Using the General Mission Analysis Tool (GMAT)

Steven P. Hughes (NASA GSFC)
Darrel J. Conway (Thinking Systems, Inc.)

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NASA Goddard Space Flight Center
Tutorial Overview

- GMAT Basics (D. Conway)
- Mission Design Walk-Through (S. Hughes)
- GMAT Navigation Example (D. Conway)
- CSALT Demo (S. Hughes)
- Wrap-Up (S. Hughes)
I. GMAT Overview

II. Key Concepts
   a. Two Parallel Interfaces
   b. Resources and Commands
   c. Fields and Parameters
   d. Execution Model

III. Tour of the Graphical User Interface
   a. GUI Controls
   b. Resources Tree
   c. Mission Tree
   d. Output Tree
   e. OrbitView

IV. 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

V. A Quick Demo
GMAT Timeline

• Design Started in 2002
• Development Began in early 2003
• Initial Fully Functional Build: 2004
• First Public Demonstration/Release: 2007
  – Alpha/pre-Beta System
  – Used for Mission Design and Analysis
• First Public Beta Release: R2011a
• First Production Release: R2013a
  – First Operationally Ready Release
  – Target: The Advanced Composition Explorer (ACE)
• Current Release: R2016a
  – First Production Ready Navigation Release
  – Target: The Solar and Heliospheric Observatory (SOHO)
System Features

- Platforms Supported: Windows, Mac, Linux
- External Interfaces: MATLAB and Python
- Development Approach: Modified Open Source
  - Developed Behind a NASA Firewall
  - Periodic Public Releases of Builds and Code
  - Supports a Robust Plugin Framework
- Extensively Tested
  - More than 13000 Core Code Tests Run Nightly
  - More than 3000 GUI Tests
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
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)
  - Training (full-day live training courses and recorded training available via YouTube channel)
Usage Summary

- 9 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
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
Execution Model (Cont’d)

- Batch execution model
GUI and script are nearly interchangeable (but not totally).
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
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

<table>
<thead>
<tr>
<th>Operators</th>
<th>Built-in Functions</th>
</tr>
</thead>
<tbody>
<tr>
<td>+</td>
<td>add</td>
</tr>
<tr>
<td>-</td>
<td>subtract</td>
</tr>
<tr>
<td>*</td>
<td>multiply</td>
</tr>
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<td>/</td>
<td>divide</td>
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<tr>
<td>( \cdot )</td>
<td>transpose</td>
</tr>
<tr>
<td>( \wedge )</td>
<td>power</td>
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<tr>
<td></td>
<td>sin</td>
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<td></td>
<td>cos</td>
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<td>tan</td>
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<td>asin</td>
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<td>acos</td>
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<td>atan</td>
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<td>atan2</td>
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<tr>
<td></td>
<td>log</td>
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<tr>
<td></td>
<td>log10</td>
</tr>
<tr>
<td></td>
<td>exp</td>
</tr>
<tr>
<td></td>
<td>DegToRad</td>
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<td></td>
<td>RadToDeg</td>
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<td></td>
<td>abs</td>
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<td></td>
<td>sqrt</td>
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<td></td>
<td>norm</td>
</tr>
<tr>
<td></td>
<td>det</td>
</tr>
<tr>
<td></td>
<td>inv</td>
</tr>
</tbody>
</table>
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
Three types of solvers:
- Targeter (using Differential Corrector)
- Optimizers (using either optimizer)
- Estimator (using Batch Estimator and, soon, EKF)

Similar to loops, with specific nested commands:
- Target: Vary, Achieve
- Optimize: Vary, NonlinearConstraint, Minimize

See the tutorials for examples
Live Demonstration

- Performing a Hohmann Transfer
General Mission Analysis Tool (GMAT)

GMAT Application to GSFC Mission Design
Steven P. Hughes
08 Feb. 2016

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

NOTE: This is a snapshot from TESS PDR and some details have changed.
01: Mission Overview

