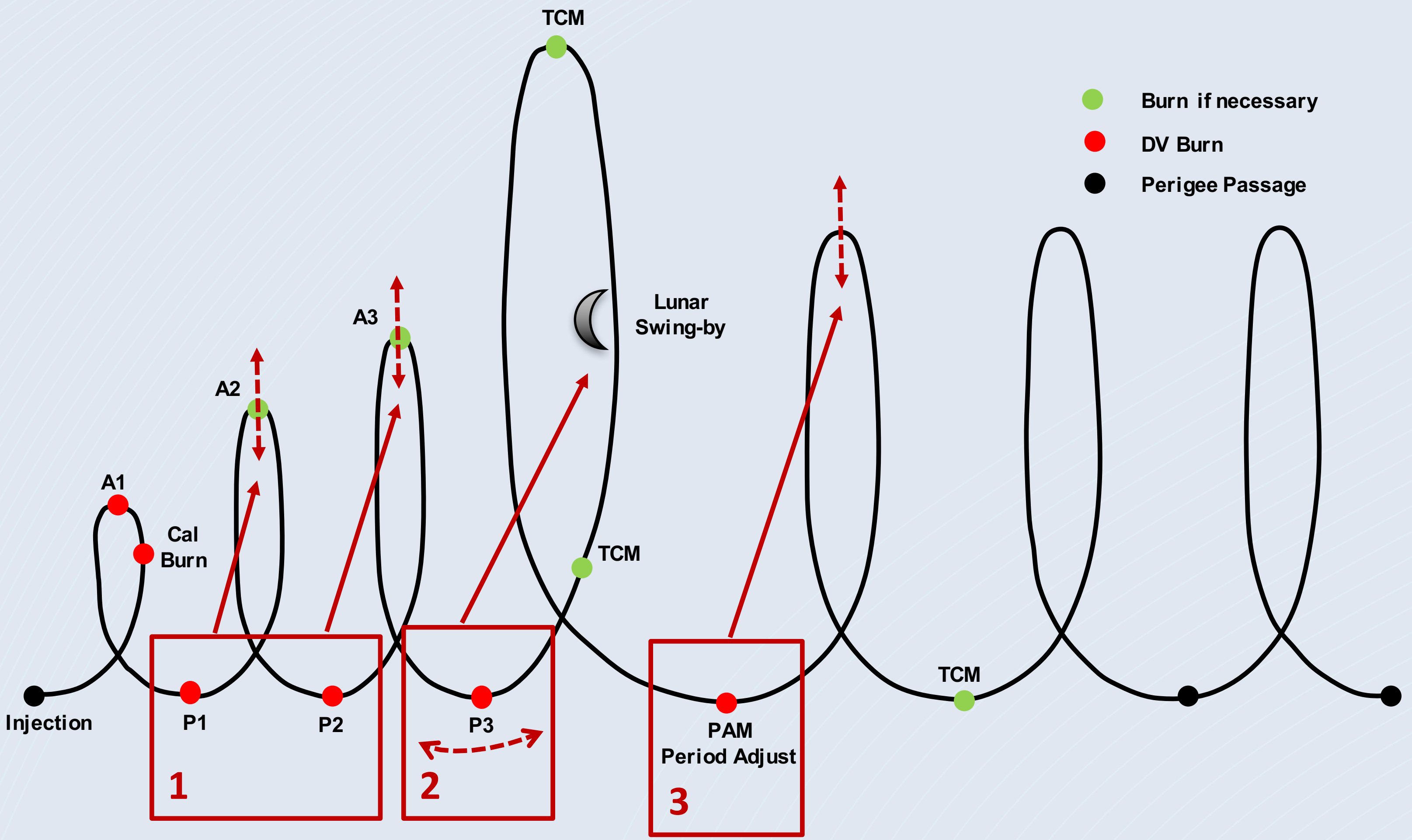
Algorithm

- If necessary, retarget the A1 maneuver so that P1 perigee is above 600 km altitude
  - *This does not appear to be necessary for the cases we modeled*
- Retarget the P1 and P2 maneuver so that P3 perigee occurs at the nominal time
  - *This allows us to return the nominal timing, to set up for Translunar Injection at P3*
  - *Replanning can change the shape of the first 3 phasing loops*
  - *Because the original trajectory was not optimized, this step can produce a reduction in DV of a few m/s, even for zero perturbation*
- Optimize the P3 maneuver (epoch and components) to achieve the nominal B-plane parameters at lunar SOI
- P1, P2 and P3 are optimized to minimize Delta-V
- Modify PAM to achieve the mission orbit energy.
  - *2-year propagation to check stability*

Nominal Phasing Loop Diagram



Current Retargeting Strategy



1. Retarget P1/P2 to correct timing at P3
2. Retarget P3 to achieve nominal B-plane (optimize)
3. Retarget PAM to achieve 2:1 resonance (optimize)

Keplerian Element Error Bounds

Hypercube Component	3-sigma Tolerance
Apogee Radius (km)	+43,000
Apogee Radius (km)	-31,000
Perigee Radius (km)	+15
Perigee Radius (km)	-15
Inclination (deg)	+0.1
Inclination (deg)	-0.1
RAAN (deg)	+0.3
RAAN (deg)	-0.3
AOP (deg)	+0.3
AOP (deg)	-0.3

- Values from current TESS Trajectory Analysis Input Specifications (ELVL-2015-0043923)

Hypercube Analysis Results

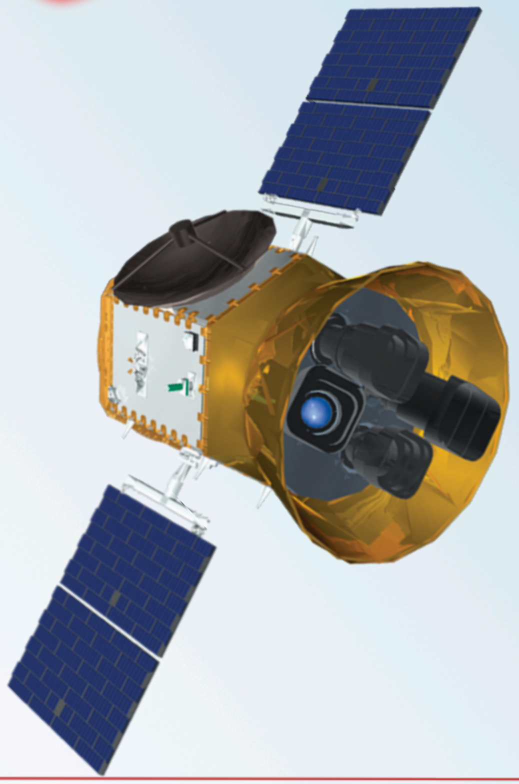
Hypercube Component	3-sigma Tolerance	Delta-V Penalty (m/s) Aug 10, 2017 Launch	Delta-V Penalty (m/s) Aug 15, 2017 Launch	Delta-V Penalty (m/s) Aug 19, 2017 Launch
Apogee Radius (km)	+43,000	-24.20	-19.47	-21.87
Apogee Radius (km)	-31,000	14.80	19.01	17.48
Perigee Radius (km)	+15	0.03	0.78	0.17
Perigee Radius (km)	-15	-0.02	-0.36	-0.16
Inclination (deg)	+0.1	-0.14	3.48	0.70
Inclination (deg)	-0.1	-0.39	4.25	0.48
RAAN (deg)	+0.3	0.68	3.43	1.27
RAAN (deg)	-0.3	3.72	2.75	4.68
AOP (deg)	+0.3	2.12	4.35	-0.17
AOP (deg)	-0.3	-2.36	4.55	3.34

- Insertion energy dispersions dominate the DV penalty
- Extreme energy errors stay within the notional 25 m/s DV budget for dispersions
- For angle perturbations we can also reduce DV cost by optimizing the phasing loop shapes

Monte Carlo Simulation

- Preliminary Monte Carlo simulation using the Target Spec values in lieu of a full launch dispersion covariance
  - *Used the lower 31,000 km apogee radius bounds*
  - *Extended hypercube analysis algorithm to handle random perturbations*
  - *The higher apogee insertion dispersion are favorable in terms of DV budget and may disproportionately skew the results positively*
- 100 random draws for these simulations
  - *Draws from all orbital elements*
  - *Bounded the random draws between +/-3 sigma*
  - *MATLAB used to make draws using a Gaussian distribution*

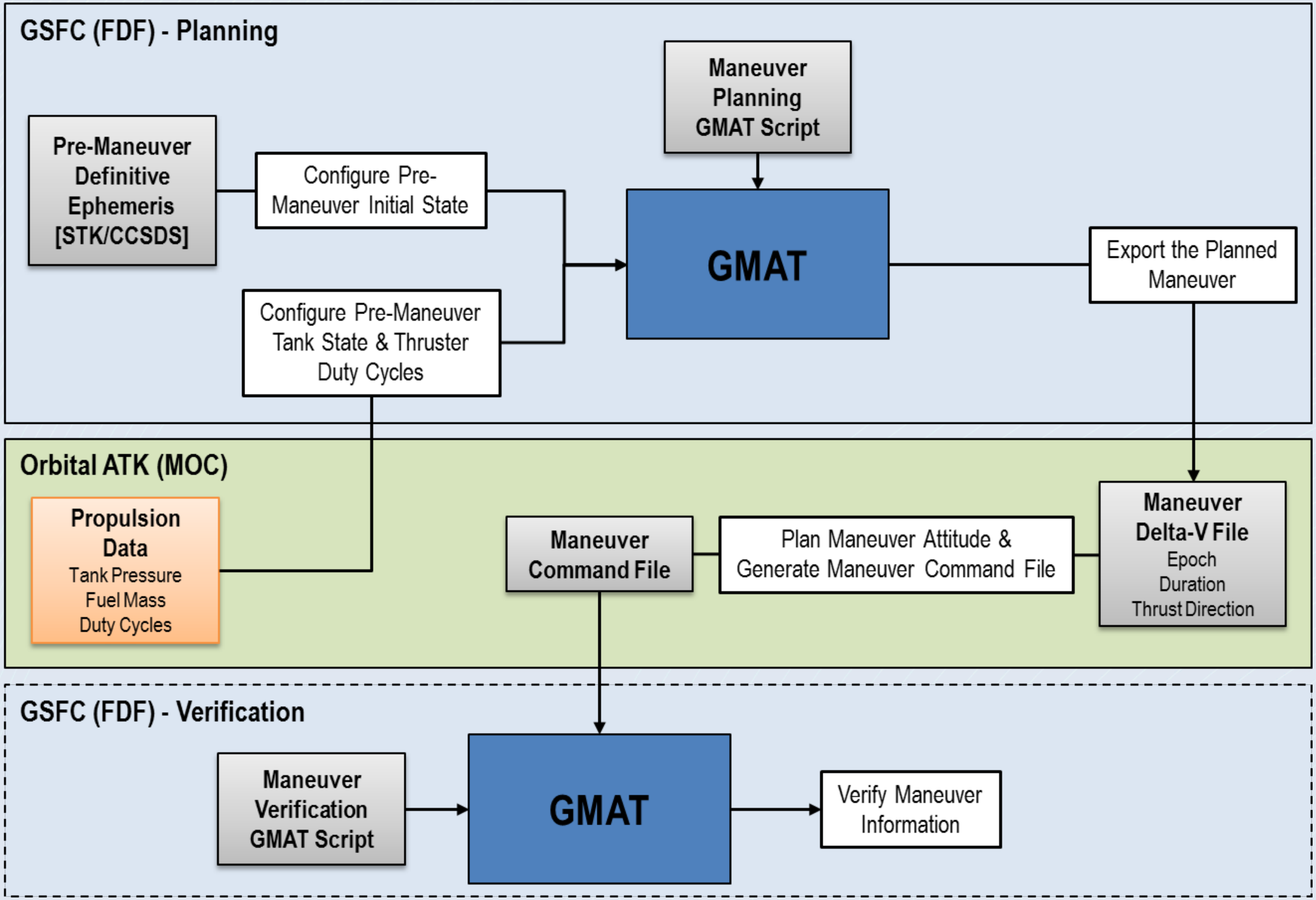
Monte Carlo Parameter	3-sigma Values
Apogee Radius Error (km)	31,000
Perigee Radius Error (km)	15
Inclination Error (deg)	0.1
RAAN Error (deg)	0.3
AOP Error (deg)	0.3

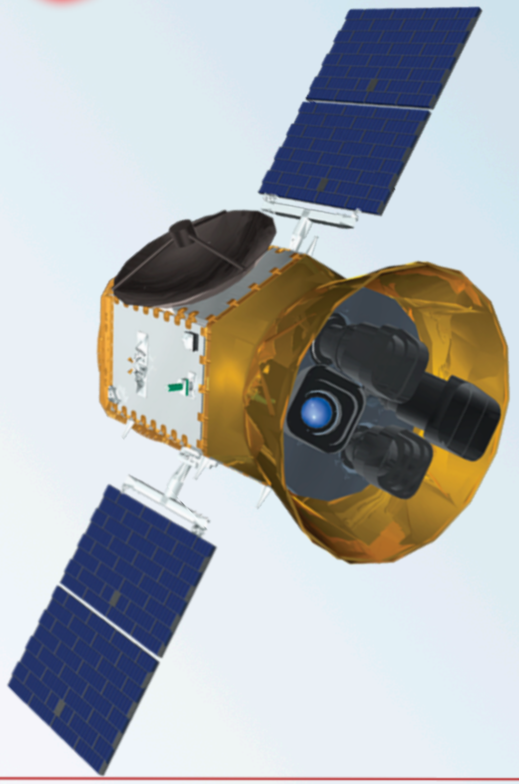


07: Maneuver Planning

Ryan Lebois
March 11, 2015

- Maneuver planning high-level data flow





08: Launch Window Analysis

TESS Mission Design Pre-CDR Peer Review

Joel Parker

March 11, 2015

Requirement

- Launch time is dictated by a lunar encounter.
  - *A large launch window is not available, because we need to correct for orbit plane errors.*
  - *Minimum window requirement is intended to allow for minor range issues at launch.*
- Launch window requirement:

ID	Title	Requirement
L3_FD_22	Launch Window	FD shall design for launch windows of at least 5 minutes during each day of the launch period.

- Two possible interpretations:
  - *Minimum requirement: 5-minute total duration (possibly non-centered)*
  - *LV Target Spec: ±5 minutes off-nominal (10 minutes total, centered)*
- Results show proposal for revised requirement

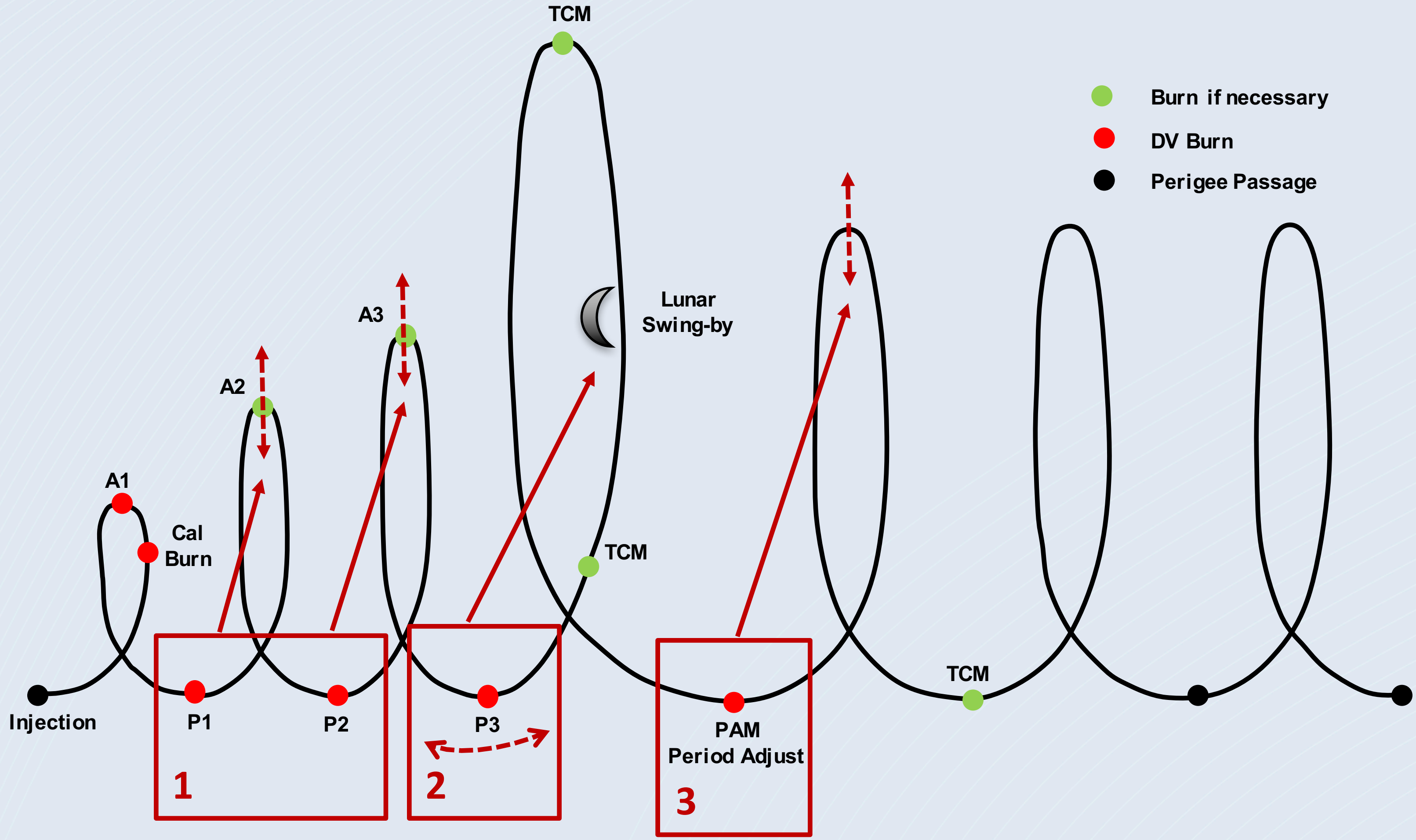
Delta-V Budget

Event	Planned ΔV (m/s)	Current ΔV (m/s) Aug 2017
A1	20	0–17
P1	35	31–50
P2	20	0-20
P3	5	0–8
Period Adjust Maneuver (PAM)	70	56–68
Deterministic Total	**150**	**109–131**
Launch Window Allowance	10	10
Launch Vehicle Dispersion	25	25–31
Trajectory Correction Maneuvers	10	15-26
Margin	20	22-55
Total	**215**	**215**

Analysis Approach

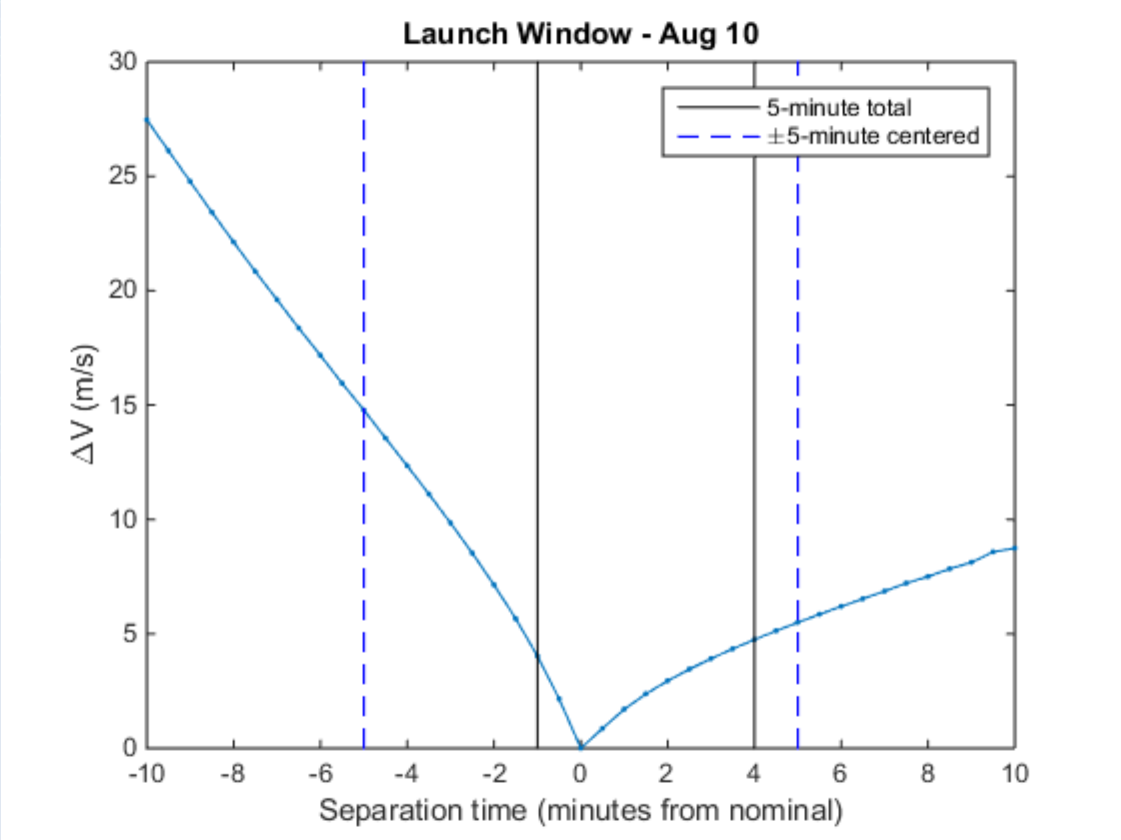
- To simulate launch time deviation:
  - *Start with an Earth-Fixed state for nominal launch*
  - *Vary separation time by appropriate amount (e.g. 1 minute)*
  - *Keep Earth-Fixed state numerically identical at new time*
  - *This maps to a RAAN dispersion (4 min = 1°)*
- Two possible retargeting strategies:
  1. *Retarget nominal flyby → achieve nominal mission orbit*
  2. *Replan flyby → achieve acceptable mission orbit*
- Preliminary analysis indicates strategy #1 is prohibitively expensive
- This analysis focuses on strategy #2
- Proof-of-concept analysis at this stage
  - *Exact strategy is a work in progress*