TESS Mission Design Pre-CDR Peer Review

Joel Parker
March 11, 2015
**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
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

Phasing loops 2 & 3 variable by launch date

- Phasing Loop 1 (5.5 d)
- Phasing Loop 2 (approx. 8 d)
- Phasing Loop 3 (approx. 10.5 d)
- Transfer Orbit (22 d)
- Period Adjust (14 d)

Ascent and Commissioning (60 days)

Science Orbit 1

Science Orbit 2

- Injection
- Cal Burn
- P1
- P2
- P3
- A1
- A2
- A3
- TCM
- Lunar Swing-by
- TCM
- TCM
- PAM
- Period Adjust
- TCM

- $r_a = 250,000\ \text{km}$
- $r_p = 108,400\ \text{km (17 RE)}$
- $r_s = 376,300\ \text{km (59 RE)}$

- a_p = 200 km
- a_p = 200 km

- Burn if necessary
- DV Burn
- Perigee Passage
Nominal Aug 10 solution: Inertial frame
Nominal Aug 10 solution: Rotating frame

For a loop in the 1st 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

Roughly 47 degree plane change at flyby
# Key L2 Mission Design Requirements

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

Green Requirements are Orbit Design Drivers
03: Trajectory Design Process

TESS Mission Design Pre-CDR Peer Review

Joel Parker
March 11, 2015
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
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*
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
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**
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 $a$ is semimajor axis (scaled by distance between the primary bodies), $e$ is eccentricity and $i$ 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 $T$ 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 $T$ to infer the shape of the orbit before flyby.
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 \( K = \cos(i)\sqrt{1 - e^2} \) is constant, where \( e \) is eccentricity and \( i \) 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)

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

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

Evolution of ecliptic AOP over 20 years.

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

Change since PDR Peer Review
All analyses share common force models, spacecraft parameters, solar system models, to the extent practical.

### Spacecraft model

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<table>
<thead>
<tr>
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</thead>
<tbody>
<tr>
<td>Mass*</td>
<td>201.9 kg</td>
</tr>
<tr>
<td>Coeff. of reflectivity (SRP)</td>
<td>1.5</td>
</tr>
<tr>
<td>SRP area</td>
<td>3.5 m²</td>
</tr>
</tbody>
</table>

*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

<table>
<thead>
<tr>
<th></th>
<th>Phasing loops</th>
<th>Flyby</th>
<th>Mission orbit</th>
<th>Solar system ephem</th>
</tr>
</thead>
<tbody>
<tr>
<td>Central-body gravity</td>
<td>JGM-2 40×40</td>
<td>Moon point mass</td>
<td>JGM-2 8×8</td>
<td>DE421</td>
</tr>
<tr>
<td>Third-body gravity</td>
<td>Sun, Moon</td>
<td>Sun, Earth</td>
<td>Sun, Moon</td>
<td></td>
</tr>
<tr>
<td>SRP</td>
<td>Enabled</td>
<td>Enabled</td>
<td>Enabled</td>
<td></td>
</tr>
<tr>
<td>Drag</td>
<td>Disabled</td>
<td>Disabled</td>
<td>Disabled</td>
<td></td>
</tr>
</tbody>
</table>
### Stage 1: Outbound Sequence Constraints

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

Guess with Discontinuities

Solution, No Discontinuities
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

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>P1–P3 altitude</td>
<td>$\geq 600 \text{ km}$</td>
<td>Phasing loop perigee altitude</td>
</tr>
<tr>
<td>A3 radius</td>
<td>$\leq$ pre-flyby radius</td>
<td></td>
</tr>
<tr>
<td>A2 radius</td>
<td>$A_1 \leq A_2 \leq A_3$</td>
<td></td>
</tr>
<tr>
<td>A1 radius</td>
<td>275,000 km</td>
<td>A1 design radius</td>
</tr>
<tr>
<td>Separation altitude</td>
<td>200 km</td>
<td>LV requirement</td>
</tr>
<tr>
<td>Separation inclination</td>
<td>28.5° TOD</td>
<td>LV requirement</td>
</tr>
<tr>
<td>Separation epoch</td>
<td>match launch modeling</td>
<td>Analytical model based on launch site</td>
</tr>
<tr>
<td></td>
<td>&amp; desired phasing loop</td>
<td>duration</td>
</tr>
<tr>
<td>Launch RA</td>
<td>Consistent w/ KSC launch</td>
<td></td>
</tr>
</tbody>
</table>
Final Converged Solution
DISCUSSION AND QUESTIONS
GMAT Navigation