Current Retargeting Strategy



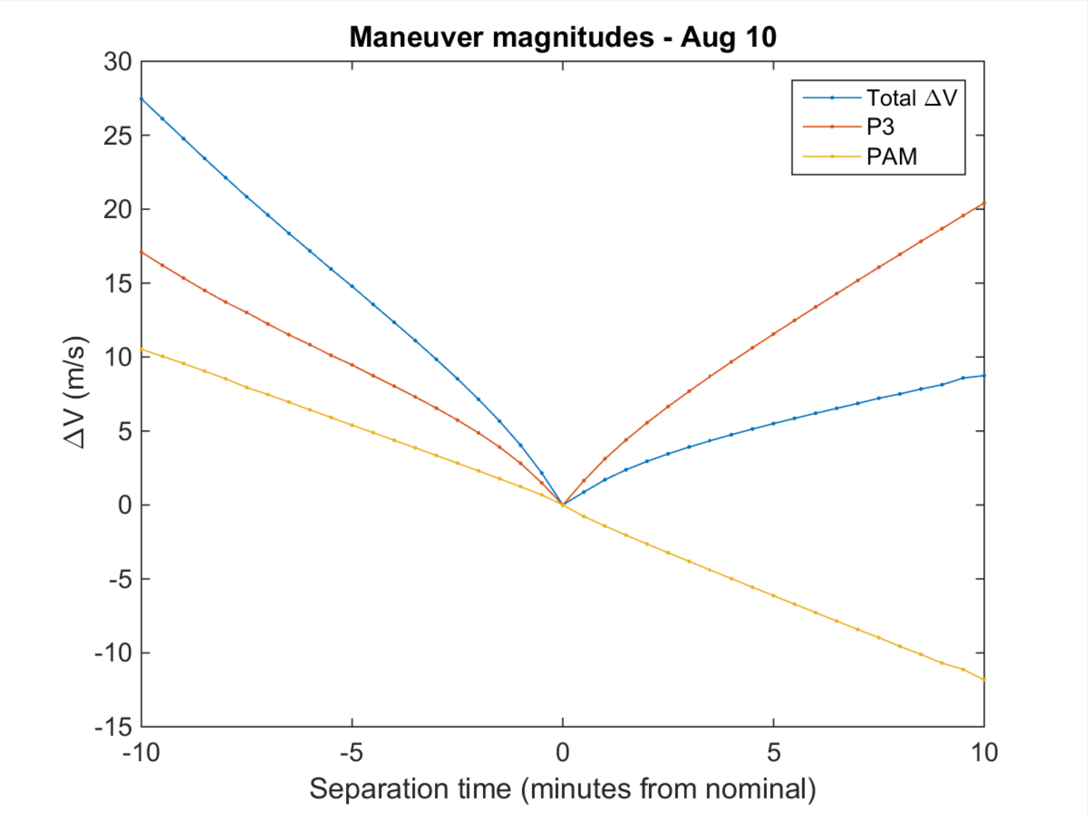
1. Retarget P1/P2 to correct timing at P3
2. Retarget P3 to achieve nominal B-plane (optimize)
3. Retarget PAM to achieve 2:1 resonance (optimize)

Single-Day Results: Aug 10



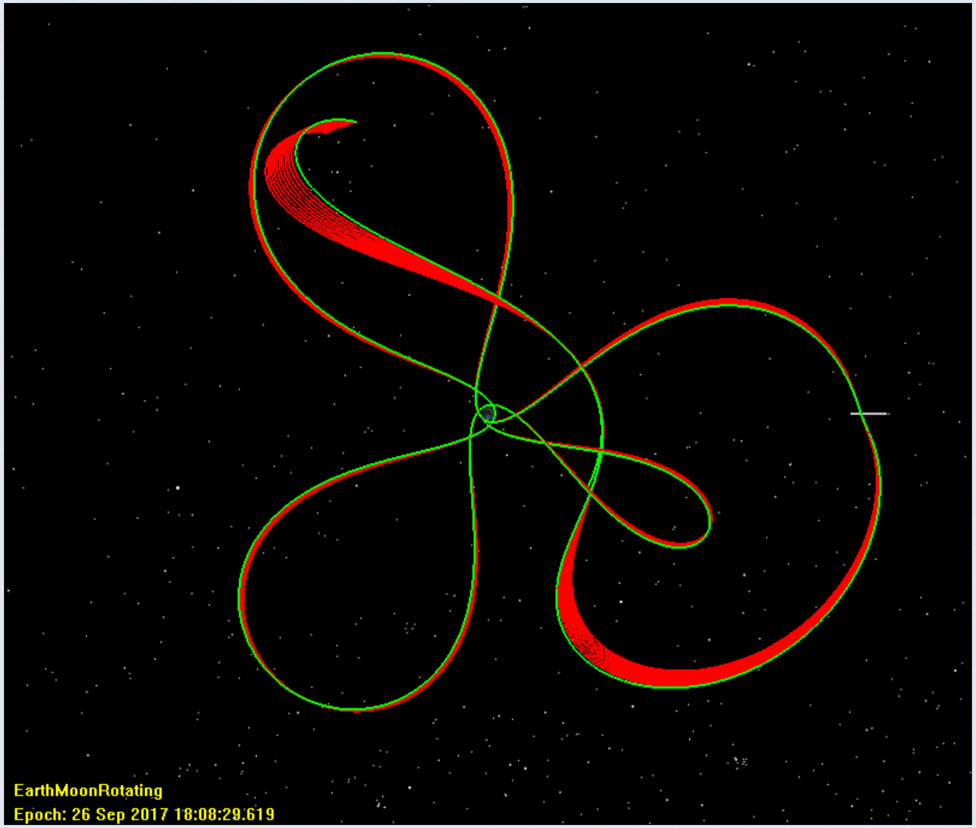
- Black region: 5-minute total window, centered to minimize average dV across window
- Blue region: ±5-minute window
- Clear asymmetrical behavior: positive offset less costly than negative offset
- Different flyby geometries causing PAM to do less work to achieve resonance

Single-Day Results: Aug 10



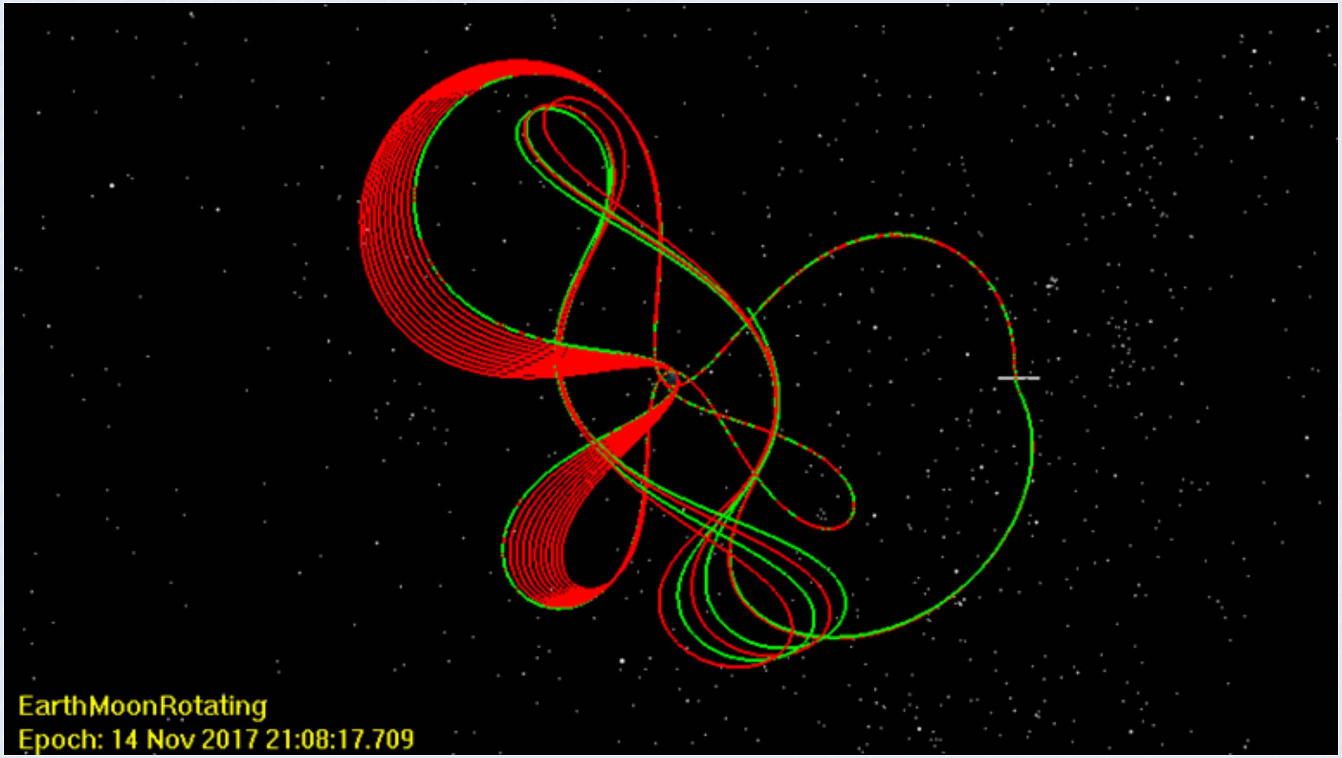
- P3 is roughly symmetric across the nominal launch time
- PAM can shrink by ~1m/s per minute for positive offsets

Single-Day Results: Aug 10



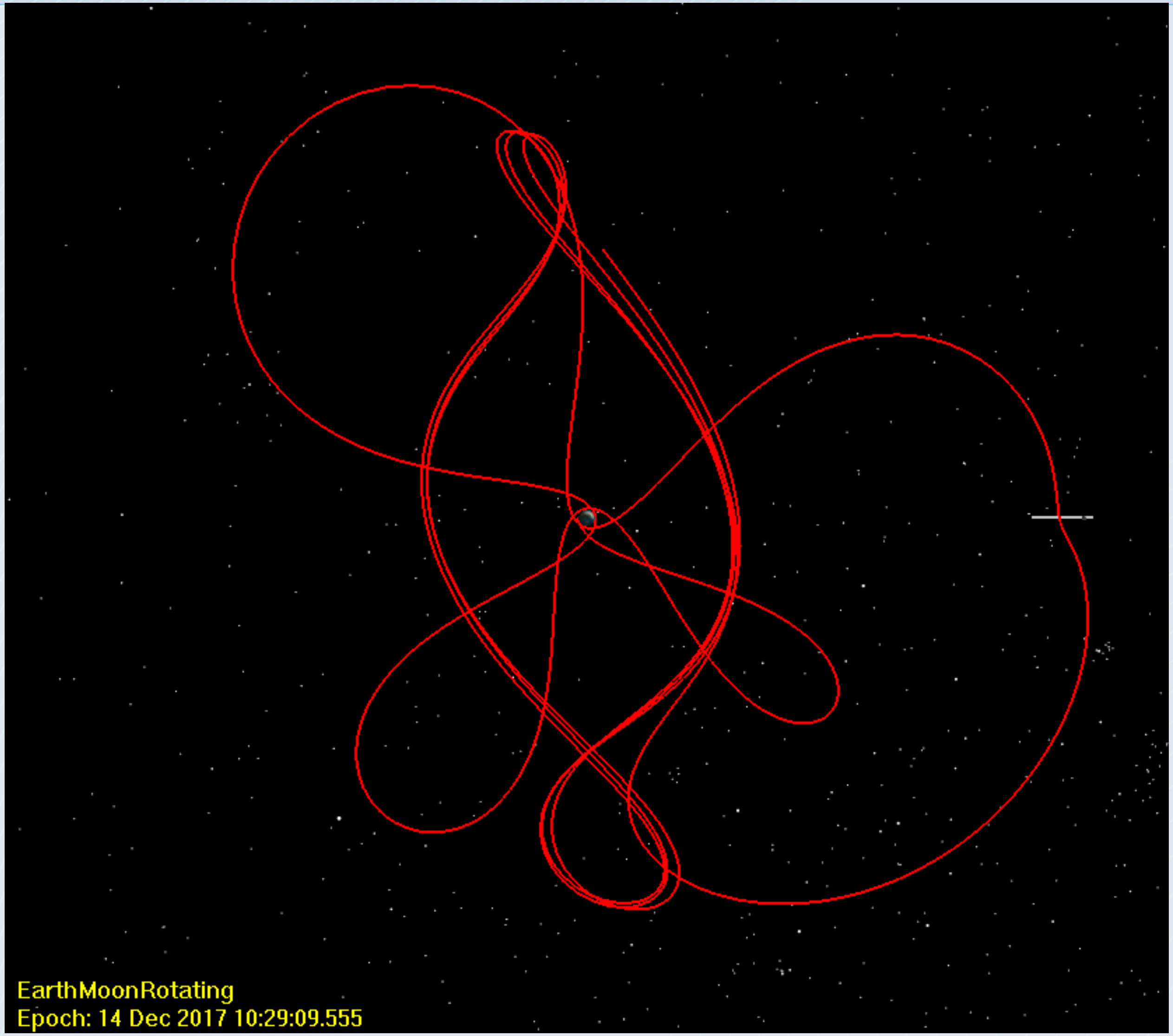
- Offset in launch time leads to rotation in RAAN
- Through flyby, leads to modified transfer orbit
  - *Timing difference produces rotation in mission orbit*

Missed/Partial P1 burn: 8/10/17 launch



- If P1 maneuver is missed/partial, we redesign P2 maneuver so we arrive at P3 at the desired time.
- DV cost is up to 15 m/s for a completely missed P1 burn.

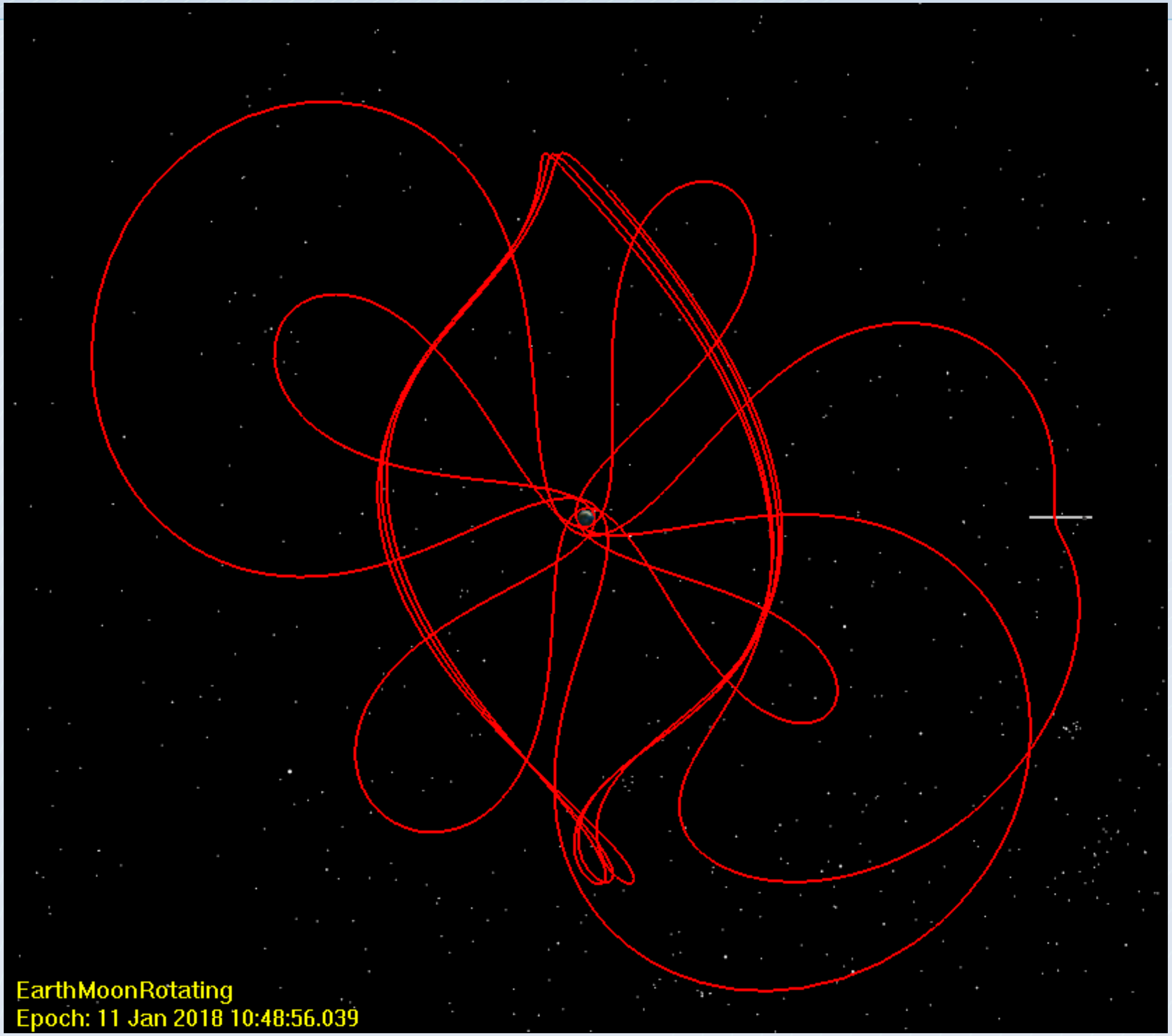
Partial P2 105%: Modify P3



If P2 is within 10% of ideal, we can simply modify the P3 burn to target B-plane parameters and achieve mission orbit

DV cost is about 20 m/s for a 10% error

Missed P2: Wait a lunar cycle and add 3 loops



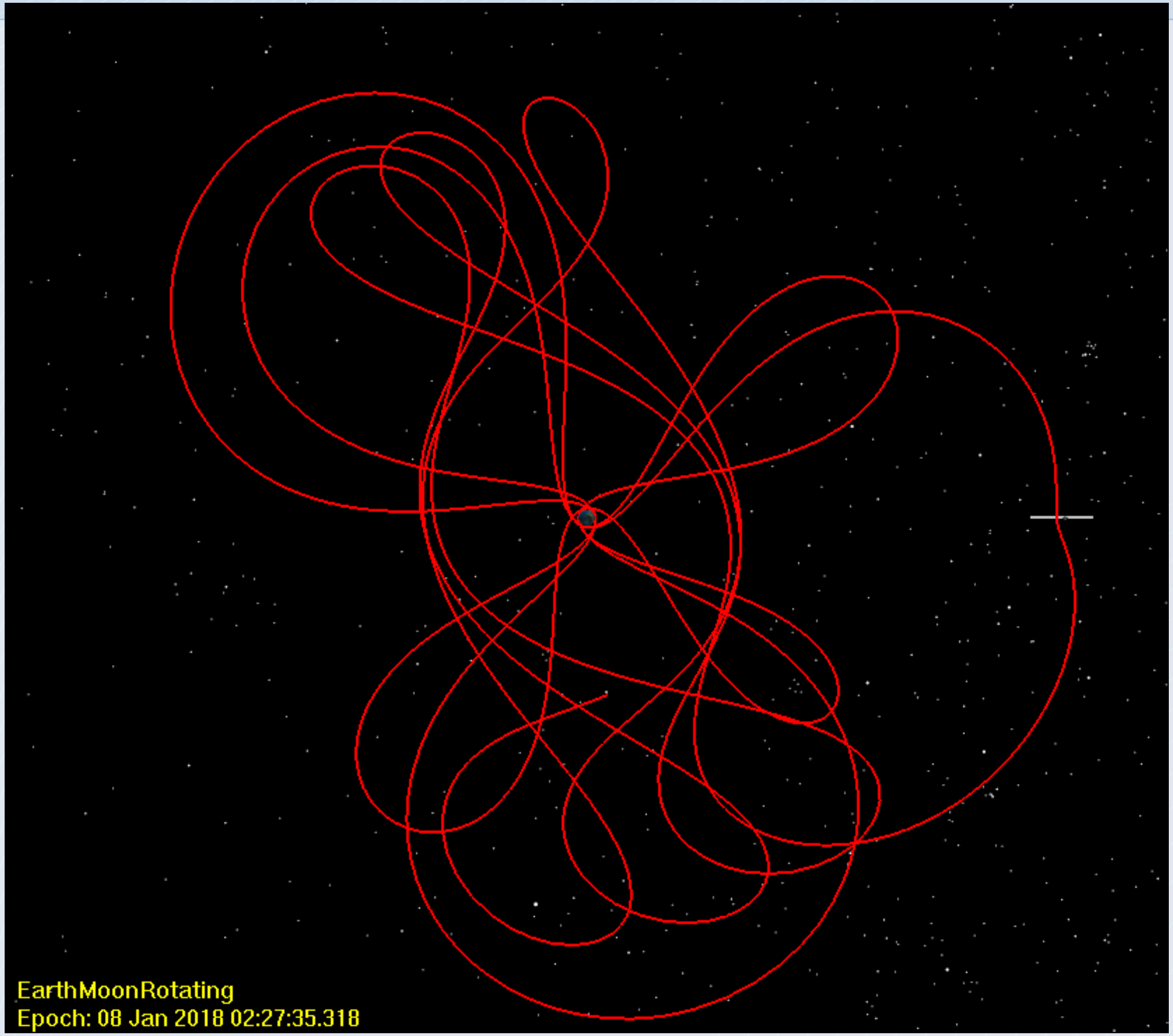
If P2 is missed entirely, we cannot achieve flyby after P3. Instead we wait a month and perform 3 more loops.

In this case we

1. perform no P3 maneuver,
2. add P4 to change phasing loop shape so we reach P6 1 lunar cycle after planned P3,
3. Perform P6 away from perigee with components in all 3 directions

DV cost is about 40 m/s for a complete miss

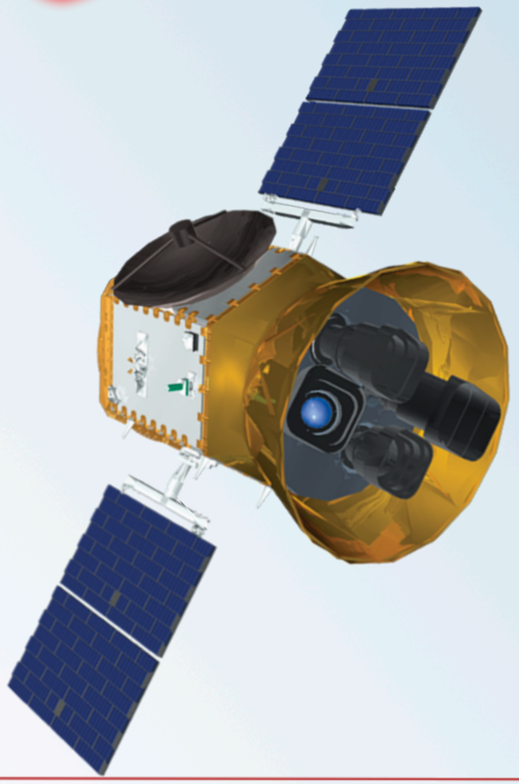
Partial P2 80%: Wait a month and add 3 loops



If P2 as much as ~80% completed, we still cannot achieve flyby after P3. But with no burn at P3 the Moon would warp the orbit badly. Instead we wait a month and perform 3 more loops.
In this case we

(1) perform P3 retrograde maneuver to lower apogee to ~300000 km
(2) add P4 to change phasing loop shape so we reach P6 1 lunar cycle after planned P3,
(3) Perform P6 away from perigee with components in all 3 directions

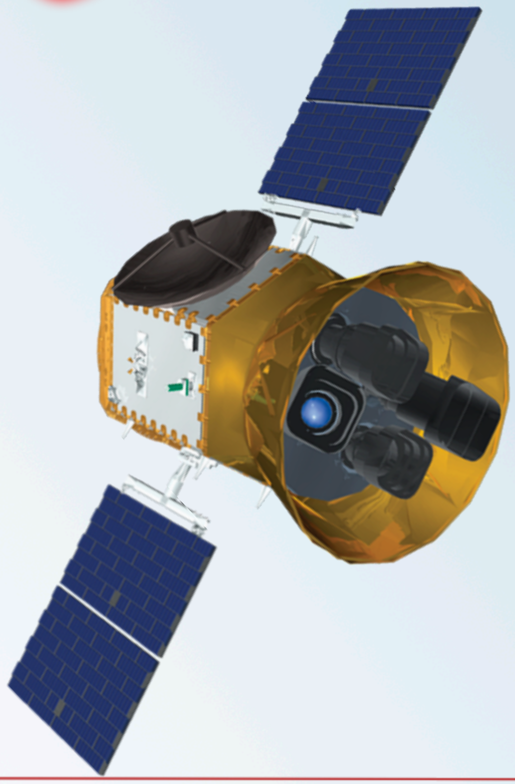
DV cost can be high, about 90 m/s, because we must lower then raise apogee. We may also need to raise perigee with an A4 maneuver.



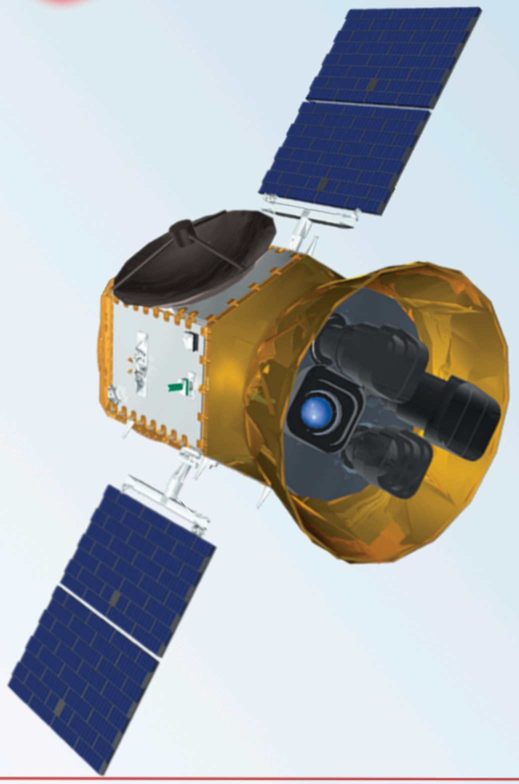
09: Recap

Recap

1. GMAT Project
2. GMAT Software Usage Fundamentals
3. Application to the Transiting Exoplanet Satellite Survey
    1. *Mission Overview*
    2. *Requirements*
    3. *Trajectory Design Process*
    4. *Solution Generation Process*
    5. *Finite Burn Modeling*
    6. *Launch Vehicle Dispersion Analysis*
    7. *Maneuver Planning*
    8. *Launch Window Analysis*



Backup



13: Results Summary

Don Dichmann
March 11, 2015

Delta-V Budget from Requirements

Event	Planned ΔV (m/s)	Current ΔV (m/s) Aug 2017
A1	20	0–17
P1	35	31–50
P2	20	0-20
P3	5	0–8
Period Adjust Maneuver (PAM)	70	56–68
Deterministic Total	**150**	**109–131**
Launch Window Allowance	10	10
Launch Vehicle Dispersion	25	25–31
Trajectory Correction Maneuvers	10	15-26
Margin	20	22-55
Total	**215**	**215**

- This budget specifies the total delta-V of 215 m/s available
- It provides guidelines on how to distribute the budget
- DV for each maneuver changes day to day
- We have conformed closely with the Launch Window Allowance the Launch Vehicle Dispersion guidelines, with some changes required based on our analysis
- Margin captures the remaining DV from 215 m/s after the other items are summed

Process Overview

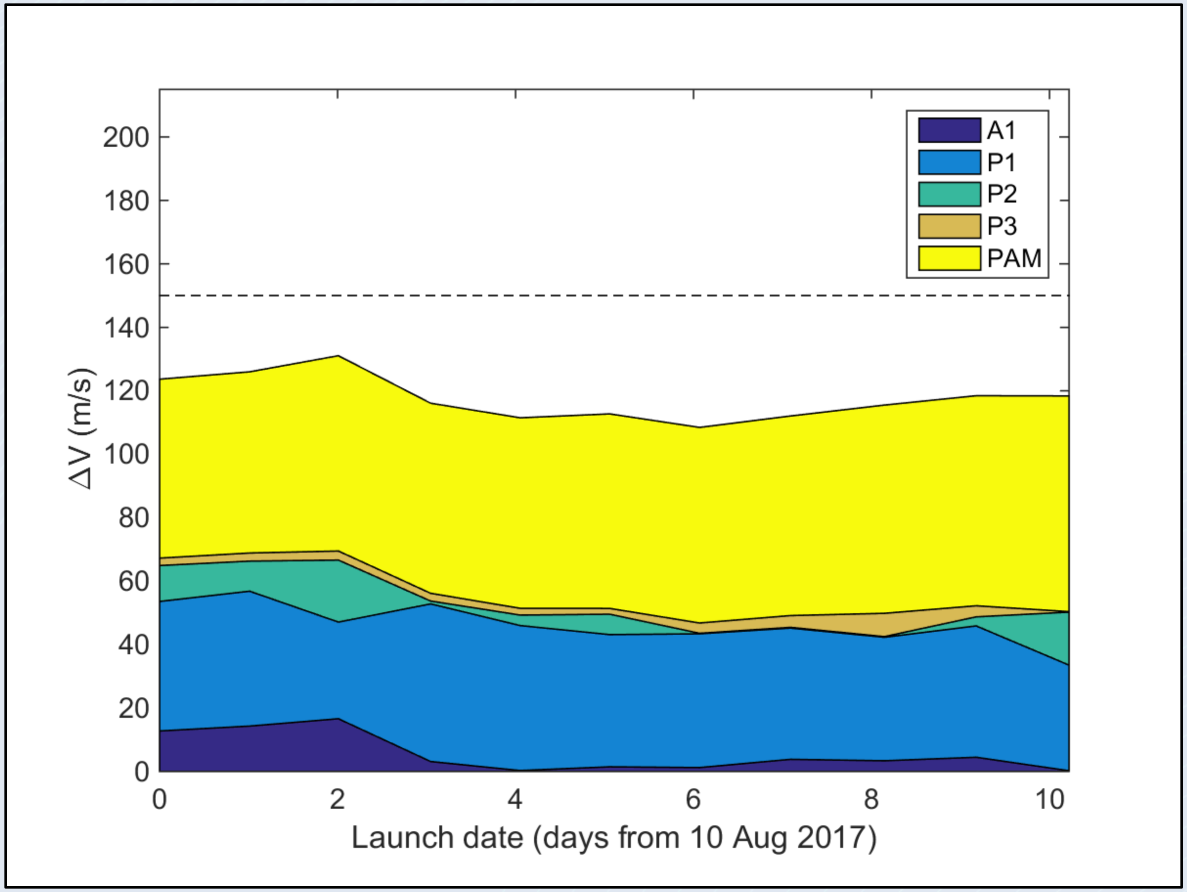
- We first generated deterministic solutions using impulsive-burn modeling
- Launch dispersion, launch window and statistical DV analysis are based on impulsive-burn trajectories
- Launch dispersion currently for 8/10, 8/15, 8/19
  - *We currently fill in with these values for other dates*
- Launch window analysis for each date
- Statistical DV values currently for 8/10, 8/15, 8/19
  - *Based on conservative estimate of 10% (3-sigma) error, to represent maneuver execution error and OD error*
  - *We currently fill in with these values for other dates*
  - *Monte Carlo simulation is being designed to enhance model fidelity*
- We do not include finite burn results explicitly in the summary DV budget
  - *However we have shown that finite burn does not produce a large DV penalty*

Delta-V Budget

					Launch Date						
Maneuver (m/s)	10-Aug	11-Aug	12-Aug	13-Aug	14-Aug	15-Aug	16-Aug	17-Aug	18-Aug	19-Aug	20-Aug
A1	12.7	14.3	16.6	3.1	0.3	1.5	1.2	3.8	3.4	4.5	0.2
P1	40.8	42.5	30.5	49.7	45.7	41.7	42.2	41.4	38.9	41.4	33.3
P2	11.3	9.5	19.5	0.9	3.3	6.4	0.2	0.2	0.2	2.9	16.7
P3	2.3	2.6	2.9	2.4	2.1	1.8	3.2	3.7	7.3	3.5	0.1
PAM	56.4	57.1	61.6	59.9	60.1	61.3	61.7	62.9	65.7	66.2	68.0
Deterministic total	123.6	126.0	131.0	116.1	111.5	112.7	108.4	112.0	115.5	118.4	118.4
Launch Window	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0
Launch Vehicle Dispersion	25.2	25.2	25.2	25.2	25.2	25.3	25.3	25.3	25.3	30.9	30.9
Statistic DV	26.4	26.4	26.4	26.4	26.4	16.4	16.4	16.4	16.4	14.8	14.8
Subtotal	185.3	187.6	192.7	177.7	173.1	164.4	160.1	163.7	167.2	174.1	174.1
Margin	29.7	27.4	22.3	37.3	41.9	50.6	54.9	51.3	47.8	40.9	40.9
total	215.0	215.0	215.0	215.0	215.0	215.0	215.0	215.0	215.0	215.0	215.0

- Eleven launch dates found in Aug 2017, exceeding 5-day requirement
- All solutions remain within 215 m/s budget
- Budget shows margin of at least 29 m/s, which could be used for contingency. This is enough to recover from many missed/partial burns, but not enough to recover from all missed P2 burns.

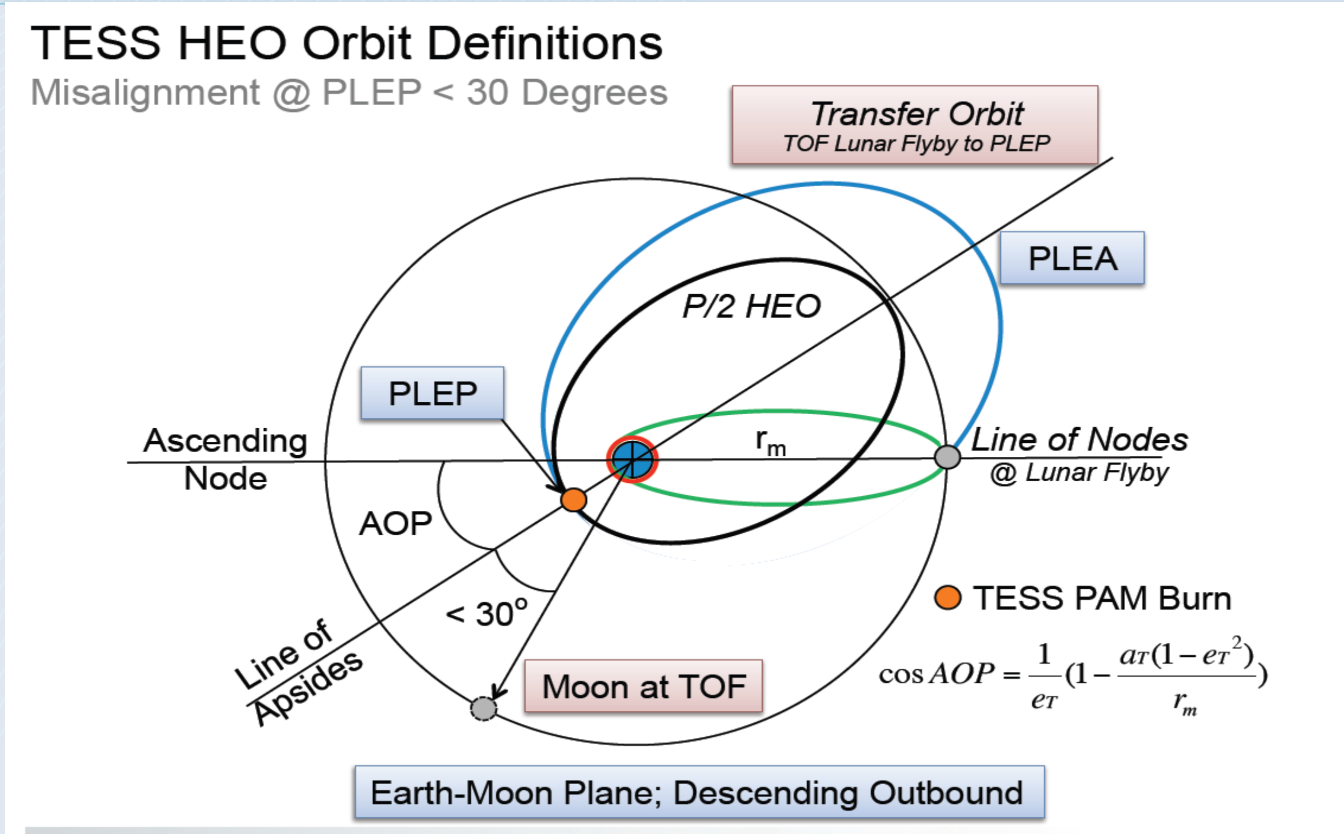
Impulsive Deterministic Budget



Solutions Meet Key Design Drivers

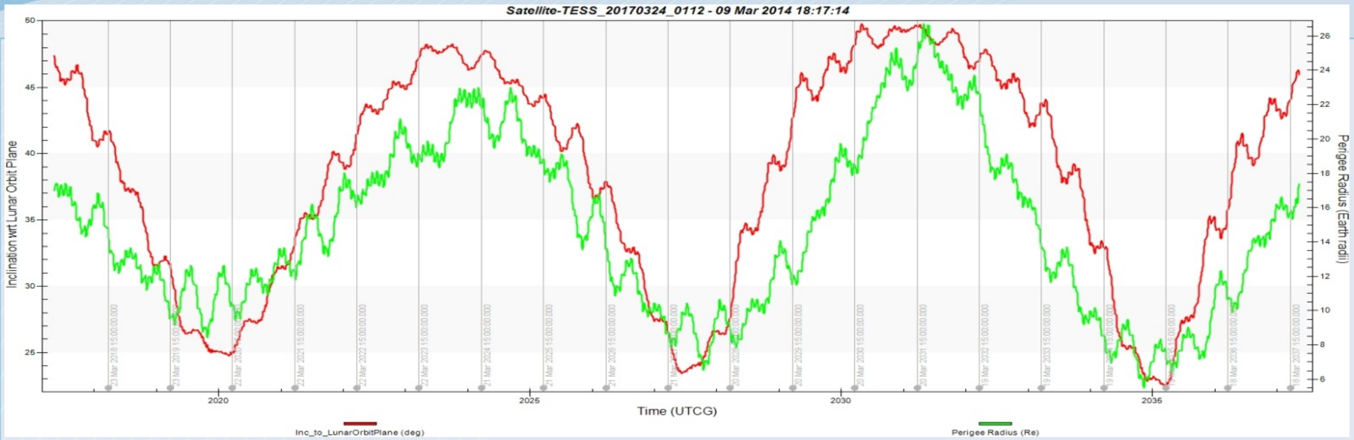
- Eleven launch dates in Aug 2017, exceedingly the requirement of five days (L3_FD_21)
  - *We also exceed five days in Sept and Oct 2017*
- Each trajectory selects for a launch date meets the Delta-V budget of 215 m/s
  - *Employ 3-sigma DV levels for Launch Dispersion and Statistical DV to meet requirement L3_FD_28*
- Achieve a 2:1 resonant orbit (L3_FD_1) and required phasing relative to the Moon for operational stability (L3_FD_2)
- Meet constraints on mission orbit for maximum perigee radius (L3_FD_29) and minimum perigee radius (L3_FD_3)
- Meet the Launch Window requirement of 5 minutes (L3_FD_22)
- Meet eclipse constraints (L3_FD_4)
- Meet the Sun angle constraint at PAM, the longest burn (L3_FD_24)

Orbit Geometry

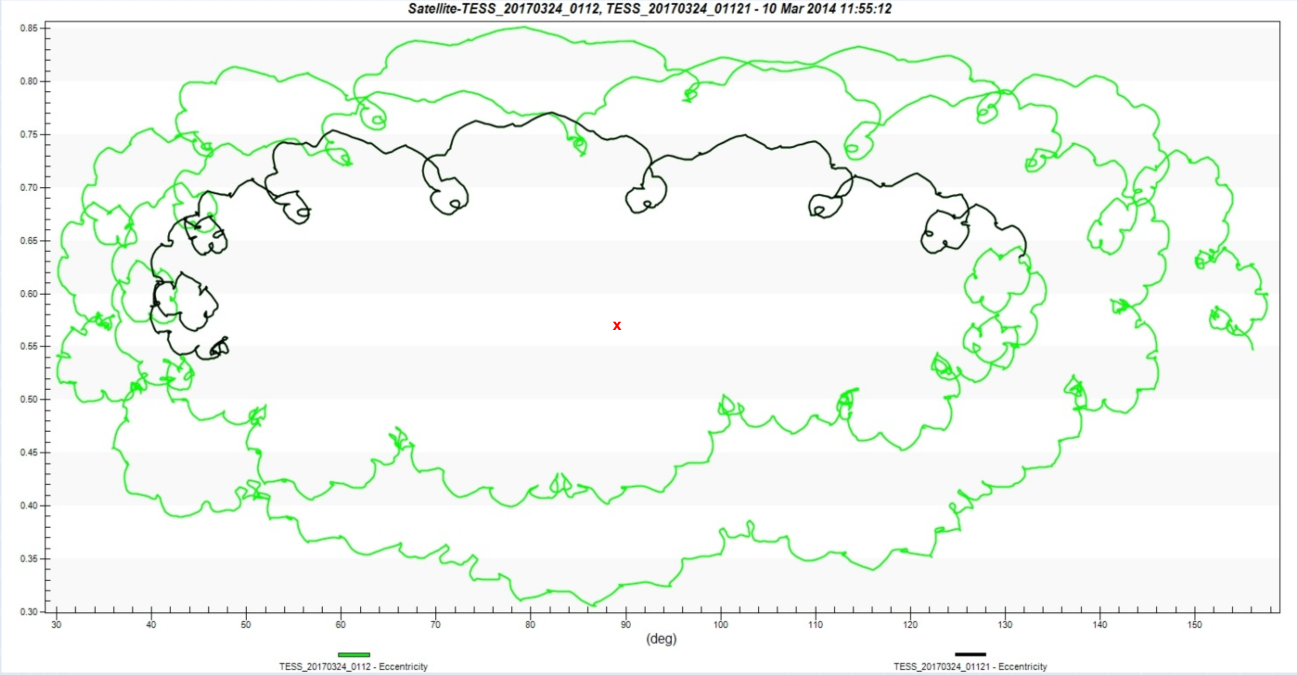


- Mission ecliptic AOP is chosen near 90 or 270 deg, to keep the line of apsides out of the ecliptic plane, and so avoid eclipses near apogee
- Transfer orbit apogee radius is chosen to achieve alignment with Moon at PLEP
  - This condition & 2:1 resonance keeps Moon 90 deg away at apogee and aids orbit stability
- Lunar Flyby is designed to achieve the desired transfer orbit from phasing loops
- Phasing loops' line of nodes is the Moon direction at flyby

Kozai Mechanism (cont'd)



Evolution of perigee radius (green) and lunar inclination (red) over 20 years. The oscillation period is about 8 years



Evolution of ecliptic AOP and eccentricity (green) over 20 years. Black curve represents 1st 4 years. The solution librates about (**X**) with AOP = 90 deg and eccentricity = 0.55

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

Modeling Process

- Two optimization scripts
- First script to design from Translunar Injection (TLI) through flyby to Science Orbit
- Second script to design from Launch Vehicle injection to Science Orbit
- In each script, we start with simplified 2- and 3-body assumptions to define the shape of the trajectory arcs
- We then use constrained optimization, high-fidelity force modeling and numerical propagation in GMAT to converge on a smooth solution

Initial Guess Constraints

- J2000 inclination at LV separation = 28.5 deg
  - *We have also modeled 38 deg for Wallops*
- Separation altitude = 200 km
- TLI (aka P3) occurs at perigee
- Science orbit initially in 2:1 resonance with Moon
  - *Implies semimajor axis is 38 Re*
- PAM radius = 17 Re
  - *Implies apogee radius and eccentricity*
  - *This is also the transfer orbit perigee radius*
- PAM occurs at perigee
- PLEP misalignment <= max value
  - *Current results assume max value = 30 deg*
  - *Note that 30 deg is not a hard boundary. Slightly larger angles would meet mission goals*

Modeling Assumptions

- Kozai parameter for Science orbit: 0.65 used for 1st guess
  - *Value based on Aerospace Corp analysis, to meet mission constraints on perigee*
- Tisserand value T before and after flyby: 1.14 used for 1st guess
  - *This value is chosen to achieve desired PLEP misalignment from transfer orbit*
- Phasing loop apogee radius
  - *A1: 250,000 km (based on LV information)*
  - *A2: 328,600 km (based on previous GSFC analysis, but subject to change)*
  - *A3: equal to A4 (so that P3 maneuver is small, and not critical)*
  - *A4: chosen based on Tisserand value (typically about 1.03 x Moon orbit radius at flyby)*
- Adapt preflyby orbit plane based on Moon argument of latitude
  - *Typically if Moon is closer to its ascending (resp. descending) node, preflyby orbit plane is ascending (resp. descending) relative to Moon*
  - *However the code retains the option to use another switching rule or to make selection manually*
- Choose post-flyby orbit plane
  - *Again we can choose ascending or descending*
  - *We typically choose ascending, as it produces good DV, but we retain the option to select manually*

Modeling Assumptions (cont'd)

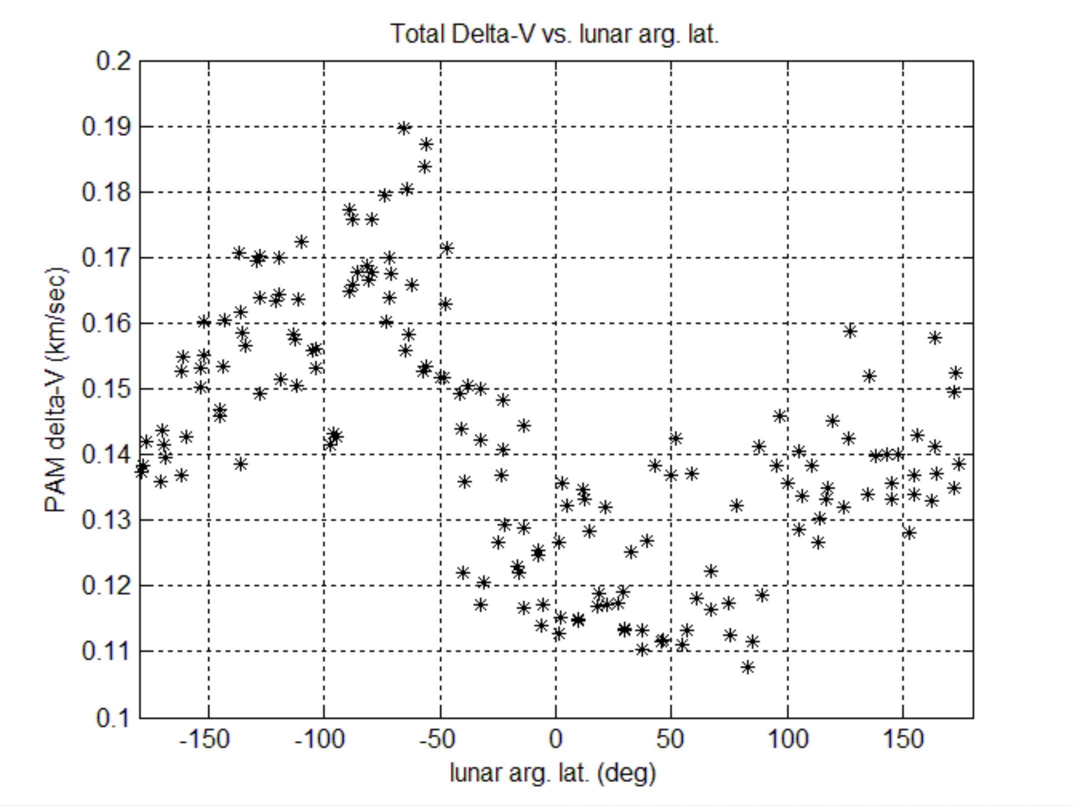
- Choose whether pre- and post-flyby arc is outbound (flyby before apogee) or inbound (flyby after apogee)
  - *Based on simulation results and Aerospace Corp analysis we typically choose inbound for pre-flyby, outbound for post-flyby*
  - *However, we retain the option to select manually*
- Perigee altitude for P1, P2, P3 >= 600 km
  - *We will not necessarily go this high, but we will need to keep perigee >= 200 km*
- J2000 inclination at TLI = 28.5 deg
  - *This is a simplified assumption that neglects change in inclination by ~ 1 deg from LV separation to TLI*
- CSR Mission plan includes maneuvers A1, P1, P2, P3 and PAM
- We previously added A2, A3 as optional maneuvers to improve convergence of scripts
  - *However, this led to some inefficient solutions, so A2 and A3 maneuvers are now zeroed out.*
- Currently we seek to find feasible solutions that are not necessarily optimal for delta-V. However, GMAT can support constrained optimization, and we will use it in the future.