Darrel J. Conway, Thinking Systems, Inc.

This presentation Uses Materials in the GMAT Simulation and Estimation Tutorials, Written by D. S. Cooley of NASA GSFC

NASA Goddard Space Flight Center
Outline

I. GMAT Navigation Overview
II. Estimation Walkthrough
III. Discussion and Questions
Navigation Capabilities

• Estimators
  – Batch Least Squares
  – Under Development: Extended Kalman Filter
• Measurement Simulator
• Measurement Models
  – DSN 2-way Range and Doppler
  – (Alpha) GN 2-way Range and Doppler
  – (Alpha) SN 4-Leg Range and 5-Leg Doppler
• Solve-For Parameters:
  – Cartesian State
  – Cd and Cr
  – Measurement Bias
• Built as a GMAT Plug-In
  – Navigation Features Script Based
  – Alpha Measurements Disabled by Default
GMAT’s BLS Estimator

- Coded from Tapley, Schutz and Born, *Statistical Orbit Determination*
- Extended with Features from GTDS
  - Robust Inversion Code
  - Sigma Editing
  - Detailed Estimation Output
  - GTDS Based Convergence Criteria
- Provides Results Comparable to GTDS
- Approved for Operations for SOHO
  - Pending successful ORR, 2017-01-31
Measurement Models

• Measurements Contained in “TrackingFileSet”s
• Specified by the Path Followed by Signals
• R2016a Measurement Models:
  – DSN Range, in Range Units
  – DSN Doppler
• Alpha Measurements:
  – 2-way Range and Doppler
  – 4-Leg Range and 5-Leg Doppler (TDRSS)
GMAT Nav Components

- Spacecraft
  - Solve-For
  - Hardware

- Transmitter
  - Antenna

- Transponder
  - Antenna

- Receiver
  - Antenna

- Tracking File Set
  - Tracking Files
  - Propagator
    - Integrator
    - Force Model

- Estimator
  - Conv’ce Criteria
    - Sigma Editor
    - Tracking File Set
    - Propagator

- Ground Station
  - Media Corr’n’s
    - Hardware
    - Error Model

- Simulator
  - Tracking File Set
  - Propagator

Bias, Sigma
ESTIMATION WALK THROUGH
Configuring Resources (1 of 4)

- **Spacecraft**
  - Solve-For
  - Hardware

- **Ground Station**
  - Media Corr’ns
  - Hardware
  - Error Model

- **Transmitter**
  - Antenna

- **Transponder**
  - Antenna

- **Receiver**
  - Antenna

- **Tracking File Set**
  - Tracking Files

- **Propagator**
  - Integrator
  - Force Model

- **Error Model**
  - Solve-For
  - Bias, Sigma

- **Estimator**
  - Conv’ce Criteria
  - Sigma Editor
  - Tracking File Set
  - Propagator

- **Simulator**
  - Tracking File Set
  - Propagator
Configuring Resources (3 of 4)
Configuring Resources (4 of 4)

Spacecraft
- Solve-For
- Hardware

Ground Station
- Media Corr’ns
- Hardware
- Error Model

Transmitter
- Antenna

Transponder
- Antenna

Receiver
- Antenna

Antenna

Tracking File Set
- Tracking Files

Propagator
- Integrator
- Force Model

Error Model
- Solve-For
- Bias, Sigma

Estimator
- Conv’ce Criteria
- Sigma Editor

Simulator
- Tracking File Set
- Propagator
Mission Sequence

- One Command: RunEstimator
Output Details

- Output Data:
  - Estimation Status (Converged, or Failure Reason)
  - Final Estimated State
  - Final Covariances and Correlations
  - Detailed Estimation Report
DISCUSSION AND QUESTIONS
Collocation Stand Alone Library and Toolkit (CSALT)
The Optimal Control Problem

\[ J = \sum_{k=1}^{N} (\Phi[y^{(k)}_o(t_o), t_o, y^{(k)}(t_f), t_f^{(k)}]) + \int_{t_o}^{t_f} \lambda^{(k)}[x^{(k)}(t), u^{(k)}(t), t^{(k)}] \, dt \]

subject to the dynamics constraints, \( f \),

\[ y^{(k)}(t) = f^{(k)}[y^{(k)}(t), u^{(k)}(t), t^{(k)}] \]

the algebraic path constraints, \( g \),

\[ g^{\text{min}}^{(k)} \leq g[y(t), u(t), t] \leq g^{\text{max}}^{(k)} \]

the integral constraints, \( w \),

\[ w^{\text{min}}^{(k)} \leq w[y(t), u(t), t] \leq w^{\text{max}}^{(k)} \]

and the boundary conditions, \( \varphi \)

\[ \varphi^{\text{min}}^{(k)} \leq \varphi[y^{(k)}_o(t_o), t_o, x^{(k)}(t_f), t_f^{(k)}] \leq \varphi^{\text{max}}^{(k)} \]
Fig. 1: Schematic of linkages for multiple-phase optimal control problem. The example shown in the picture consists of five phases where the ends of phases 1, 2, and 3 are linked to the starts of phases 2, 3, and 4, respectively, while the end of phase 2 is linked to the start of phase 5.

Note, the figure above was taken from Patterson, M, and Rao, A, A MATLAB Software for Solving Multiple-Phase Optimal Control Problems Using hp–Adaptive Gaussian Quadrature Collocation Methods and Sparse Nonlinear Programming".
## User Stories: Trajectory Regimes

<table>
<thead>
<tr>
<th>ID</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>CO.TD-1</td>
<td><strong>Basic Orbit Transfer:</strong> I need to design an optimal maneuver to transfer to a nearby orbit.</td>
</tr>
<tr>
<td>CO.TD-2</td>
<td><strong>Body to body transfer:</strong> I need to design a trajectory that does a direct transfer from one body to another, in my case, from Earth to Moon.</td>
</tr>
<tr>
<td>CO.TD-3</td>
<td><strong>Interplanetary Multiple flyby:</strong> I need to design a trajectory that leaves Earth, performs multiple gravitational flybys, and arrives at a final target celestial body. I have a guess at the flyby times, order, and flyby conditions at each body.</td>
</tr>
<tr>
<td>CO.TD-4</td>
<td><strong>Spiral:</strong> I need to design a trajectory that spirals into and/or away from a celestial body requiring many (100s) of revolutions.</td>
</tr>
<tr>
<td>CO.TD-5</td>
<td><strong>Libration point maintenance:</strong> I need to use collocation to improve the robustness and quality of libration stationkeeping solutions.</td>
</tr>
</tbody>
</table>
## User Stories: Trajectory Regimes

<table>
<thead>
<tr>
<th>ID</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>CO.TD-6</td>
<td><strong>Weak stability transfer:</strong> I need to use collocation to find trajectory transfers in a weak stability, multi-body environment.</td>
</tr>
<tr>
<td>CO.TD-7</td>
<td><strong>Formation:</strong> I need to use collocation to optimize maneuvers of multiple spacecraft.</td>
</tr>
<tr>
<td>CO.TD-8</td>
<td><strong>Rendezvous/Prox Ops:</strong> I need to use collocation to optimize rendezvous and docking operations and proximity operations.</td>
</tr>
<tr>
<td>CO.TD-9</td>
<td><strong>Probe Separation:</strong> I need to use collocation to determine optimal trajectories for probe release and spacecraft separation planning.</td>
</tr>
</tbody>
</table>
GMAT and CSALT