Optimization Scripts

- Represents the trajectory from LV separation to Science Orbit in 5 segments
    1. *LV separation to TLI*
    2. *TLI to flyby – 3 days*
    3. *Flyby -3 days to flyby + 6 days*
    4. *Flyby + 6 days to PAM*
    5. *PAM to next Earth apoapsis*

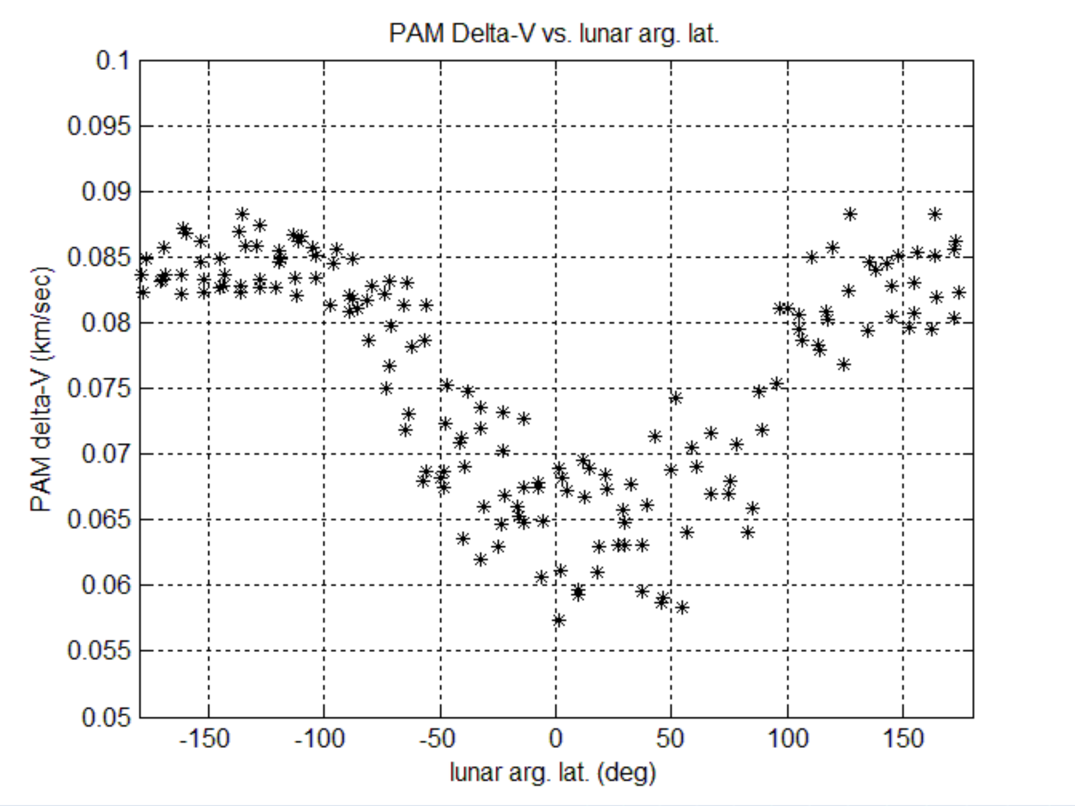
- Optimization scripts enforce continuity between segments

Deterministic Delta-V vs. Lunar Arg. Lat.



- Solutions over one year

PAM Delta-V vs. Lunar Arg. Lat.



Solutions over one year

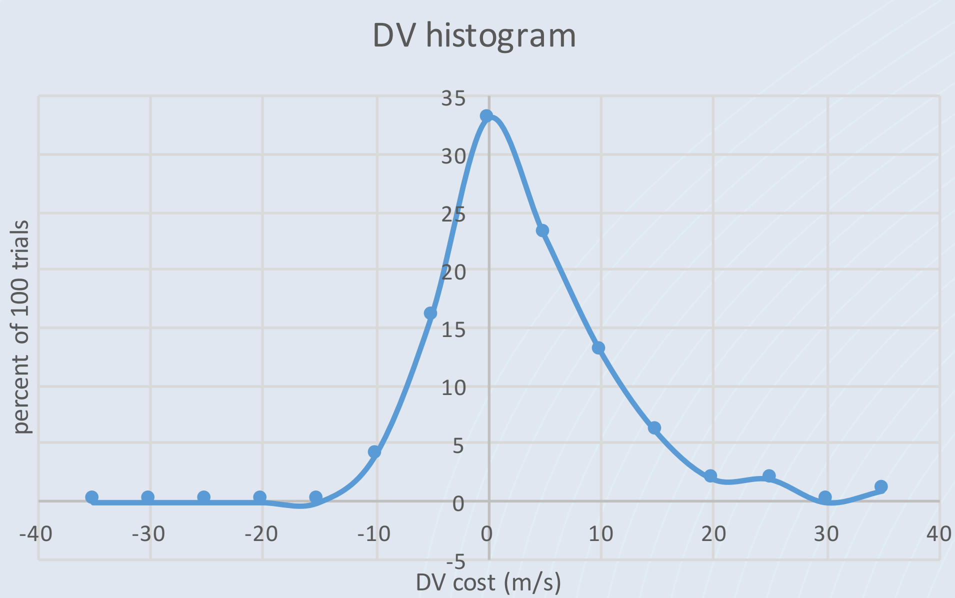
Phasing Loops Sequence Constraints

- Phasing loop apogee radius
  - *A1: 250,000 km (based on LV information)*
  - *A2: 328,600 km (based on previous GSFC analysis, but subject to change)*
  - *A3: equal to A4 (so that P3 maneuver is small, and not critical)*
  - *A4: chosen based on Tisserand value (typically about 1.03 x Moon orbit radius at flyby)*
- Phasing loop perigee altitude >= 600 km
  - *Lowering this to 200km may be possible, if necessary*
  - *Lunar perturbations makes this unnecessary for most dates*
- We previously added A2, A3 as optional maneuvers to improve convergence of scripts
  - *However this led to some inefficient solutions, so A2 and A3 maneuvers are now zeroed out.*

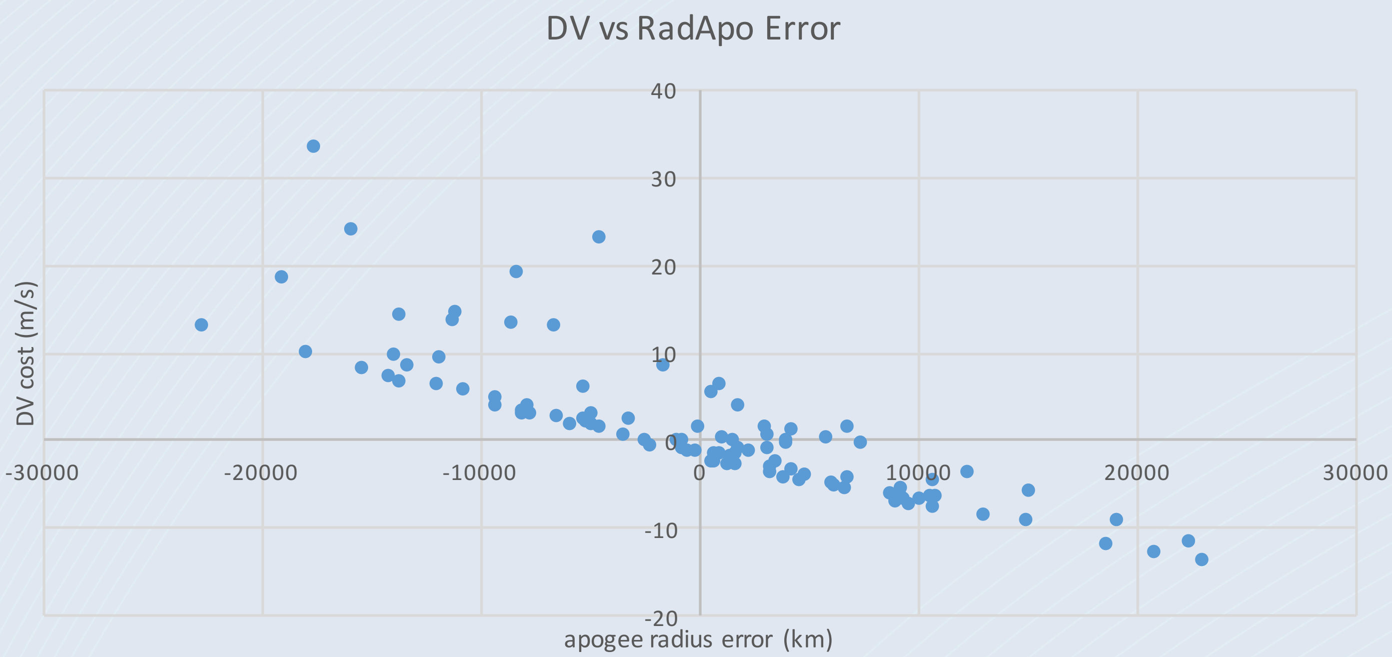
Results for Initial Monte Carlo Simulation

- For 8/10/2017 solution
- Based on 100 trials
- Mean DV cost is close to zero
- Sigma is 8 m/sec
- Result: mean + 3 sigma = 25.2 m/s

mean (m/s)	1.076589
sigma (m/s)	8.053759
mean + 3*sigma (m/s)	25.23786



Initial Monte Carlo Simulation (cont'd)

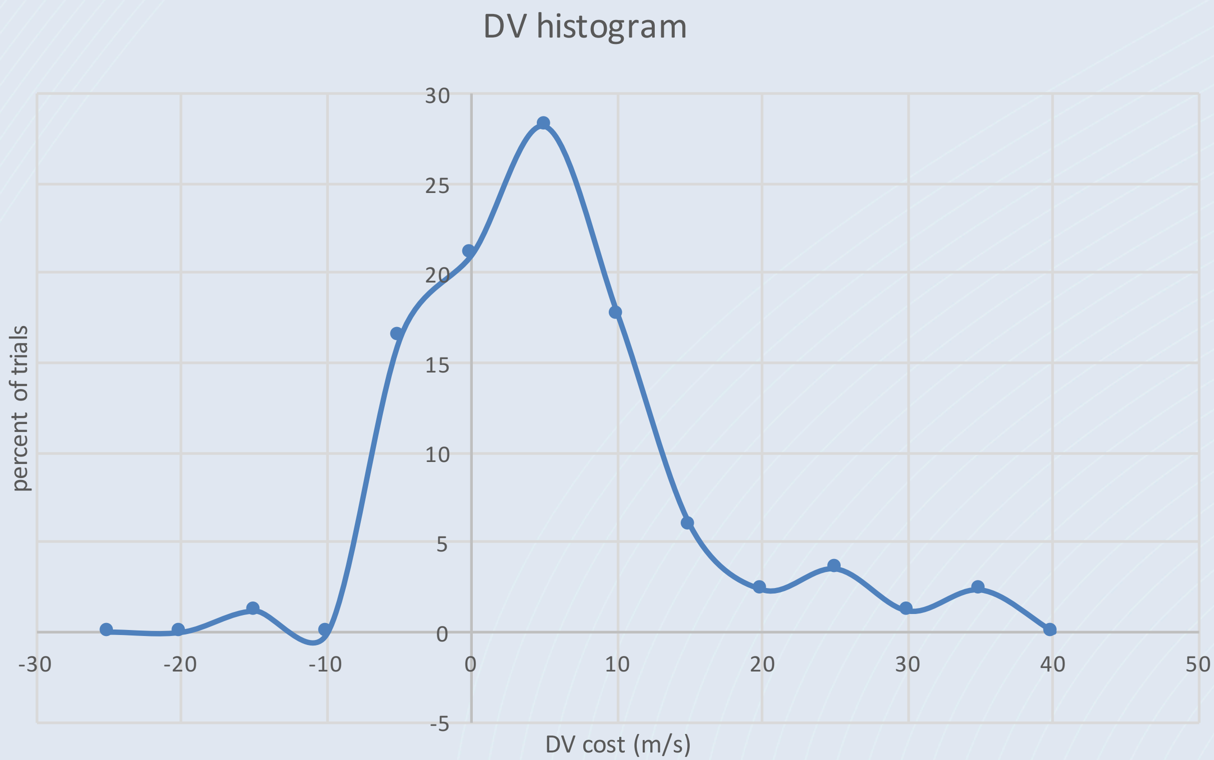


- The DV cost is dominated by Rad Apo error, as seen in Hypercube results
- Meaningful statistical

Results for Initial Monte Carlo Simulation

- For 8/15/2017 solution

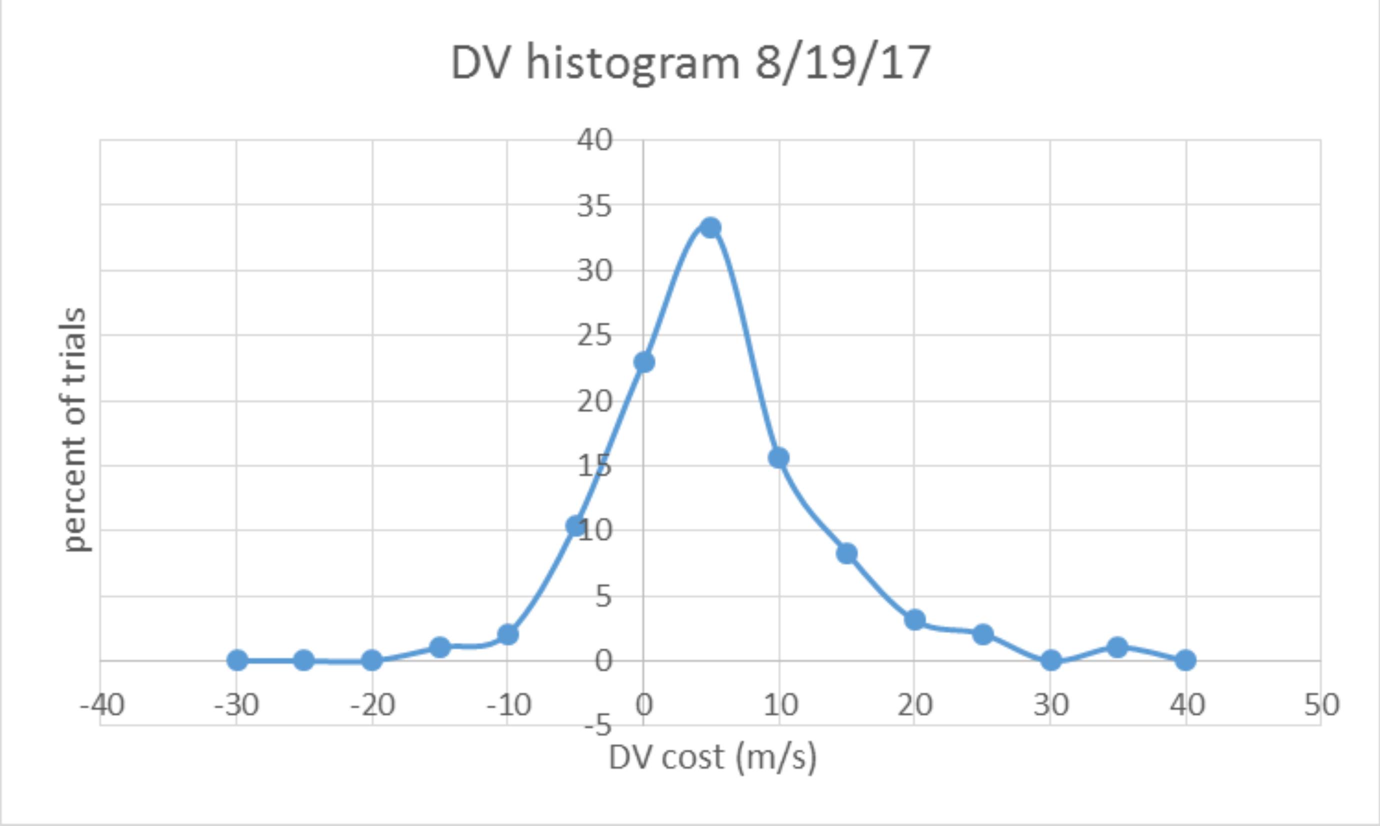
mean (m/s)	2.375522
sigma (m/s)	7.635464
mean + 3*sigma (m/s)	25.28191



Results for Initial Monte Carlo Simulation

- For 8/19/2017 solution

mean (m/s)	3.262704
sigma (m/s)	9.211424
mean + 3 sigma (m/s)	30.89698



Monte Carlo Results Summary

- Initial results show that the Mean + 3-sigma values (99% probability requirement) are consistent with the planned 25 m/s DV budget for launch dispersions
- The later launch dates leave less time in phasing loops to correct for energy errors that impact the timing of the flyby
  - *Will confirm this trend by extending the Monte Carlo simulations to all launch days*

Launch Date	10-Aug	15-Aug	19-Aug
Mean (m/s)	1.077	2.376	3.263
Sigma (m/s)	8.054	7.635	9.211
Mean + 3-sigma (m/s)	25.238	25.282	30.897

Conclusions

- Hypercube analysis indicates DV penalties are below 21 m/s
- Apogee radius error
  - *Dominates the potential DV cost*
  - *Also potential DV benefit from a positive apogee radius injection*
- The initial Monte Carlo results, with 100 draws, all have 3σ DV penalty of 30.9 m/s or less
  - *Mean value is 1.1 m/s for Aug 10, 2017 launch date*
  - *Mean value is 2.4 m/s for Aug 15, 2017 launch date*
  - *Mean value is 3.3 m/s for Aug 19, 2017 launch date*
- Thus the initial results fit well with the DV budget of 25 m/s originally allocated for launch dispersion error
  - *Extreme low apogee radius cases violate this*

ID	Parent ID	Title	Requirement	Compliance
L3_FD_28	MRD_104	Delta-V Budget	FD shall design ascent-to-mission orbit to require no more than 215 m/s delta-V with 99% probability of success.	**Comply.** May have to reallocate DV margin for extreme cases

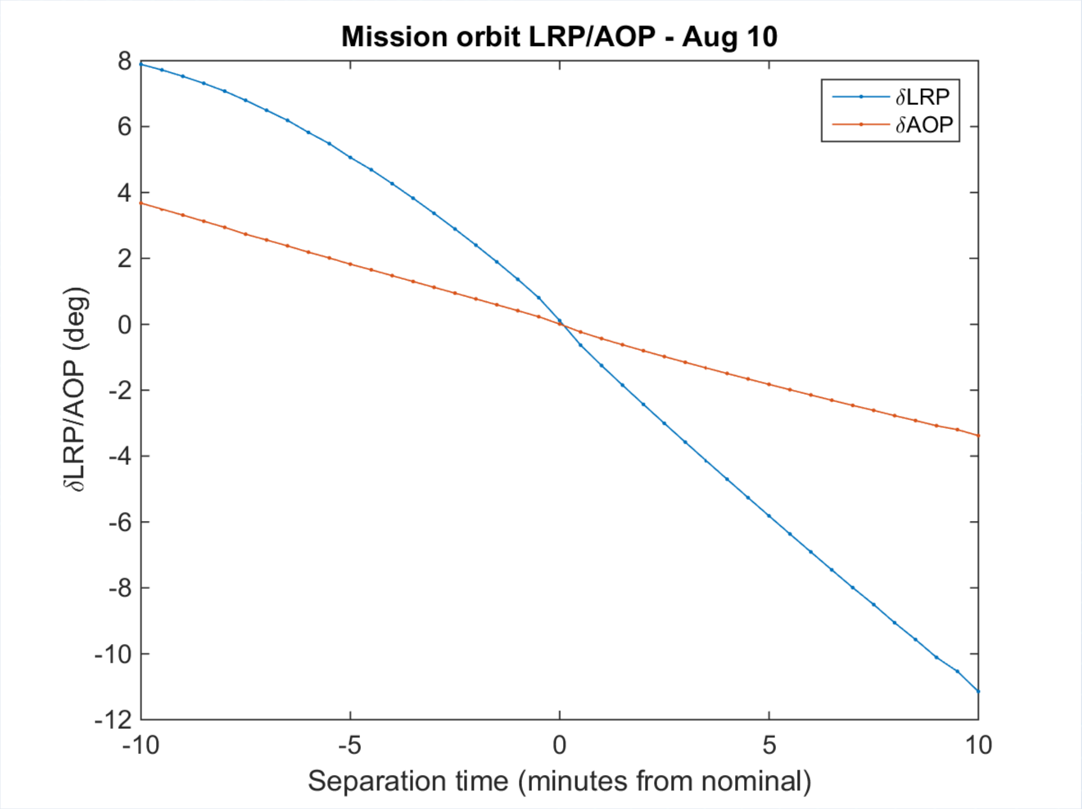
Delta-V Budget Revisited

Event	Planned ΔV (m/s)	Current ΔV (m/s) Aug 2017
A1	20	0–17
P1	35	31–50
P2	20	0-20
P3	5	0–8
Period Adjust Maneuver (PAM)	70	56–68
Deterministic Total	**150**	**109–131**
Launch Window Allowance	10	10
Launch Vehicle Dispersion	25	25–31
Trajectory Correction Maneuvers	10	15-26
Margin	20	22-55
Total	**215**	**215**