GMAT High Level Design

GMAT Software Demonstration
## CSALT MATLAB Prototype Benchmarking Results

<table>
<thead>
<tr>
<th>Problem Name</th>
<th>Truth Source</th>
<th>GMAT Prototype Optimal Cost Function</th>
<th>External Truth Optimal Cost Function</th>
<th>(gmat-truth)/truth</th>
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</thead>
<tbody>
<tr>
<td>Rayleigh (control path constraint)</td>
<td>SOS</td>
<td>44.72093885</td>
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<td>5.925E-08</td>
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<td>Rayleigh (state and path constraint)</td>
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<td>Goddard Rocket Problem</td>
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<td>Conway Low Thrust</td>
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<td>9.51233830E-02</td>
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<td>PSOPT</td>
<td>4.571044</td>
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<td>MoonLander</td>
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GMAT Software Demonstration AAS Guidance and Control Conference, Feb. 7, 2017
CSALT Applications

DRO to L2 Transfer

Obstacle Avoidance Problem

GMAT Software Demonstration

AAS Guidance and Control Conference, Feb. 7, 2017
Interplanetary Application with Comparison to EMTG
DISCUSSION AND QUESTIONS
A significant part of certification of navigation functionality is using GMAT in the FDF operational environment.
We are performing final testing for using GMAT as the core tool in the SDO MOC for maneuver planning and product generation.
GMAT API and Potential Future Efforts

We are currently in the early phases of developing a GMAT API, and considering the possibility of deconstructing GMAT to allow components to be used onboard for real time mission planning and opportunistic science.
Finding Out More

• **Main information portal:** gmatcentral.org

• For New Users
  ▪ Obtain GMAT
  ▪ Training Videos
  ▪ Training Material
  ▪ Sample Missions (scripts distributed with application)
  ▪ User Guide (also pdf distributed with application)

• For New Developers
  ▪ Obtain Code
  ▪ Style Guide
  ▪ Compilation Instructions
  ▪ Design Docs (also pdf distributed with application)
Thanks! Questions?
DISCUSSION
AND
QUESTIONS
# Software Tool Comparison

<table>
<thead>
<tr>
<th>Evaluation Area</th>
<th>GMAT</th>
<th>FreeFlyer</th>
<th>STK</th>
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</tbody>
</table>

* In progress/under development
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

- **GMAT R2015a**
  - Customizable orbit segment colors
  - Mars-GRAM 2005 atmosphere model
  - LHS parameter dependencies
  - New solver algorithms

- **GMAT R2015b (internal)**
  - 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
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
## L3 Mission Design Requirements

<table>
<thead>
<tr>
<th>ID</th>
<th>Parent ID</th>
<th>Title</th>
<th>Requirement</th>
<th>Compliance</th>
</tr>
</thead>
<tbody>
<tr>
<td>L3_FD_1</td>
<td>MRD_10, MRD_51</td>
<td>Mission Orbit SMA</td>
<td>The target mission orbit Semi-Major Axis (SMA) shall be 38 Re.</td>
<td>Comply. Design constraint.</td>
</tr>
</tbody>
</table>

**Diagram:**

- **TESS HEO Orbit Definitions**
  - Lunar Resonance Phasing (LRP) or PLEP Angle
  - Transfer Orbit
  - TOF Lunar Flyby to PLEP
  - Line of Nodes @ Lunar Flyby
  - Line of Apices
  - PLEP
  - P/L HEO
  - Moon at TOF
  - Earth-Moon Plane; Descending Outbound

**Equation:**

\[
\cos AOP = \frac{1}{r_e} \left(1 - \frac{a_r (1 - e_r^2)}{r_m}\right)
\]
## L3 Mission Design Requirements