Future Work

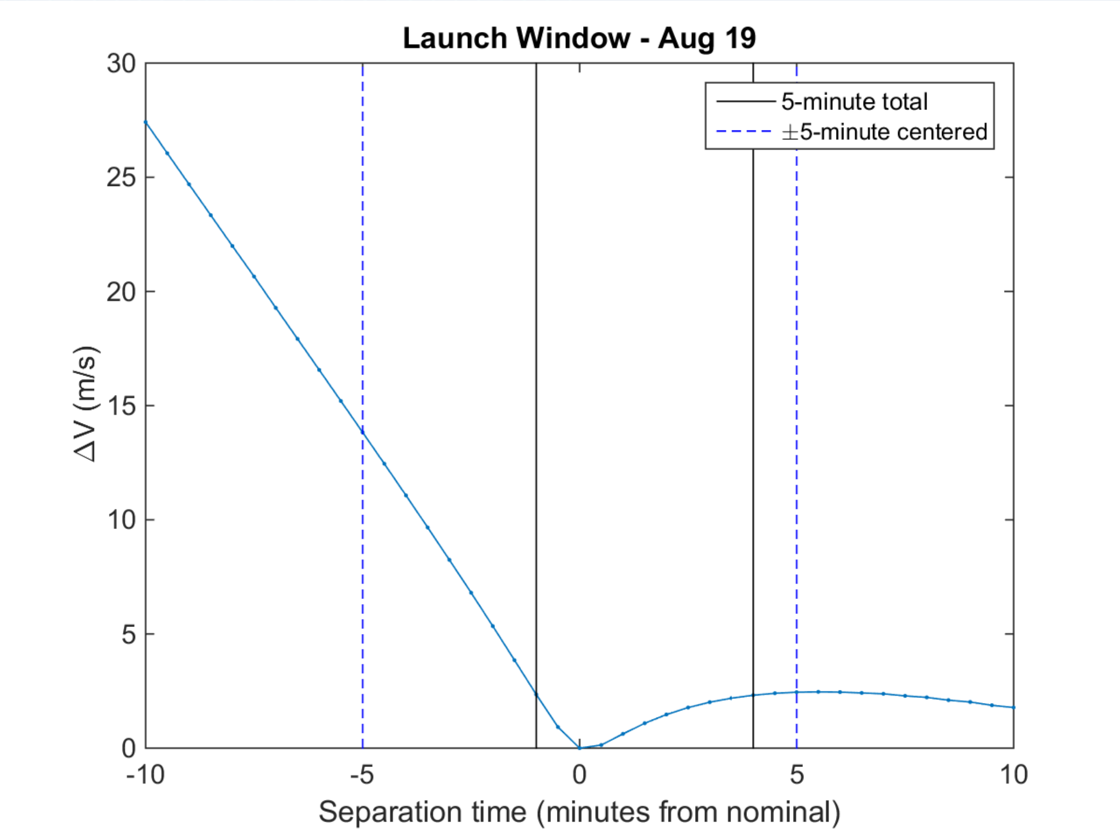
- Extend hypercube and Monte Carlo analysis to all launch days
- Try to improve the algorithm to be more efficient
  - *Inclination, RAAN, and AOP adjustments may be more efficient at/near apogee, and could reduce the burden and risk associated with P3*
  - *Examine updating nominal cases in which P2 is critical to do more apogee raising at P1; potential trade off between a less-than-optimal DV solution and reducing the risk and criticality of P2*
- Examine extreme (+25 m/s DV penalty) cases for better optimization and phasing loop design
  - *Determine is DV margin budget needs to be reallocated for launch dispersions*
- Add finite burn modeling
- Validate trajectory algorithms
- Further validate nominal mission orbit, add 25-year propagation to the Monte Carlo simulation
- Full launch covariance values from SpaceX are expected April 17, 2015, and we will run the Monte Carlo analysis based on those values

Single-Day Results: Aug 10



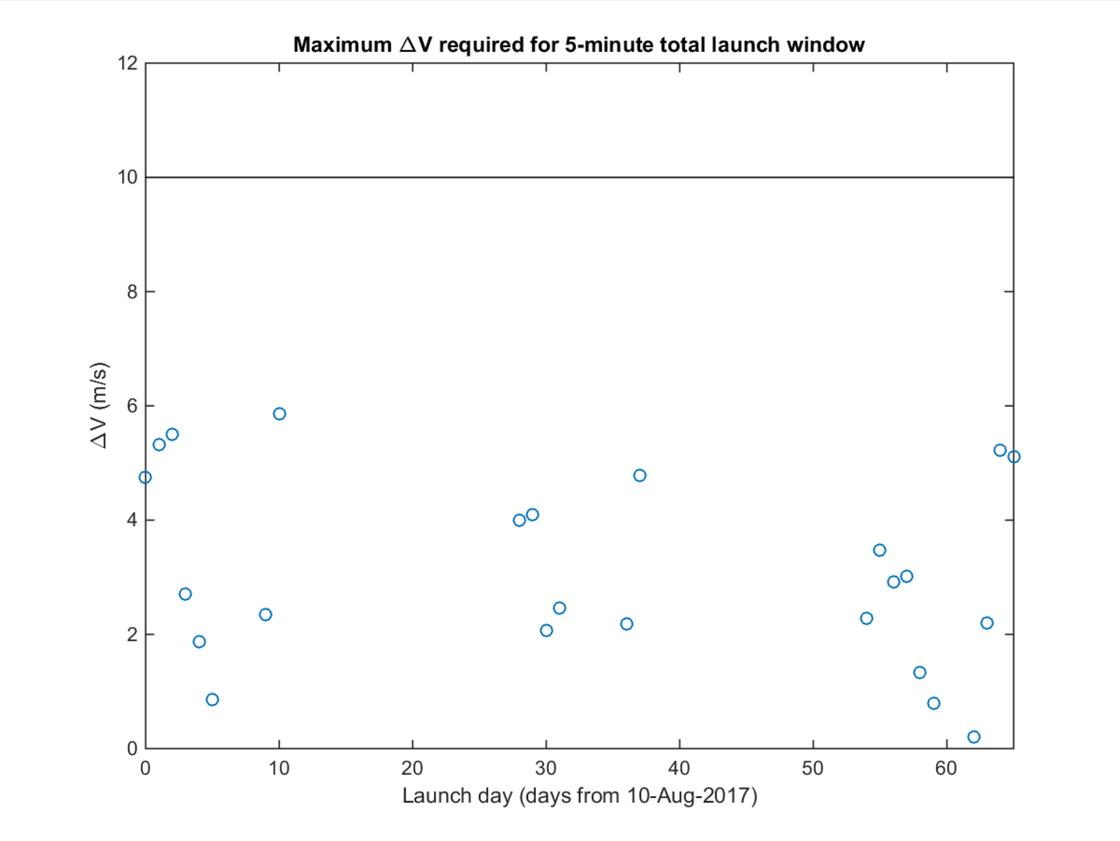
- Mission orbit LRP & AOP angles change due to change in flyby (and transfer orbit)
- Generally, we want higher AOP & lower LRP
- But, other differences within a few degrees do not indicate a bad orbit
- LRP/AOP are design metrics for stability & eclipse behavior

Single-Day Results: Aug 19



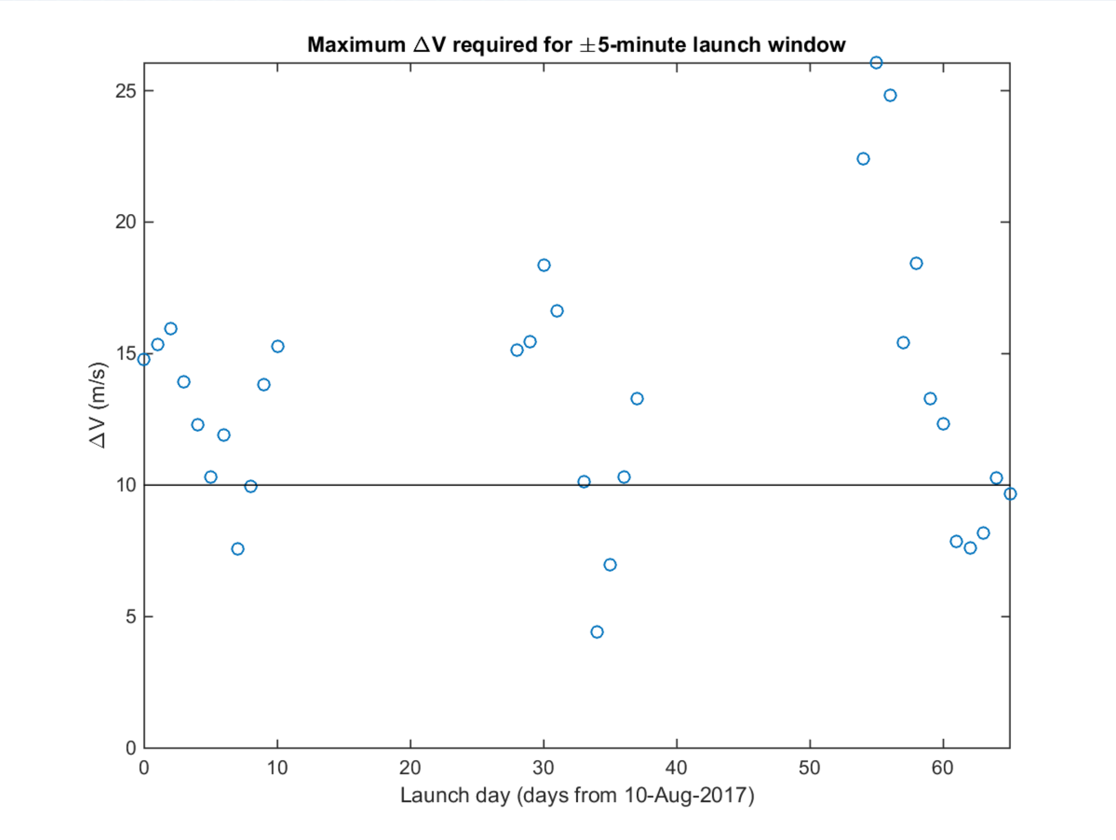
- The asymmetry is more apparent further in the lunar cycle
- Reduction in PAM overcoming increase in P3

Summary Results



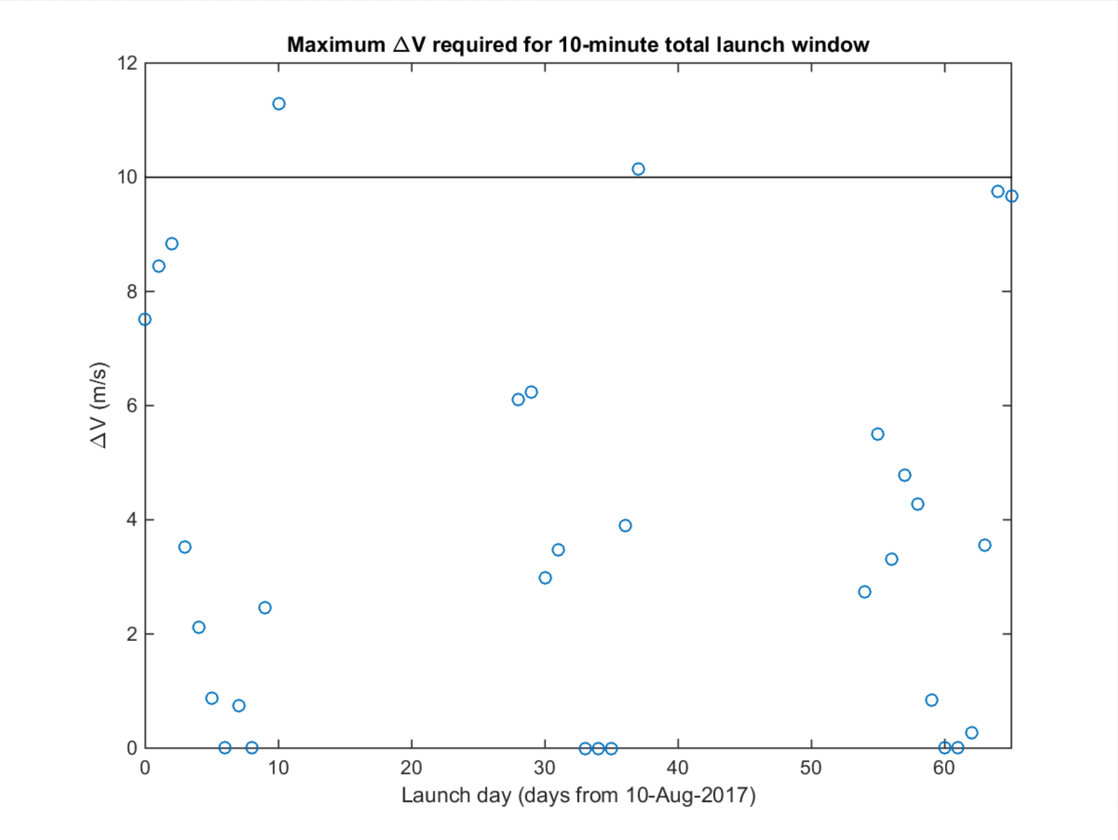
- Here we look at the required dV for all targets, for a 5-minute total window
- With this retargeting strategy, all launch days fit within 6 m/s

Summary Results



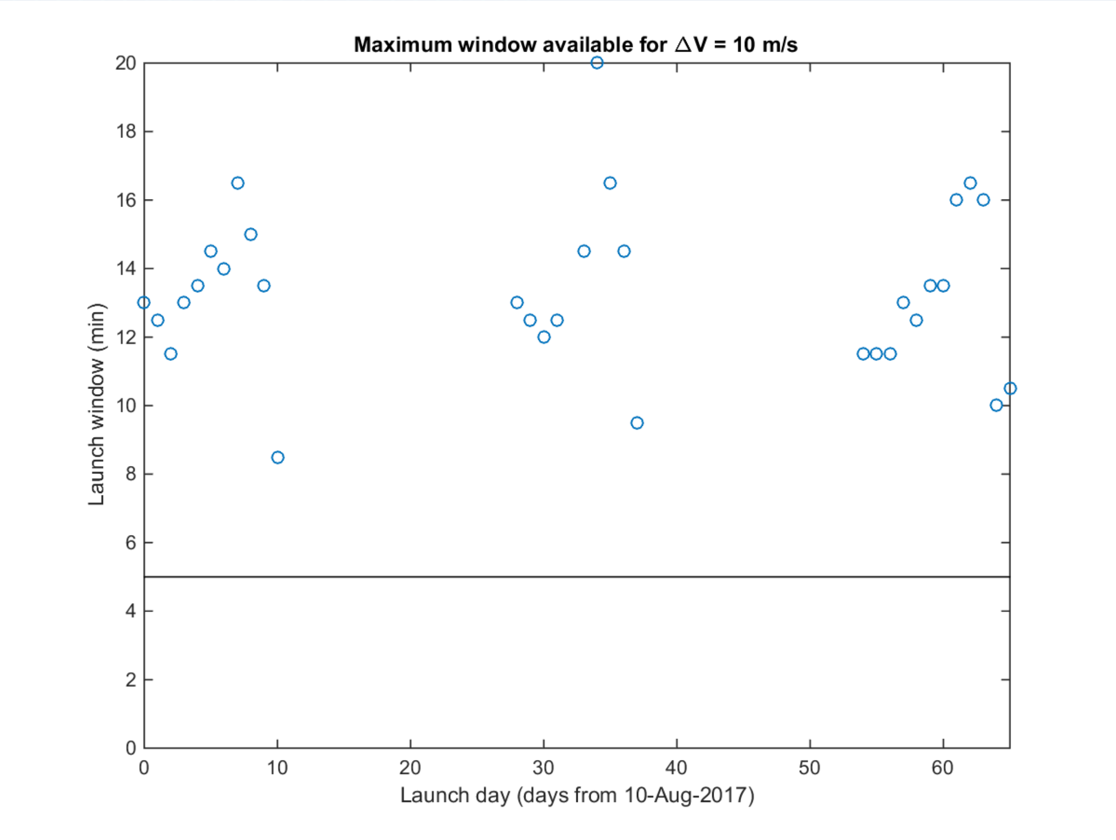
- With a ±5-minute window, the cost is much greater – prohibitive in most cases

Summary Results



- If we need a 10-minute window, we can allow it to be non-centered.
- All but 2 launch days work.

Summary Results



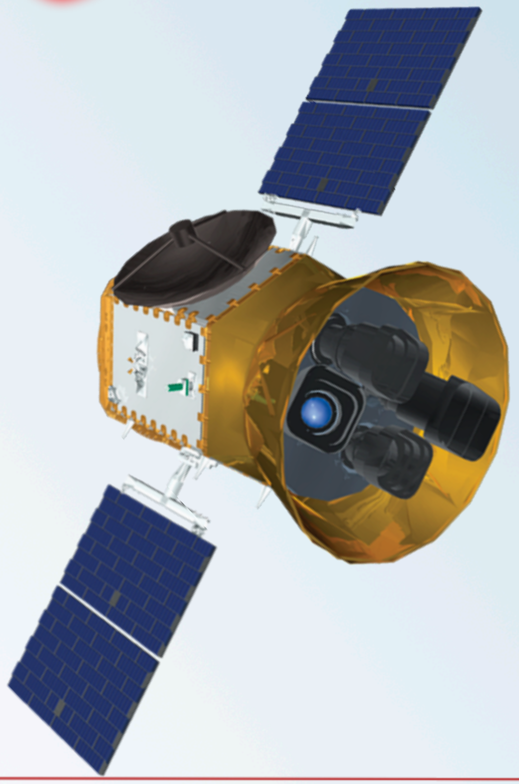
- Here, we look at the reverse question:
  - *What's the largest window we can achieve on each day for 10 m/s?*
- With this retargeting strategy, minimum window is 8 minutes

Conclusions

- Our launch time is dictated by a lunar encounter
  - *Can't expect a large launch window (hours)*
- Several retargeting strategies are possible
  - *Here, we've chosen one that results in a flyby close to nominal, and a mission orbit that remains 2:1 resonant*
  - *This is proof-of-concept work; more detailed analysis of specific cases is needed*
- Using the current strategy:
  - *Basic requirement of 5-minute total launch window is met*
  - *With 10 m/s, we can get at least 8 minutes*
  - *Minimum-dV window is normally not symmetric about dt=0.*
- Proposed requirements change (MRD_55):
  - *"The TESS Project shall provide for total launch windows of at least 5 minutes during each day of the launch period."*

Future Work

- Refine the retargeting strategy
  - *Limit other mission parameters as necessary (LRP/AOP)*
  - *Retarget through flyby to maximize value*
  - *Explore implications of reduced PAM via launch time offset*
- Achieved mission orbits need further analysis:
  - *Screen achieved mission orbits against other requirements (PAM sun angle, minimum perigee, etc.)*



12: Statistical Delta-V and Contingency

Don Dichmann

March 11, 2015

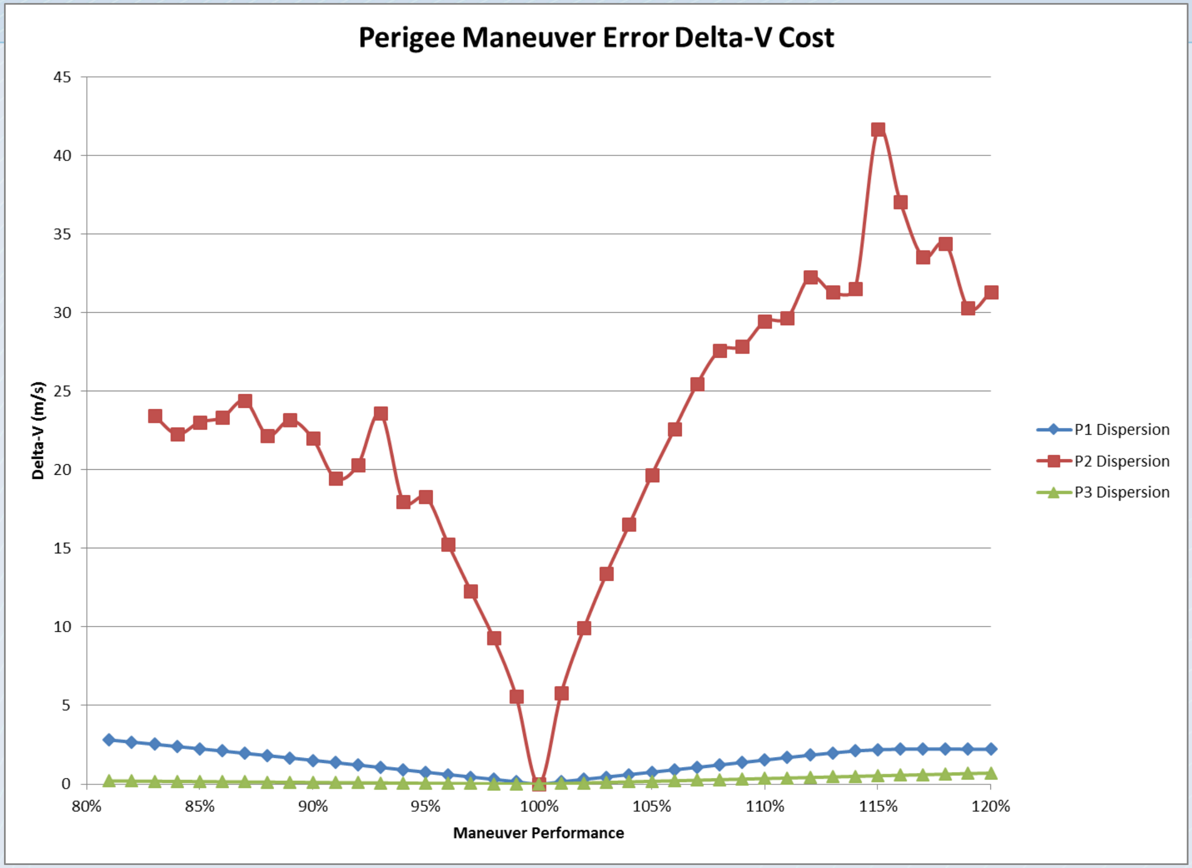
Overview

- Statistical Delta-V Budget represents the cost to correct for statistically likely errors:
  - *Maneuver execution error: 5% 3-sigma in magnitude*
  - *OD errors: analysis shows < 7% velocity error at perigee maneuvers*
- Contingencies represent unexpected errors such as a missed burn or a partial burn with error magnitude not statistically likely
- This analysis uses similar algorithms to correct for both kinds of errors
- Current Assumptions:
  - *We model only maneuver magnitude error, not pointing error (Pointing error has a smaller effect in general)*
  - *We model individual maneuver errors, then perform a statistical analysis to combine the results*
  - *In statistical analysis we assumed 10% 3-sigma maneuver execution error for P1, P3 maneuver, and 7% for P2 and PAM, to account for OD error.*
    - *For P1 and P3 we Root-Sum-Square (RSS) the 5% execution error with 7% OD error to get 8.6%, then round up to 10% to be conservative.*
    - *For P2 and PAM with smaller error we Root-Sum-Square (RSS) the 5% execution error with 2% OD error to get 5.3%, then round up to 7% to be conservative.*
- A Monte Carlo simulation is being developed to handle a more general class of errors

Delta-V Budget

Event	Planned ΔV (m/s)	Current ΔV (m/s) Aug 2017
A1	20	0–17
P1	35	31–50
P2	20	0-20
P3	5	0–8
Period Adjust Maneuver (PAM)	70	56–68
Deterministic Total	**150**	**109–131**
Launch Window Allowance	10	10
Launch Vehicle Dispersion	25	25–31
Trajectory Correction Maneuvers	10	15-26
Margin	20	22-55
Total	**215**	**215**

Delta-V cost for maneuver error: 8/10/17 launch



- For this 8/10/17 launch, the P2 error dominates. This is because we can only correct for P2 error in timing at P3, which is not very efficient
- For P1 we can correct timing at P2.
- For P3 (small) we can correct at TCM one day later
- For PAM we can correct at next perigee
- We did not yet model A1 error, which can be corrected at several subsequent maneuvers

Statistical Analysis

- For each P1 and P3 maneuvers we assumed a 10% 3-sigma error; for P2 and PAM we assumed 7% 3-sigma error
- We applied the normal distribution to find the mean and standard deviation for each maneuver
- We then combined the results to obtain the cumulative mean and standard deviation, assuming independence:
  - *Mean of sum is sum of means*
  - *Sigma of sum is RSS of sigma*
- Finally we compute the mean + 3*sigma value of 26.4 m/s to represent the cumulative statistical error for 8/10/17
- For this launch date, and for generally launches before 8/16/17, the P2 error dominates

8/10/2017	P1	P2	P3	PAM	total
mean	0.40	8.30	0.06	0.90	9.66
sigma	0.37	5.57	0.10	0.12	5.58
mean + 3*sig	1.50	25.00	0.35	1.25	**26.41**

Statistical Analysis (cont'd)

- Start of window

8/10/2017	P1	P2	P3	PAM	total
mean	0.40	8.30	0.06	0.90	9.66
sigma	0.37	5.57	0.10	0.12	5.58
mean + 3*sig	1.50	25.00	0.35	1.25	**26.41**

- Middle

8/15/2017	P1	P2	P3	PAM	total
mean	1.56	0.08	0.18	0.90	2.73
sigma	4.48	0.57	0.54	0.12	4.55
mean + 3*sig	15.00	1.80	1.80	1.25	**16.38**

- End

8/19/2017	P1	P2	P3	PAM	total
mean	0.71	0.63	0.20	0.90	2.44
sigma	4.10	0.29	0.27	0.12	4.12
mean + 3*sig	13.00	1.50	1.00	1.25	**14.80**

Contingency Analysis

- A contingency is an unlikely maneuver execution error: missed, partial, or delayed burn
- There is no longer a requirement that we be able to recover from any single maneuver error. It is also very difficult to plan for all contingencies
- Nevertheless we want to plan as far as practical for contingencies, and to assess whether there is sufficient DV available in the budget

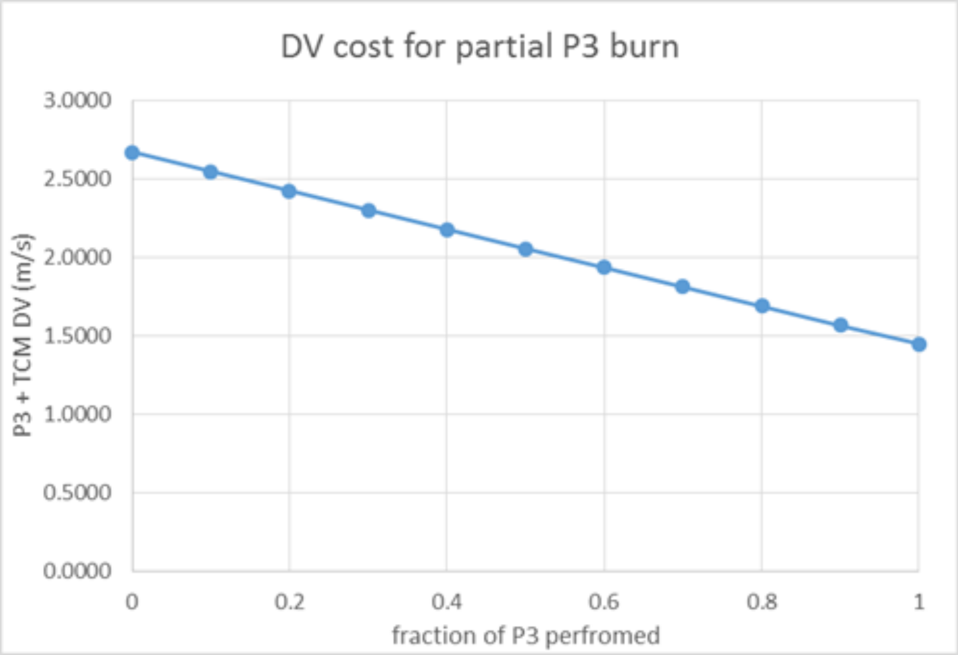
22-N Thruster is Needed

- There are four 5-N thrusters used for attitude control
  - *Discussed further in Finite Burn Modeling section*
- The 5 N thrusters are pointed nearly orthogonal to the 22-N thruster
- If the 22-N thruster were to fail, it would not be possible to perform orbit maneuvers using the 5-N thrusters only

P2 Maneuver is a Critical Burn

- Our simulations show that the P2 maneuver is a critical burn, at least for launch dates early in the monthly window
- We have found that correcting for a large P2 error can cost as much at 90 m/s
  - *The worst case occurs if we get enough (50-80%) of the burn to get close to the Moon on loop 4, but not enough to perform a successful flyby (see following charts)*
- The complete delta-V budget shows that we have at least 22 m/s margin for contingency, depending on the launch date. That is enough to recover from some contingencies, but not enough to correct for all P2 maneuver execution errors.

Missed/Partial P3 burn: 8/10/17 launch



- If P3 is missed/partial, we can perform a TCM one day later to correct.
- DV cost for a completely missed P3 is only about 1.5 m/s, since P3 is small by design.

Missed/Partial P2: 8/10/17 launch

- An off-nominal P2 burn is the most difficult to handle in general
- We need different strategies depending on whether
  1. *We get most of P2 (~90-110%)*
  2. *We get little of P2 (0-50%)*
  3. *We get some but not most of P2 (~50-90%). This is the most challenging case:*
     - *The 4th apogee is not high enough to perform the flyby, but apogee is high enough to be significantly perturbed by the Moon*
     - *We can find a strategy to recover*
     - *However there does not appear to be sufficient DV to recover*
- If we launch near lunar perigee, where the P2 maneuver is close to zero, then P2 is not longer a critical maneuver
  - *From this observation, we are looking to see if we could redesign phasing loops in early part of the window to make P2 less critical*

Backup charts

Contingency Analysis

- Missed/Partial burns
  - *Missed/Partial P1*
  - *Missed/Partial P3*
  - *Missed/Partial P2*
  - *Missed/Partial PAM*
- A1 burn
  - *Unless there is a failure of the thruster, because s/c is near A1 for about a day we expect A1 not to be missed, though perhaps delayed/partial*
  - *Because we get a lift in P1 perigee, A1 is not critical. If A1 is missed/partial, we can plan to make it up at A2.*
- Delayed burns
  - *Currently we have results from PDR based on 2-body approximations*
  - *P1, P2, P3 can be delayed for ~10 minutes with a cost of less than 10 m/s*
  - *PAM can be late/early by ~12 hours with cost of less than 10 m/s*
  - *Time permitting it should be possible to modify current scripts to model a delay in each burn*
- Emphasize that the Launch Window, Launch Dispersion and Missed/Partial burn analysis reuse code, since they all address the same kind of question: What If a maneuver is not performed as expected? How do I need to modify subsequent maneuvers to get back to the nominal maneuver schedule?

Missed/Partial P2 (cont'd): ~90-105%

- If we get most of P2 then we can arrive at perigee P3 at about the right time to still accomplish the flyby as planned
- There is sufficient DV to adjust the P3 burn (all 3 components) and the P3 epoch to achieve the nominal B-plane parameters

Missed/Partial P2: 0-~50%

- If we get little or none of the P2 burn then the timing for the flyby is too far off to achieve the flyby at the planned epoch.
- Instead we can wait one lunar cycle to set up for the flyby essentially as planned
- In this case we
  - *Do not perform P3 at all. This allows us to keep apogee 4 well below the Moon to avoid detrimental perturbations*
  - *We add loops 4, 5 and 6 to the time line, each with orbit period near 9 days. TLI is now at perigee 6*
  - *At perigee P4 we perform a burn to resize the loops so that*
    - *Epoch of perigee P6 = nominal epoch of P3 + 27.3 days*
  - *At perigee P6 we optimize the burn epoch and 3 components of P6 burn to achieve the nominal B-plane parameters*
- This approach takes about 40 m/s

Missed/Partial P2: ~50-90%

- As noted above, this is the most challenging case
- As in the previous case, we can wait one lunar cycle to set up for the flyby essentially as planned
- However if we performed no P3 burn, the orbit is significantly perturbed by the Moon and we may not be able to recover.
- Instead we perform a retrograde P3 burn to lower apogee to about 300,000 km to avoid the Moon
- Then as in previous case
  - *We add loops 4, 5 and 6 to the time line, each with orbit period near 9 days. TLI is now at perigee 6*
  - *At perigee P4 we perform a burn to resize the loops so that*
    - *Epoch of perigee P6 = nominal epoch of P3 + 27.3 days*
  - *At perigee P6 we optimize the burn epoch and 3 components of P6 burn to achieve the nominal B-plane parameters*
- This approach takes about 90 m/s, which is outside our budget
- Conclusion: P2 must be treated as a critical burn: If it is not performed as planned, we may not be able to recover the mission
- *IF the launch vehicle can deliver s/c to lunar distance, and if we change to 2.5 phasing loops, then P2 may not be critical.*

2013 IAA Planetary Defense Conference

Trajectory and Mission Design for the Origins Spectral Interpretation Resource Identification Security Regolith Explorer (OSIRIS-REx) Asteroid Sample Return Mission

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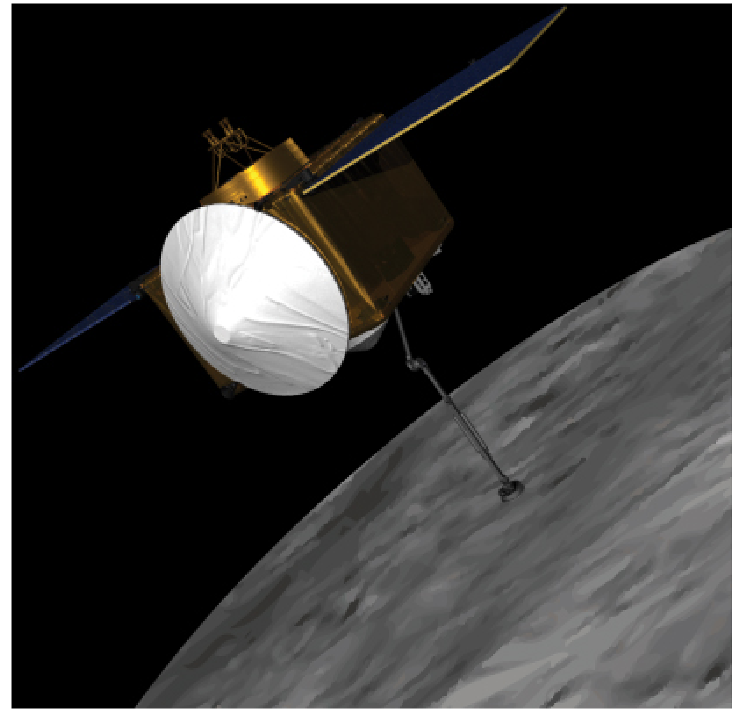
NASA/GSFC* / KinetX Aerospace, Inc.† / Lockheed Martin‡

April 16th, 2013

OSIRIS-REx Mission Overview

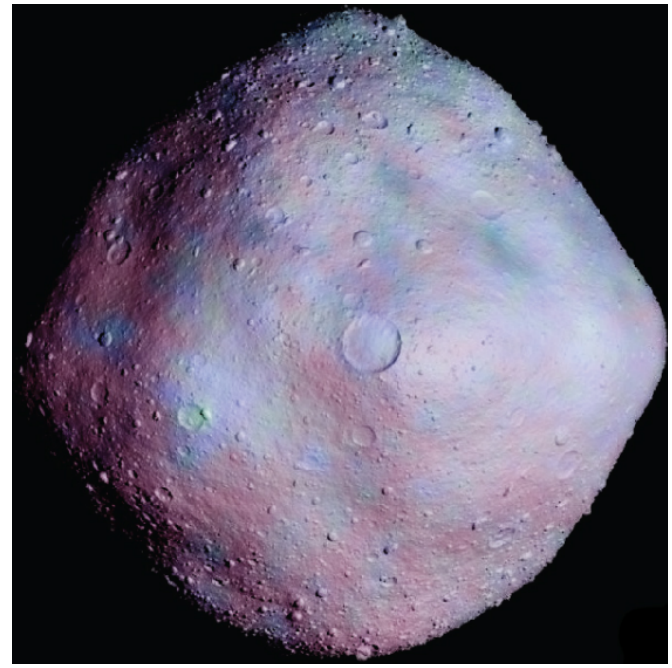
- **O**rigins **S**pectral **I**nterpretation **R**esource **I**dentification **S**ecurity **R**egolith **Ex**plorer (OSIRIS-REx) is the third mission selected as part of NASA's New Frontiers Program.
- Launch in September of 2016, encountering near-Earth asteroid (NEA) 101955 (1999 RQ_{36}) in October of 2018.
- Study 1999 RQ_{36} for up to 505 days, globally mapping the surface from a distance of 5 km to a distance of 0.5 km.
- Obtain at least 60 g of pristine regolith and a surface material sample.
- Return the Stardust-heritage Sample Return Capsule (SRC) to Earth in September of 2023.
- Deliver samples to the NASA Johnson Space Center (JSC) curation facility for world-wide distribution.



Credit: NASA/GSFC/UA

Destination: 1999 RQ_{36}

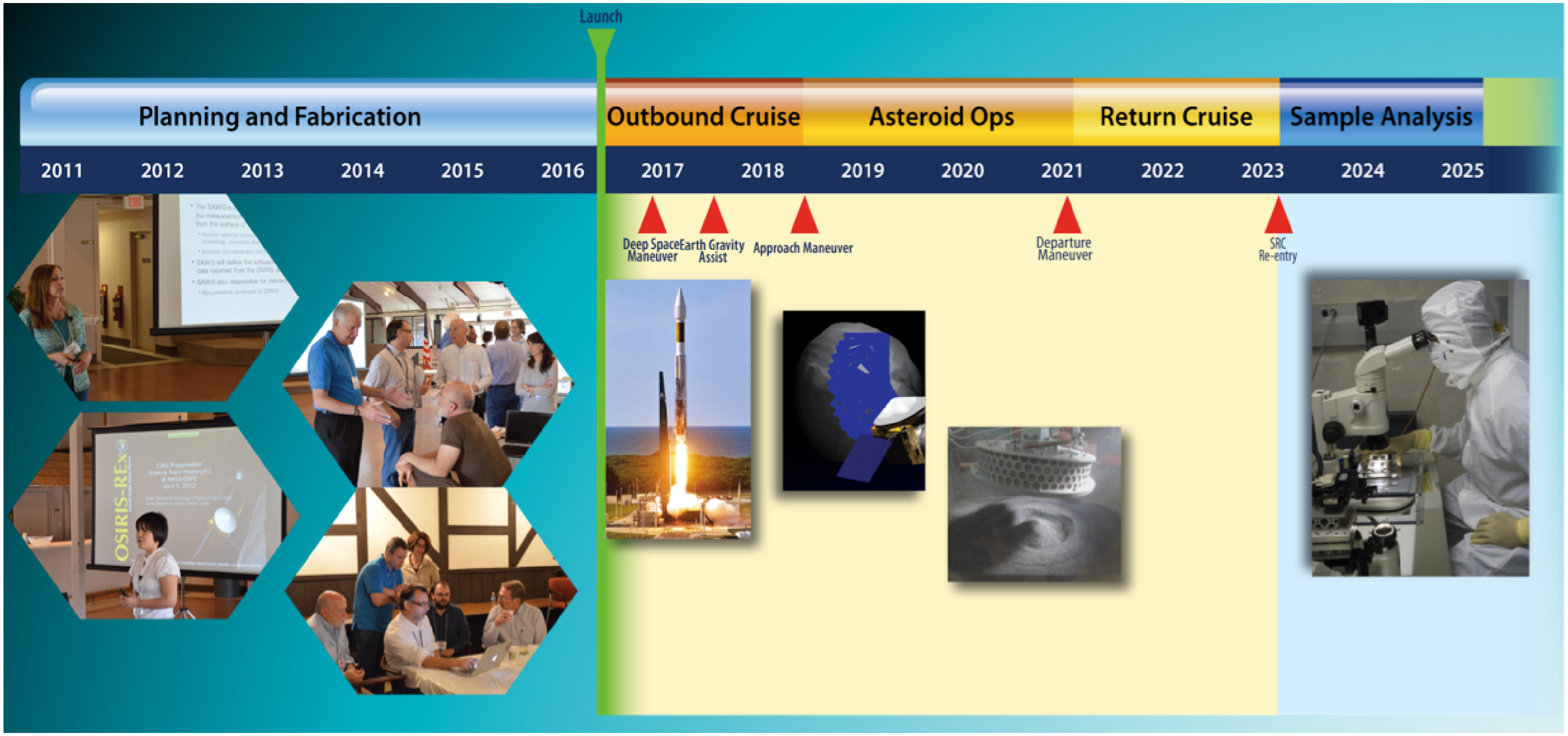
- 101955 (1999 RQ_{36}) is one of 1391 currently known Potentially Hazardous Asteroids (PHAs).
- One of the most hazardous of the PHAs based on its probability of future Earth collision and impact energy of approximately 2700 MT.
- A member of the rare B-type subgroup of the carbonaceous C-type asteroids
- Its relatively low-inclination, Earth-like orbit makes it accessible to spacecraft missions.
- One of the best characterized NEAs due to the significant number of optical and radar observations collected since discovery in 1999
- Approximate diameter of 550 m.
- Retrograde rotation (obliquity of $174° \pm 10°$) with a period of 4.2978 hours; no non-principal axis rotation detected thus far.
- Low, nominal, and high estimates for its gravitational parameter are 2.93×10^{-9}, 4.16×10^{-9}, and 6.6249×10^{-9} km^3/s^2, respectively (from radar-derived shape models and constraints on bulk density).



Simulated image of 1999 RQ_{36} - topography overlaid on radar imagery.
Credit: NASA/GSFC/UA

Orbital Element	Value
Semi-major axis, a (AU)	1.12600
Eccentricity, e	0.20373
Inclination, i	6.03491°
Longitude of Ascending Node, Ω	2.04227°
Argument of Perihelion, ω	66.2686°
Mean Anomaly at Epoch, M	72.8280°

Mission Timeline



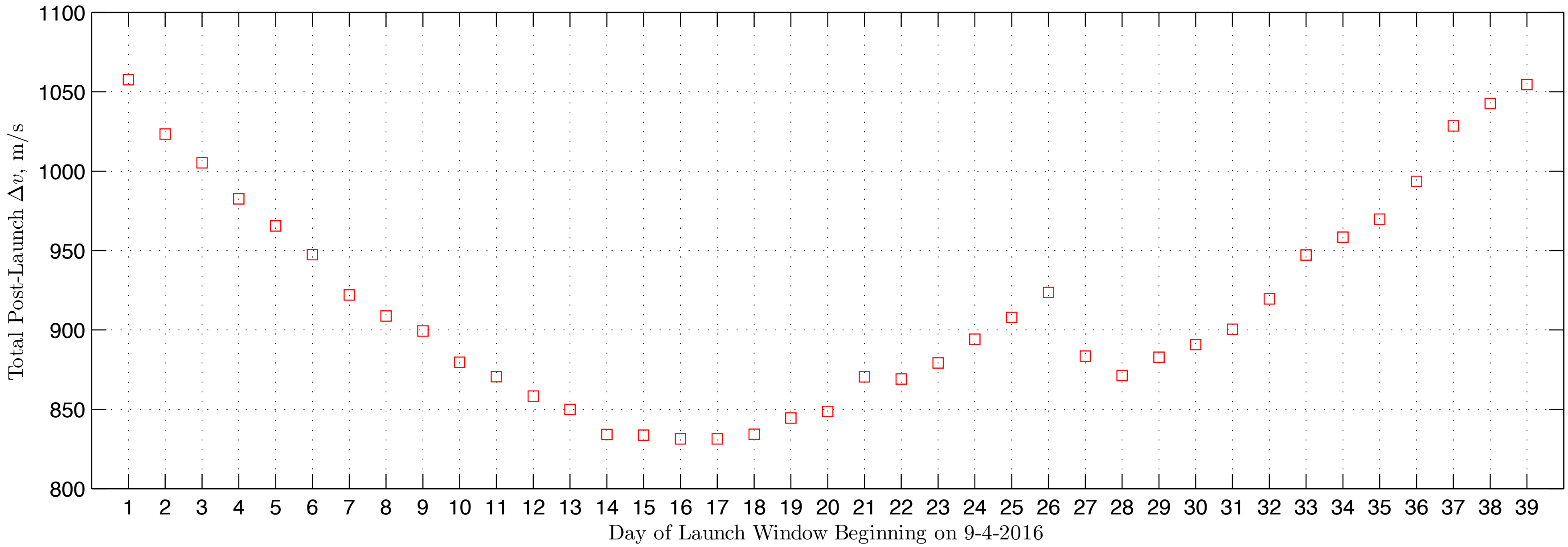
Outbound Cruise Trajectory Optimization

- The primary and backup launch windows are defined by computing the optimal (minimum post-launch Δv) outbound trajectory sequence for each day of the launch windows.
- The total post-launch Δv is the sum of the magnitudes of the DSM1, DSM2, AAM1, AAM2, and AAM3 maneuvers.
- The optimization is performed by holding C_3 constant and varying the following parameters on each launch day:
  - The DLA and RLA.
  - The times, orientations, and magnitudes of DSM1, DSM2, and the orientations and magnitudes of AAM1, AAM2, and AAM3.
  - The time, orientation, and altitude of the EGA.
- For the backup launch window cases there is only one DSM (DSM1) and C_3 is also varied.

Asteroid Arrival Sequence Overview

- Asteroid arrival conditions are purposely standardized such the same arrival sequence will be executed regardless of which launch window and day of launch window are utilized.
  - Single set of arrival circumstances for which the spacecraft must be designed.
  - Favorable natural illumination of the asteroid from the spacecraft's point of view.
- The AAM is divided into 3 parts to create a gentle and robust approach.
  - Adequate time to optically acquire the asteroid during approach.
  - Adequate time for natural satellite survey.
  - Gracefully recover if the first AAM is not executed.
- AAM1 is performed on 2018-10-01, targeting arrival at a location 6300 km from the asteroid 14 days later on 2018-10-15; that is the same location relative to the asteroid that is targeted by DSM2.
  - Thus, if AAM1 is not executed, the spacecraft simply arrives at that same location early, on 2018-10-05.
- AAM2 is nominally performed on 2018-10-15, targeting arrival at a location 270 km from the asteroid 14 days later on 2018-10-29.
- AAM3 is nominally performed on 2018-10-29, targeting arrival at a location 19.3 km from the asteroid on 2018-11-12.

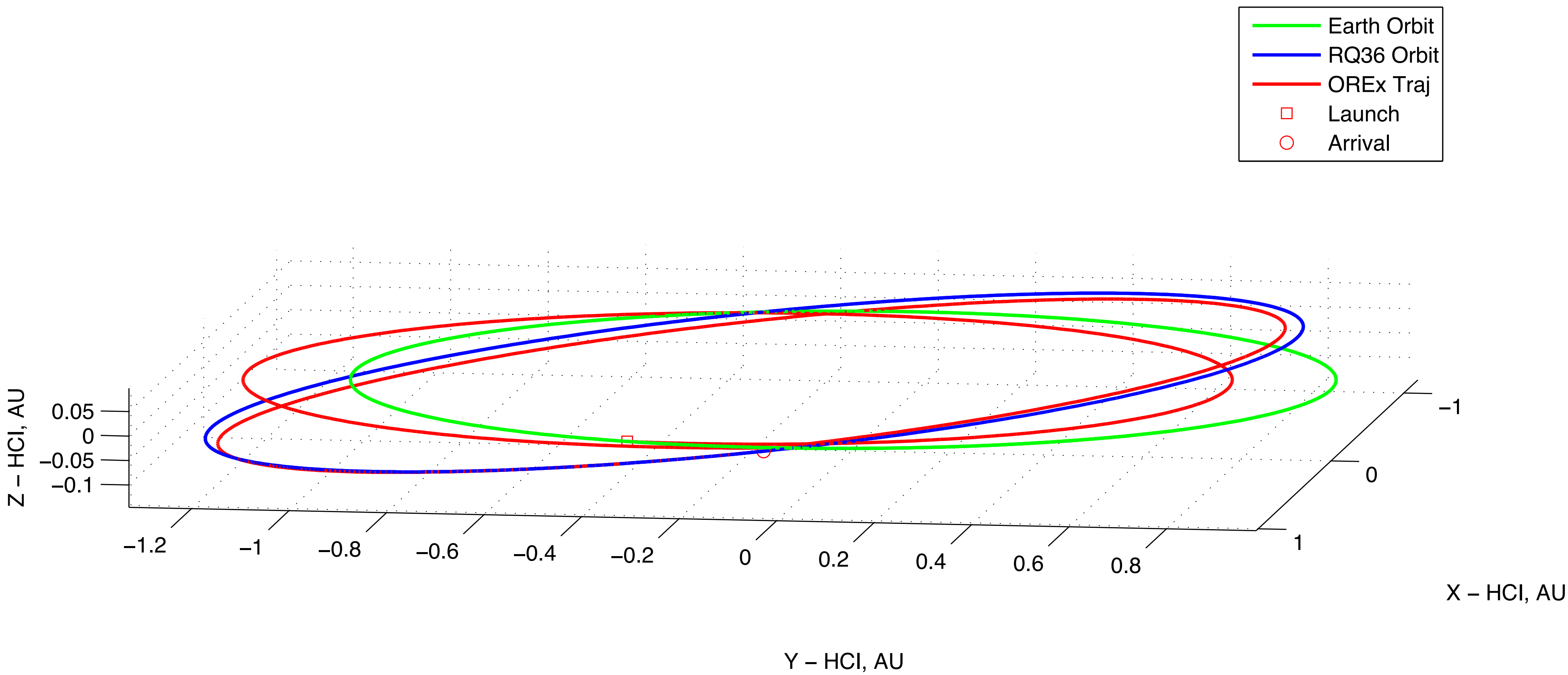
Primary Launch Window Results



Total Post-Launch Δv variation throughout the primary launch window.