<table>
<thead>
<tr>
<th>ID</th>
<th>Parent ID</th>
<th>Title</th>
<th>Requirement</th>
<th>Compliance</th>
</tr>
</thead>
<tbody>
<tr>
<td>L3_FD_3</td>
<td>MRD_53</td>
<td>Mission Orbit Minimum Perigee</td>
<td>FD shall target a mission orbit with a minimum perigee that shall stay above GEO radius + 200 km.</td>
<td>Comply. Results shown to 100 years.</td>
</tr>
<tr>
<td>L3_FD_29</td>
<td>MRD_52</td>
<td>Mission Orbit Maximum Perigee</td>
<td>FD shall target a mission orbit with a maximum perigee that shall stay below 22 Re for the duration of the mission.</td>
<td>Comply. All &lt;20.5 Re</td>
</tr>
<tr>
<td>L3_FD_30</td>
<td>MRD_10</td>
<td>Transfer Orbit Maximum Apogee</td>
<td>FD shall target a lunar flyby that results in a transfer orbit with a maximum apogee less than 90 Re.</td>
<td>Comply. All &lt;80 Re</td>
</tr>
</tbody>
</table>

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

<table>
<thead>
<tr>
<th>ID</th>
<th>Parent ID</th>
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<th>Requirement</th>
<th>Compliance</th>
</tr>
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<tbody>
<tr>
<td>L3_FD_21</td>
<td>MRD_54</td>
<td>Launch Period</td>
<td>FD shall design for at least 5 launch days in any given Lunar cycle.</td>
<td>Comply. At least 9 sol’ns/mo for current period.</td>
</tr>
<tr>
<td>L3_FD_22</td>
<td>MRD_55</td>
<td>Launch Window</td>
<td>FD shall design for launch windows of at least 5 minutes during each day of the launch period.</td>
<td>Comply. Current strategy meets req.</td>
</tr>
<tr>
<td>L3_FD_27</td>
<td>MRD_42</td>
<td>Commissioning Duration</td>
<td>FD shall design the phasing loops and post lunar encounter transfer orbit to achieve mission orbit within 2 months after launch.</td>
<td>Comply. PAM at &lt; 43 days.</td>
</tr>
<tr>
<td>L3_FD_24</td>
<td>MRD_85</td>
<td>Sun in Instrument Boresight</td>
<td>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.</td>
<td>Comply. Basis for sol’n selection.</td>
</tr>
<tr>
<td>L3_FD_28</td>
<td>MRD_104</td>
<td>Delta-V Budget</td>
<td>FD shall design ascent-to-mission orbit to require no more than 215 m/s delta-V with 99% probability of success.</td>
<td>Comply. See detailed analysis.</td>
</tr>
<tr>
<td>L3_FD_25</td>
<td>MRD_129</td>
<td>Maneuver Magnitude</td>
<td>The largest maneuver magnitude shall be &lt;95m/s.</td>
<td>Comply. PAM &lt; 75 m/s</td>
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<tr>
<td>L3_FD_4</td>
<td>MRD_56</td>
<td>Eclipse Frequency and Duration</td>
<td>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.</td>
<td>Comply. No more than 11 &lt; 4hr + 1 &lt;= 6hr Needs updating</td>
</tr>
</tbody>
</table>

Requirements added to flow from L2

Change since PDR Peer Review

Consistent with EXP-TESS-GSFC-RQMT-0015 Rev 104

AAS Guidance and Control Conference, Feb. 7, 2017
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.

Acronyms

- CSALT Collocation Stand Alone Library and Toolkit
- EMTG Evolutionary Mission Trajectory Generator
- GMAT General Mission Analysis Tool
- OGA Other Governmental Organization
- PAM Post Apogee Maneuver
- PLEP Post Lunar Encounter Periapsis
- SNOPT Sparse Nonlinear Optimizer
- TESS Transiting Exoplanet Survey Satellite