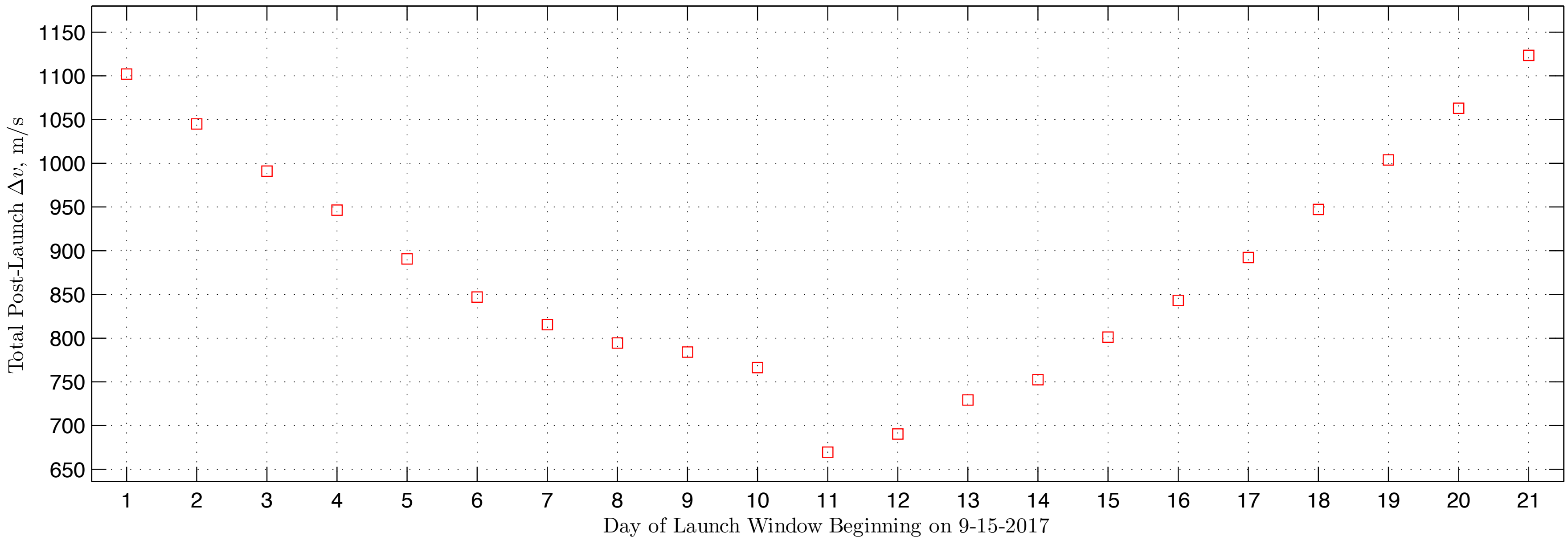
- The DLA is within the range of $-9°$ to $+3°$ throughout the primary launch window.
- The C_3 is kept constant at 29.3 km^2/s^2 throughout.
- Total post-launch Δv reaches a minimum of 831.3 m/s on days 16 and 17.
- The higher Δv at the extremes of the 39 day launch window are feasible, but the launch window could be restricted to the middle 21 days if needed to reduce Δv requirements.
- The discontinuity in post-launch Δv between days 26 and 27 is due to a relatively close lunar encounter during Earth departure.

Example Primary Launch Window Trajectory



OSIRIS-REx primary launch window outbound cruise trajectory to 1999 RQ_{36}.

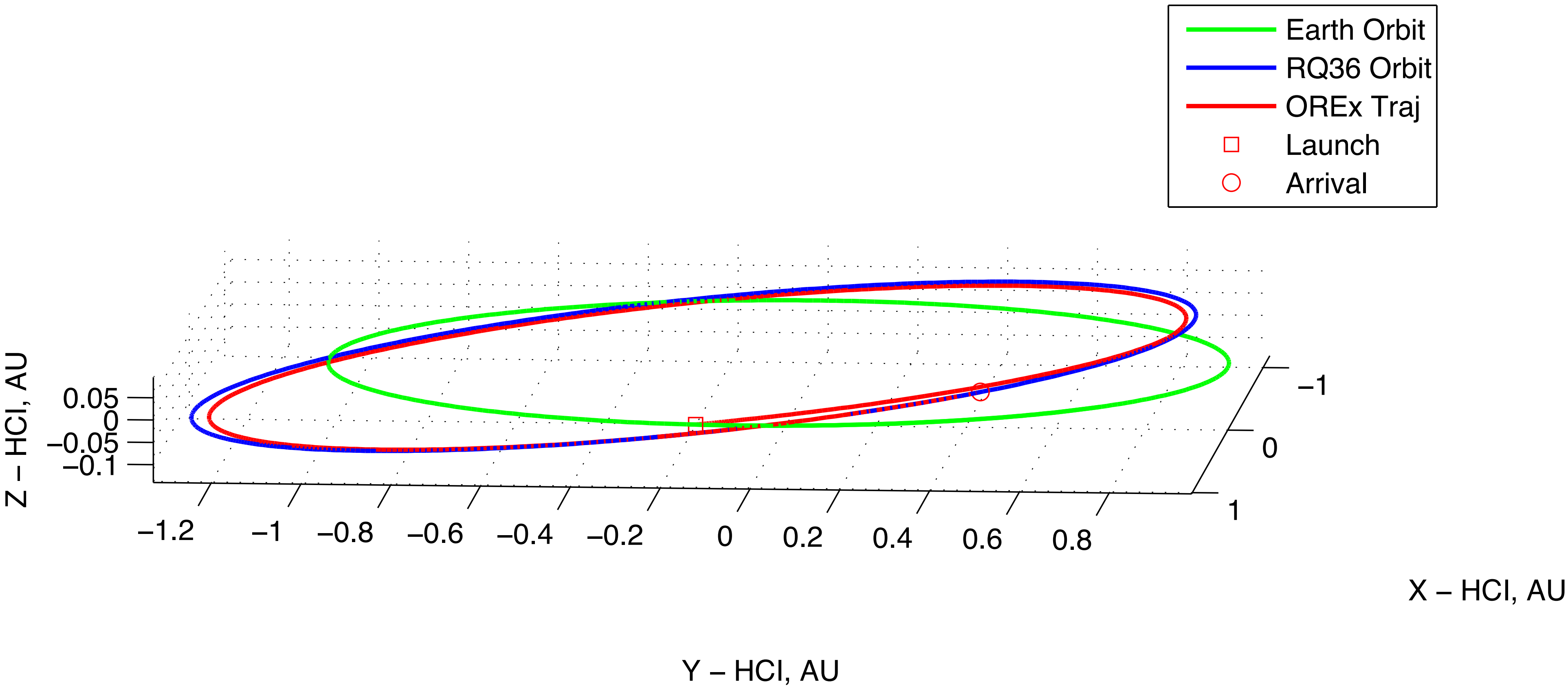
Backup Launch Window Results



Total Post-Launch Δv variation throughout the backup launch window.

- The DLA is within the range of 31° to 36° throughout the backup launch window.
- The C_3 is within the range of 28.5 to 29.3 km^2/s^2 throughout.
- Total post-launch Δv reaches a minimum of 669.6 m/s on day 11.
- Reduced launch vehicle performance is possible (for launches from KSC) because all of the backup launch window DLA values are outside the range of ±28.5°.
- However, the optimization strategy will maintain a constant launch vehicle payload mass of 1955 kg by adjusting C_3 as needed on each day of the launch window.

Example Backup Launch Window Trajectory



OSIRIS-REx backup launch window outbound cruise trajectory to 1999 RQ_{36}.

Remarks About Launch Windows

- The highest Δv value in the backup launch window (1123.43 m/s) exceeds the highest Δv in the primary launch window (1057.57 m/s).
- From that perspective, additional Δv margin is available if the mission launches during the primary launch window.
- However, when considering only individual days within the launch windows, we note that the overall minimum Δv in the backup launch window (669.6 m/s) is actually less than the overall minimum Δv in the primary launch window (831.3 m/s).
- The primary launch window is nearly twice as wide as the backup launch window.
- The backup launch window reduces mission complexity by forgoing the EGA and one DSM.

Trajectory Type Comparison

Comparison of trajectory design types for the OSIRIS-REx mission.

		Primary LW (DSMs, multi-rev, EGA)		Backup LW (DSM, multi-rev)	
	Type II Lambert	Best	Worst	Best	Worst
Earth Departure Date	09/27/2017	09/19/2016	09/04/2016	09/25/2017	10/05/2017
Earth Departure DLA	33.36°	0.11°	2.13°	33.09°	33.25°
Earth Departure C_3 (km^2/s^2)	29.0	29.3	29.3	29.1	29.0
Flight Time to RQ_{36} (days)	382	784	799	413	403
NEA Arrival Date	10/14/2018	11/12/2018	11/12/2018	11/12/2018	11/12/2018
Total Post-launch Δv to Arrive at RQ_{36} (m/s)	874	831	1058	670	1123
Stay Time at RQ_{36} (days)	1387	842	842	842	842
RQ_{36} Departure Date	08/01/2022	03/03/2021	03/03/2021	03/03/2021	03/03/2021
RQ_{36} Departure Δv (m/s)	494	320	320	320	320
Flight Time to Earth (days)	422	935	935	935	935
Earth Arrival Date	09/27/2023	09/24/2023	09/24/2023	09/24/2023	09/24/2023
Atmospheric Entry Speed (km/s)	12.88	12.20	12.20	12.20	12.20
Total Post-launch Round-Trip Δv (m/s)	1368	1151	1378	990	1443
Total Round-Trip Mission Duration (years)	6.00	7.01	7.05	6.00	5.97

- Late September of 2017 is an optimal time to depart Earth for asteroid rendezvous because Earth happens to be near the line of intersection between the orbit planes.
- The advanced trajectory solutions of the primary and backup launch windows trade some time at the asteroid for the benefit of reducing the mission Δv.
- The more advanced trajectory solutions also provide larger Earth departure and asteroid departure windows and better manage Earth return atmospheric entry speed.
- The straightforward Type II Lambert optimal total mission Δv is a reasonable predictor of the amount of Δv required by the more advanced methods.

Conclusions

- The OSIRIS-REx team is developing a robust set of designs that will ensure successful return of a pristine regolith sample from the potentially hazardous near-Earth asteroid 1999 RQ_{36}.
- Advanced trajectory design techniques including multi-rev trajectories, optimized DSMs, and an Earth Gravity Assist are employed to trade time at the asteroid and mission complexity for reduced Δv requirements and wider, more robust launch windows.
- The Earth Gravity Assist enables launch in 2016, a full year earlier than would be possible otherwise, and provides a wide primary launch window.
- The backup launch window in 2017 provides a viable alternative if needed.
- 1999 RQ_{36} is an exciting science target and our interactions with it will provide crucial knowledge for future missions to asteroids for robotic and human exploration, scientic understanding, and defending our planet against asteroid impacts.

Acknowledgments

This work is supported by the OSIRIS-REx Asteroid Sample Return Mission project, for which **Dr. Dante Lauretta** is the Principal Investigator, within NASA's New Frontiers Program. The authors are grateful for contributions to this work by the OSIRIS-REx Flight Dynamics team and other OSIRIS-REx subsystem teams. The authors are also appreciative of support and technical input from the OSIRIS-REx Science team liaisons to the Flight Dynamics team, **Dr. Steven Chesley** and **Dr. Daniel Scheeres**.

http://osiris-rex.lpl.arizona.edu/

Appendices

Overview of Mission Phases

- Outbound Cruise
  - Earth Gravity Assist (EGA) following launch during the Primary Launch Window is bracketed by two deterministic Deep Space Maneuvers (DSMs), DSM1 and DSM2.
  - Backup Launch Window trajectories involve only one DSM (labeled DSM1, occurring between launch and asteroid arrival) and no EGA.
- Approach
  - Three deterministic Asteroid Arrival Maneuvers (AAMs): AAM1–AAM3.
  - Search vicinity of asteroid for natural satellites $>$ 10 cm in size.
- Preliminary & Detailed Survey
  - Preliminary: Three slow ($\sim$ 20 cm/s) flybys of asteroid to within 7 km.
  - Detailed: Observations collected from specific solar phase angle stations.
  - Estimate improved values of asteroid physical characteristics that affect subsequent proximity operations (spin state, gravitational parameter, gravity field coefficients).

Overview of Mission Phases

- Orbital Phase
  - Spacecraft enters into gravitationally captured orbit about the asteroid.
  - Terminator plane orbits (for stability relative to solar pressure).
  - Orbit A: 2018-12-31 to 2019-01-20, nominal radius is 1.5 km (50 hour period).
  - Orbit B: 2019-01-21 to 2019-03-05, nominal radius is 1.0 km (27 hour period).
  - Candidate sampling sites are selected during orbital phase.
- Recon Phase
  - Obtain more detailed observations of candidate sampling sites.
  - Flybys reaching an approach distance of 225 m are performed in a prograde sense across sunlit side of asteroid, departing from and returning to terminator plane orbit.
- TAG Rehearsals and TAG
  - Touch And Go (TAG) rehearsals begin two weeks after final recon flyby.
  - Three TAG rehearsals are performed prior to the actual TAG.
  - During TAG the spacecraft is guided to contact the asteroid's surface with a vertical speed of 10 cm/s, regolith is ingested by the sampling mechanism, and a 0.5 m/s escape maneuver is performed to move up and away from the asteroid.
- Asteroid Departure and Earth Return
  - Nominal asteroid departure occurs on 2021-03-03 and delivers the SRC to Earth 935 days later on 2023-09-24 with an atmospheric entry speed of 12.198 km/s.

Inbound Trajectory Overview

- Round-trip mission duration approximately equal to one Earth/asteroid synodic period effectively decouples the outbound and inbound trajectories such that they can be optimized independently.
  - The same inbound trajectory may be flown regardless of which launch window and day within launch window are utilized for the outbound cruise.
- A continuum of asteroid departure opportunities is available:
  - 2021-03-03: $\Delta v = 316$ m/s, 2023-09-24 Earth return, entry speed = 12.198 km/s
  - 2021-05-22: $\Delta v = 250$ m/s, 2023-09-25 Earth return, entry speed = 12.390 km/s
  - 2021-06-28: $\Delta v = 313$ m/s, 2023-09-27 Earth return, entry speed = 12.385 km/s
- There is a small chance of early departure from the asteroid on 2020-01-03 with departure Δv of 935 m/s, early Earth return on 2022-09-24, and entry speed of 12.24 km/s. Early return is only an option if a number of criteria are all met.
  - Spacecraft dry mass must not grow by more than a very small amount between now and launch.
  - Launch must occur during the middle 21 days of the 39 day primary launch window.
  - AAM1 must occur on or after 2018-10-01.
  - Science observations and sample collection must be complete within 460 days or less after AAM1.

Earth Return Overview

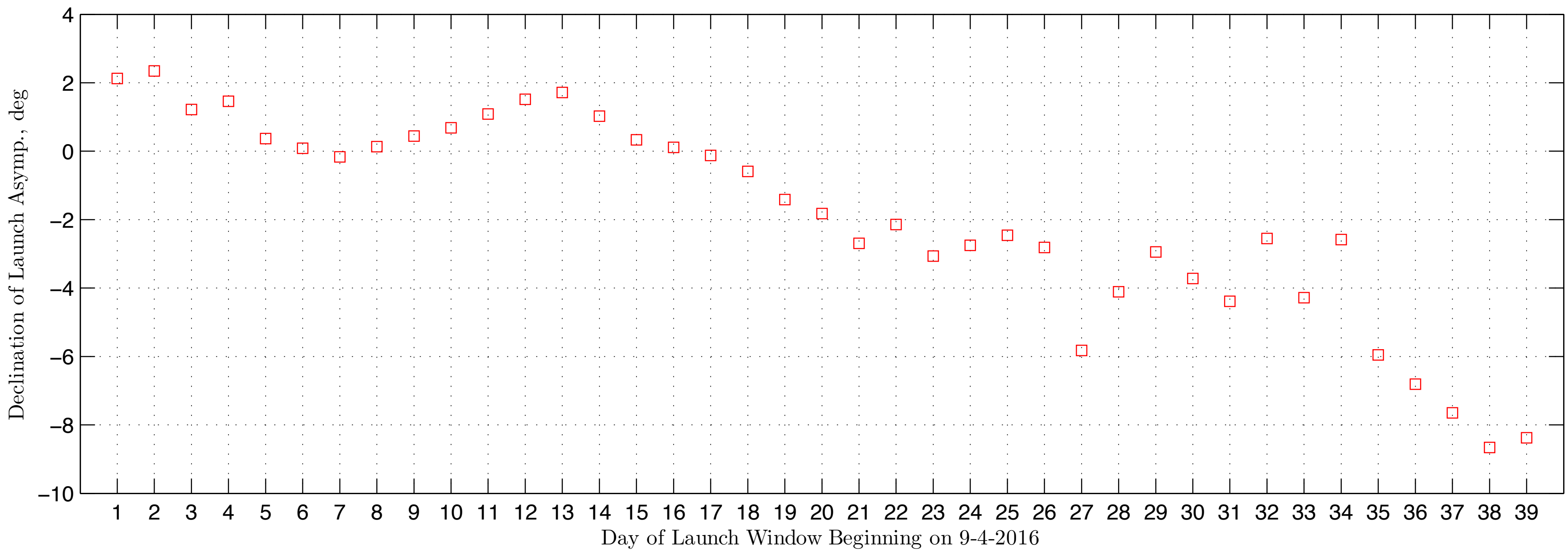
- Asteroid departure maneuver initially targets an Earth flyby at a distance of at least 10000 km.
- A series of planned "walk-in" maneuvers (total Δv of 4 m/s) are used to gradually lower perigee altitude, following Stardust mission heritage.
- After achieving appropriate entry trajectory, the SRC separates to continue on the entry trajectory while the OSIRIS-REx spacecraft performs a 17 m/s Δv to raise perigee and comply with the Planetary Protection requirement that the spacecraft reside in a solar orbit that will not approach any closer than 250 km to the Earth, Moon, or other solar system body.
- Final OSIRIS-REx spacecraft orbit has a perihelion distance of 0.5 AU, aphelion distance of 1.0 AU, and a period of 0.66 years.
- SRC entry conditions are defined by a 6503.14 km atmospheric entry interface radius, a 12.2 km/s nominal entry speed, and an inertial entry flight path angle of -8.2°.
- The entry trajectory is targeted to deliver the SRC to the Utah Test and Training Range (UTTR) for retrieval after landing.

Example Mission Δv Budget

- This example assumes launch on the most demanding day of the primary launch window.

Maneuver/Event	Pre-Event Mass (kg)	Main Δv (m/s)	Monoprop Mass (kg)	Pulse Mode Δv (m/s)	Pulse Prop Mass (kg)	Post-Event Mass (kg)
Post Launch - Initial Acquisition	1955.0	0.0	0.0	1.0	1.4	1953.6
Post Launch TCMs	1953.6	52.0	44.4			1909.3
DSM1	1909.3	472.6	359.9			1549.3
Outbound Cruise ACS Desat	1549.3				4.0	1545.3
DSM2	1545.3	49.0	33.1			1512.2
AAM1	1512.2	375.8	231.5			1280.7
AAM2	1280.7	150.9	82.6			1198.1
AAM3	1198.1	4.7	2.5			1195.6
Preliminary Survey	1195.6			1.3	1.1	1194.5
Detailed Survey	1194.5			1.3	1.0	1193.5
Orbit Operations	1193.5			0.1	0.1	1193.4
Surface Reconnaissance	1193.4			1.0	0.8	1192.6
Sampling Rehearsals	1192.6			1.6	1.3	1191.2
Sampling Operations	1191.2			1.2	1.0	1190.3
Δv to Repeat Rehearsals and Sampling Twice	1190.3			5.6	4.6	1185.7
10 Orbit Departures and Recaptures	1185.7			10.0	8.2	1177.5
Proximity Operations ACS Desat	1177.5				4.2	1173.3
1999 RQ_{36} Departure & Earth Targeting	1173.3	320.1	154.8			1018.5
Inbound Cruise TCMs	1018.5	10.0	5.2			1013.3
Earth Return Cruise ACS Desat	1013.3				4.6	1008.7
Deflection from Earth (after sep of 50 kg SRC)	958.7	17.5	7.4			951.4
Unallocated Δv Margin	951.4	73.0	30.2	22.0	13.9	907.2
Final Totals	891.9	1525.6	951.6	45.0	46.2	

Primary Launch Window DLA

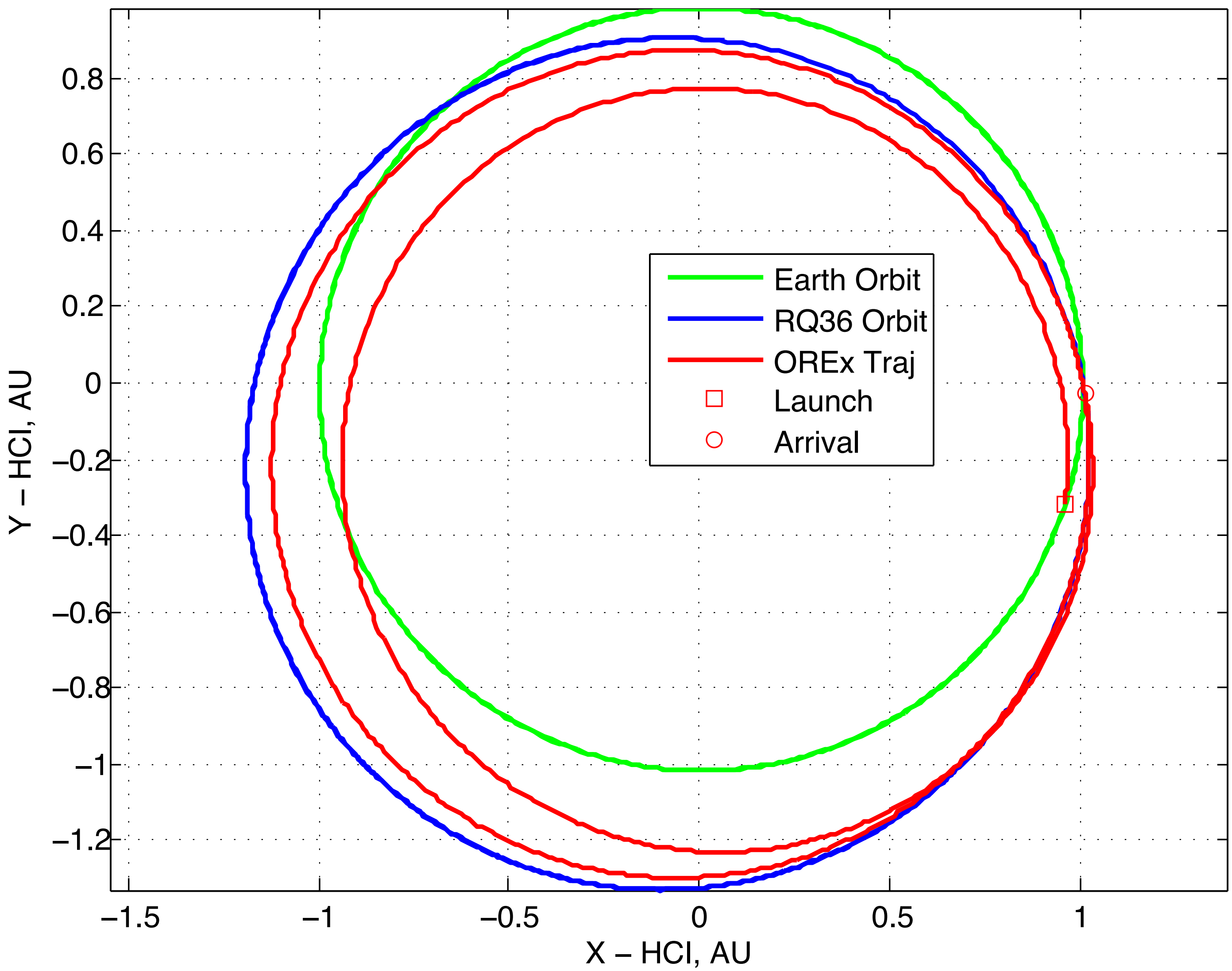


DLA variation throughout the primary launch window.

Primary Launch Window Details

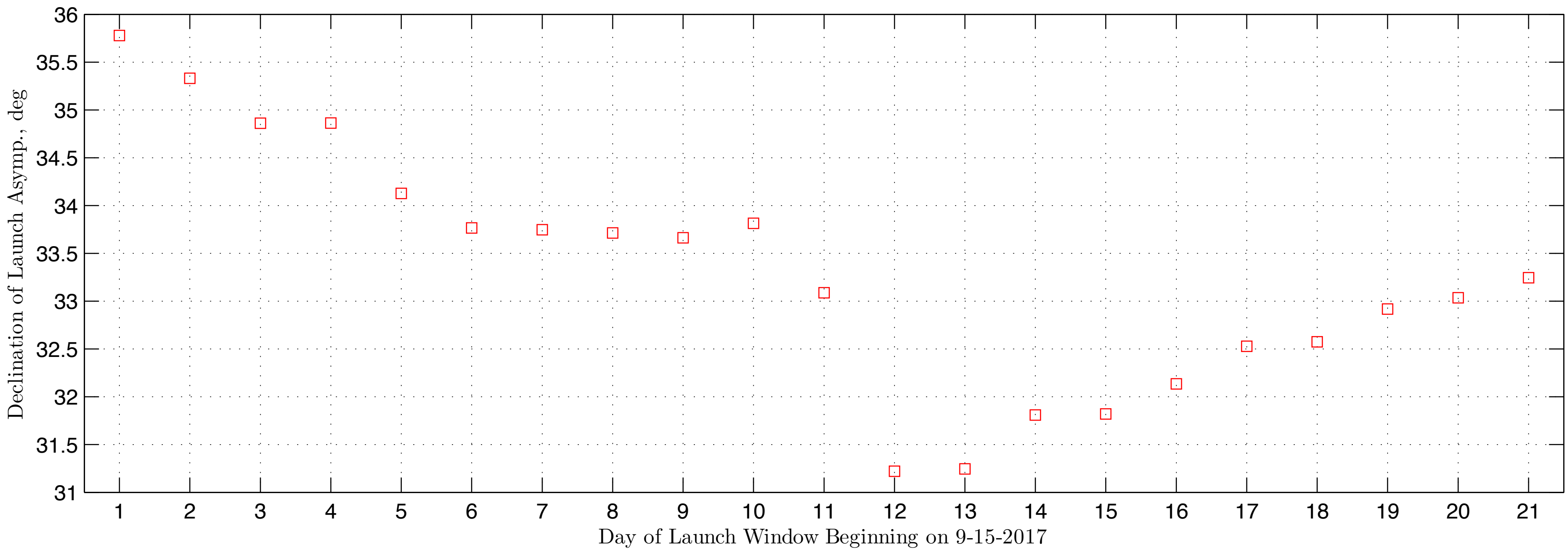
Day	Date	C_3 (km^2/s^2)	DLA	RLA	DSM1 (m/s)	DSM2 (m/s)	AAM1 (m/s)	AAM2 (m/s)	AAM3 (m/s)	Total Δv (m/s)
1	9-4-2016	29.3	2.13°	173.31°	530.03	7.05	365.80	149.94	4.77	1057.57
2	9-5-2016	29.3	2.34°	174.28°	505.47	1.09	363.19	148.84	4.77	1023.37
3	9-6-2016	29.3	1.22°	174.74°	488.91	0.89	362.17	148.55	4.71	1005.24
4	9-7-2016	29.3	1.46°	175.78°	465.46	0.30	363.19	148.84	4.77	982.56
5	9-8-2016	29.3	0.37°	176.16°	448.16	1.32	362.98	148.30	4.71	965.48
6	9-9-2016	29.3	0.09°	176.93°	425.33	1.86	365.75	149.68	4.77	947.39
7	9-10-2016	29.3	-0.16°	177.66°	406.52	0.93	366.13	143.66	4.71	921.96
8	9-11-2016	29.3	0.13°	178.69°	385.97	13.31	358.00	146.61	4.77	908.65
9	9-12-2016	29.3	0.44°	179.73°	370.80	28.23	351.35	144.15	4.71	899.24
10	9-13-2016	29.3	0.69°	180.76°	356.38	25.04	350.49	142.93	4.71	879.55
11	9-14-2016	29.3	1.09°	181.83°	337.51	30.93	353.39	144.01	4.71	870.55
12	9-15-2016	29.3	1.52°	183.07°	331.28	17.00	359.62	145.62	4.71	858.23
13	9-16-2016	29.3	1.72°	184.04°	315.52	24.17	359.90	145.50	4.71	849.80
14	9-17-2016	29.3	1.02°	184.62°	297.06	26.98	359.90	145.50	4.71	834.16
15	9-18-2016	29.3	0.33°	185.21°	280.05	27.94	372.02	148.97	4.71	833.69
16	9-19-2016	29.3	0.11°	186.03°	265.07	30.84	379.31	151.37	4.71	831.30
17	9-20-2016	29.3	-0.12°	186.82°	248.63	32.13	390.75	155.06	4.71	831.30
18	9-21-2016	29.3	-0.59°	187.48°	231.35	38.89	400.64	158.59	4.71	834.18
19	9-22-2016	29.3	-1.41°	187.82°	201.43	36.83	432.48	169.02	4.71	844.49
20	9-23-2016	29.3	-1.82°	188.48°	184.43	48.45	438.99	172.07	4.71	848.64
21	9-24-2016	29.3	-2.69°	188.83°	156.13	50.55	474.64	184.34	4.71	870.38
22	9-25-2016	29.3	-2.14°	189.96°	141.92	55.97	479.49	186.98	4.71	869.06
23	9-26-2016	29.3	-3.07°	190.39°	118.14	59.25	501.52	195.52	4.71	879.14
24	9-27-2016	29.3	-2.75°	191.25°	93.83	68.76	523.07	203.75	4.71	894.13
25	9-28-2016	29.3	-2.46°	192.12°	64.93	71.33	552.13	214.68	4.71	907.79
26	9-29-2016	29.3	-2.81°	192.71°	36.32	78.40	578.98	225.12	4.71	923.54
27	9-30-2016	29.3	-5.82°	192.16°	184.44	19.35	486.67	188.19	4.78	883.43
28	10-1-2016	29.3	-4.11°	193.11°	232.00	34.49	431.40	168.62	4.71	871.22
29	10-2-2016	29.3	-2.94°	194.92°	263.39	44.92	408.56	161.11	4.71	882.70
30	10-3-2016	29.3	-3.72°	195.64°	264.57	53.69	406.64	161.16	4.71	890.77
31	10-4-2016	29.3	-4.39°	196.31°	291.09	46.75	399.41	158.37	4.71	900.34
32	10-5-2016	29.3	-2.55°	198.04°	319.11	47.10	392.45	156.07	4.71	919.44
33	10-6-2016	29.3	-4.28°	198.16°	321.45	53.17	407.12	160.65	4.71	947.11
34	10-7-2016	29.3	-2.58°	199.93°	369.32	50.39	381.33	152.63	4.71	958.37
35	10-8-2016	29.3	-5.95°	199.46°	379.43	52.71	380.39	152.43	4.71	969.66
36	10-9-2016	29.3	-6.81°	200.11°	413.02	56.78	369.60	149.38	4.71	993.50
37	10-10-2016	29.3	-7.65°	200.77°	443.61	73.92	359.41	146.79	4.71	1028.44
38	10-11-2016	29.3	-8.66°	201.16°	433.26	55.99	392.48	156.03	4.71	1042.48
39	10-12-2016	29.3	-8.38°	202.17°	474.65	49.84	374.49	150.90	4.71	1054.60

Example Primary Launch Window Trajectory



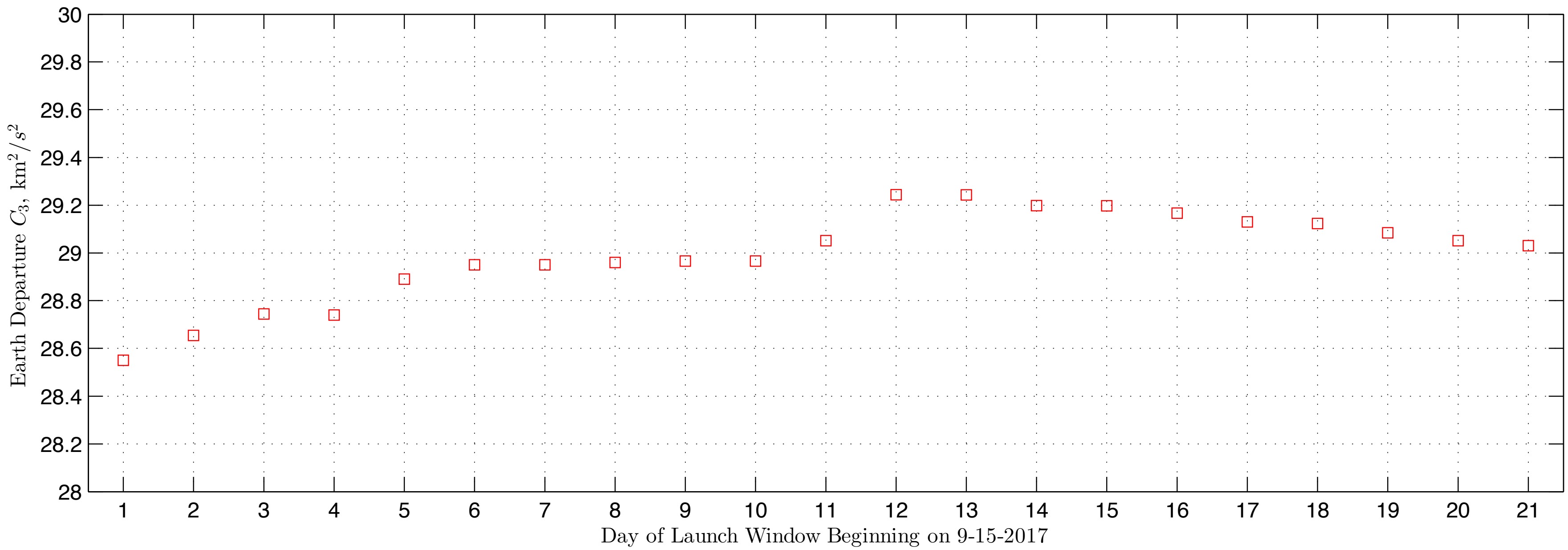
OSIRIS-REx primary launch window outbound cruise trajectory to 1999 RQ_{36}, ecliptic plane projection.

Backup Launch Window DLA



DLA variation throughout the backup launch window.

Backup Launch Window C_3

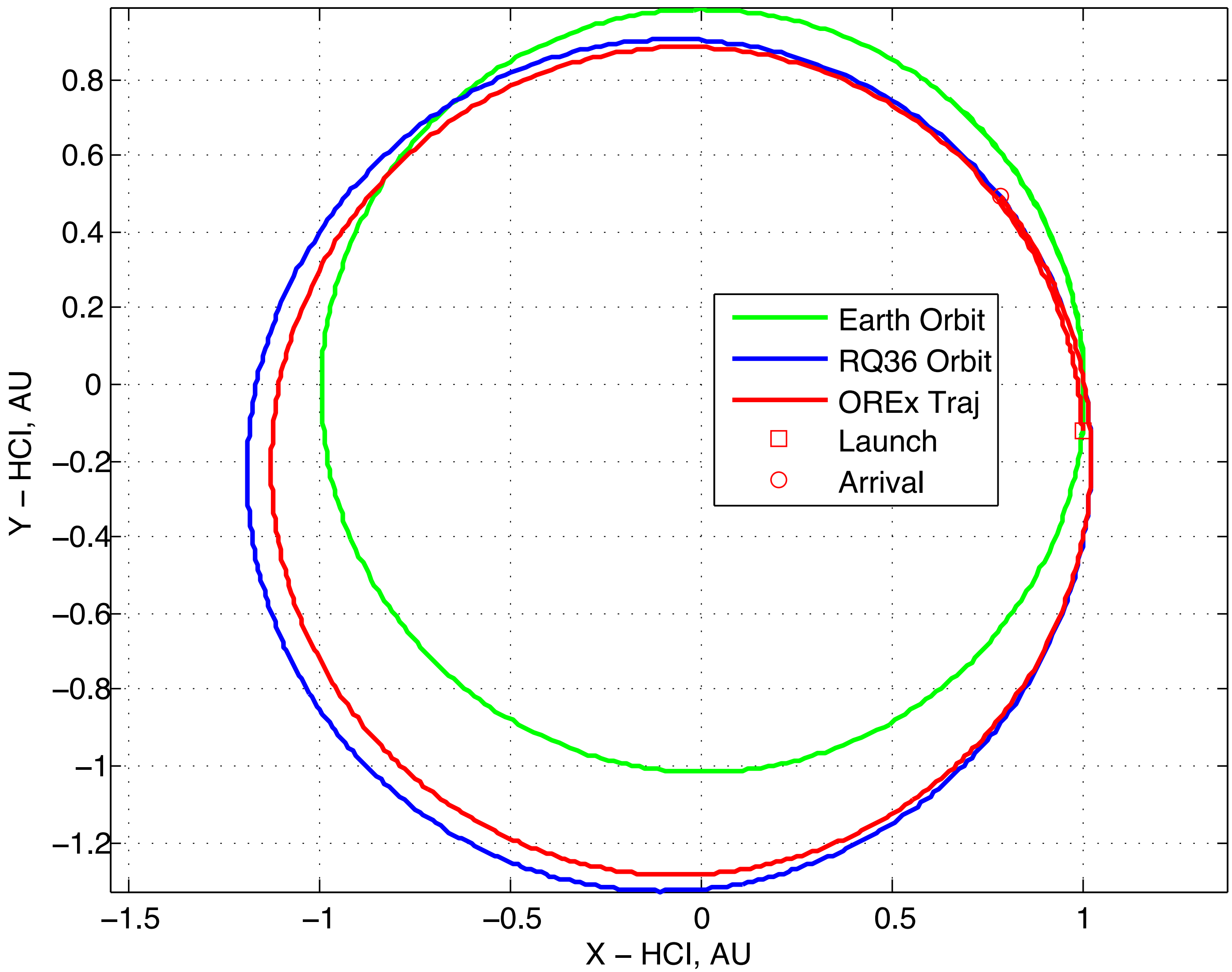


C_3 variation throughout the backup launch window.

Backup Launch Window Details

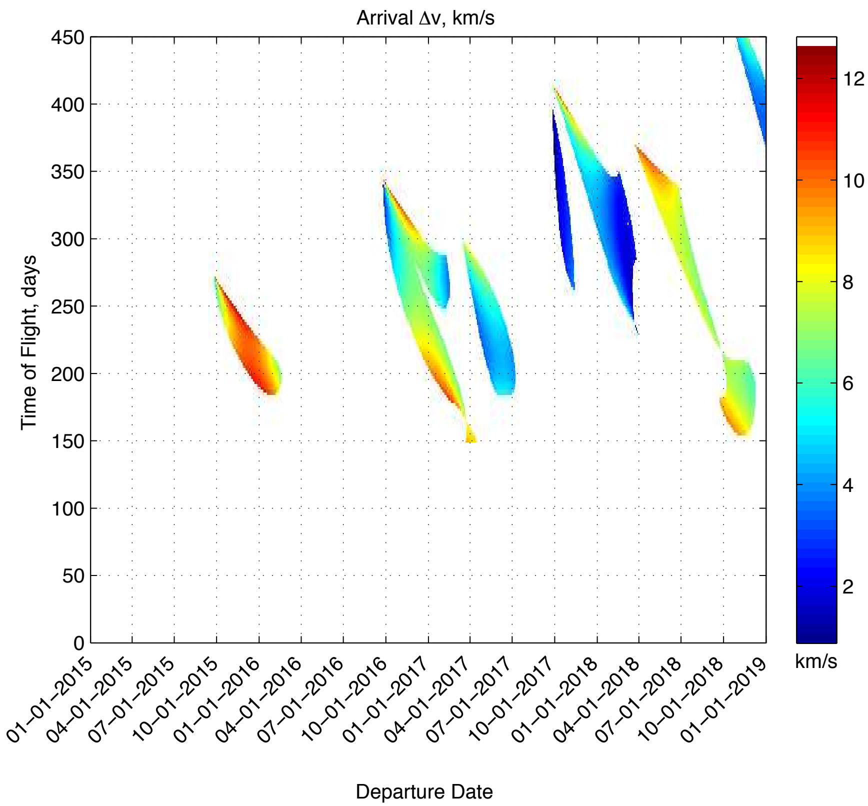
Day	Date	C_3 (km^2/s^2)	DLA	RLA	DSM1 (m/s)	AAM1 (m/s)	AAM2 (m/s)	AAM3 (m/s)	Total Δv (m/s)
1	9-15-2017	28.5	35.78°	185.63°	684.67	288.29	124.42	4.77	1102.15
2	9-16-2017	28.7	35.33°	186.17°	621.44	292.94	126.03	4.70	1045.10
3	9-17-2017	28.7	34.86°	186.76°	562.92	296.10	127.21	4.70	990.93
4	9-18-2017	28.7	34.86°	186.80°	482.37	323.28	136.09	4.70	946.44
5	9-19-2017	28.9	34.13°	187.53°	436.46	315.63	133.74	4.70	890.53
6	9-20-2017	29.0	33.76°	187.90°	374.00	329.66	138.56	4.70	846.91
7	9-21-2017	28.9	33.75°	187.92°	289.71	368.72	152.25	4.70	815.37
8	9-22-2017	29.0	33.71°	187.97°	209.25	412.39	168.11	4.70	794.45
9	9-23-2017	29.0	33.66°	188.03°	134.48	459.35	185.57	4.70	784.10
10	9-24-2017	29.0	33.81°	188.88°	85.04	482.20	194.26	4.70	766.20
11	9-25-2017	29.1	33.09°	189.26°	39.73	444.37	180.80	4.70	669.60
12	9-26-2017	29.2	31.22°	189.28°	59.29	444.38	181.88	4.70	690.25
13	9-27-2017	29.2	31.24°	190.27°	101.60	441.47	181.52	4.70	729.28
14	9-28-2017	29.2	31.81°	191.56°	157.39	417.00	173.27	4.70	752.36
15	9-29-2017	29.2	31.82°	192.49°	206.40	416.42	173.58	4.70	801.10
16	9-30-2017	29.2	32.13°	193.58°	263.01	405.42	170.06	4.70	843.19
17	10-1-2017	29.1	32.53°	194.70°	321.22	398.45	167.91	4.70	892.28
18	10-2-2017	29.1	32.57°	195.63°	378.31	396.52	167.70	4.70	947.22
19	10-3-2017	29.1	32.92°	196.70°	437.79	394.26	167.07	4.70	1003.83
20	10-4-2017	29.1	33.03°	197.64°	495.71	394.89	167.65	4.70	1062.94
21	10-5-2017	29.0	33.25°	198.63°	561.13	391.36	166.24	4.70	1123.43

Example Backup Launch Window Trajectory

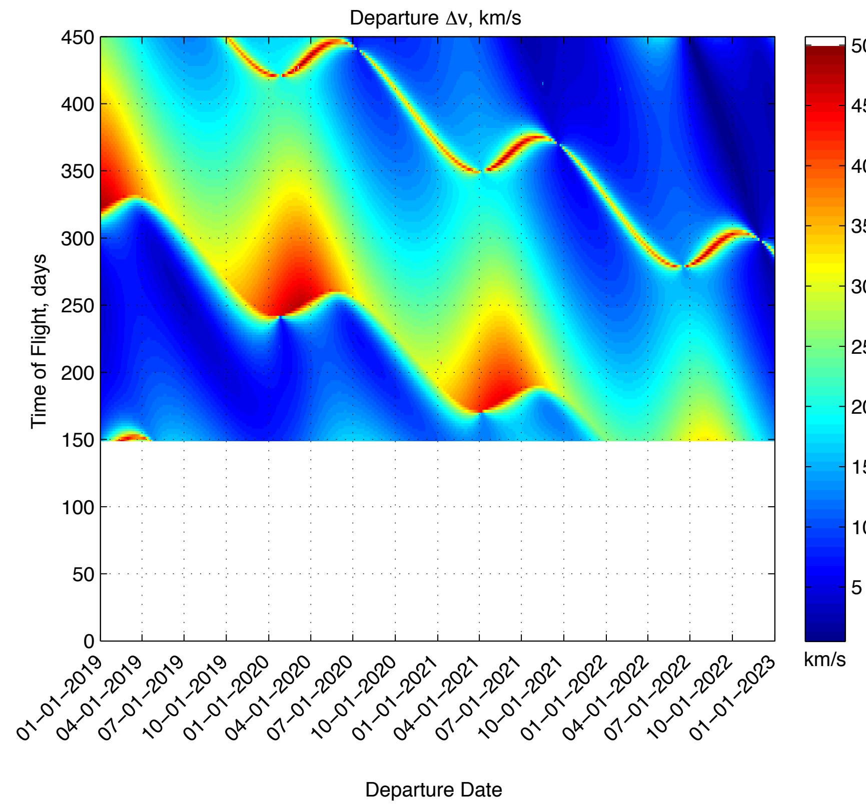


OSIRIS-REx backup launch window outbound cruise trajectory to 1999 RQ_{36}, ecliptic plane projection.

Preliminary Trajectory Analysis Scans



Δv to arrive at RQ_{36}.



Δv to depart from RQ_{36}.