The University of Texas at Austin
Aerospace Engineering and Engineering Mechanics
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ASE374K
CONOPS and Attitude Control by Operation Modes

2/29/2024
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Contents

Mission Patch

I. Review of Mission Scope

Mission Statement:

- Design Low-Earth-Orbit Solar Power Satellite ‘Redstone’ operation concept which receives solar power via solar panel and transmit energy to designated ground plant.

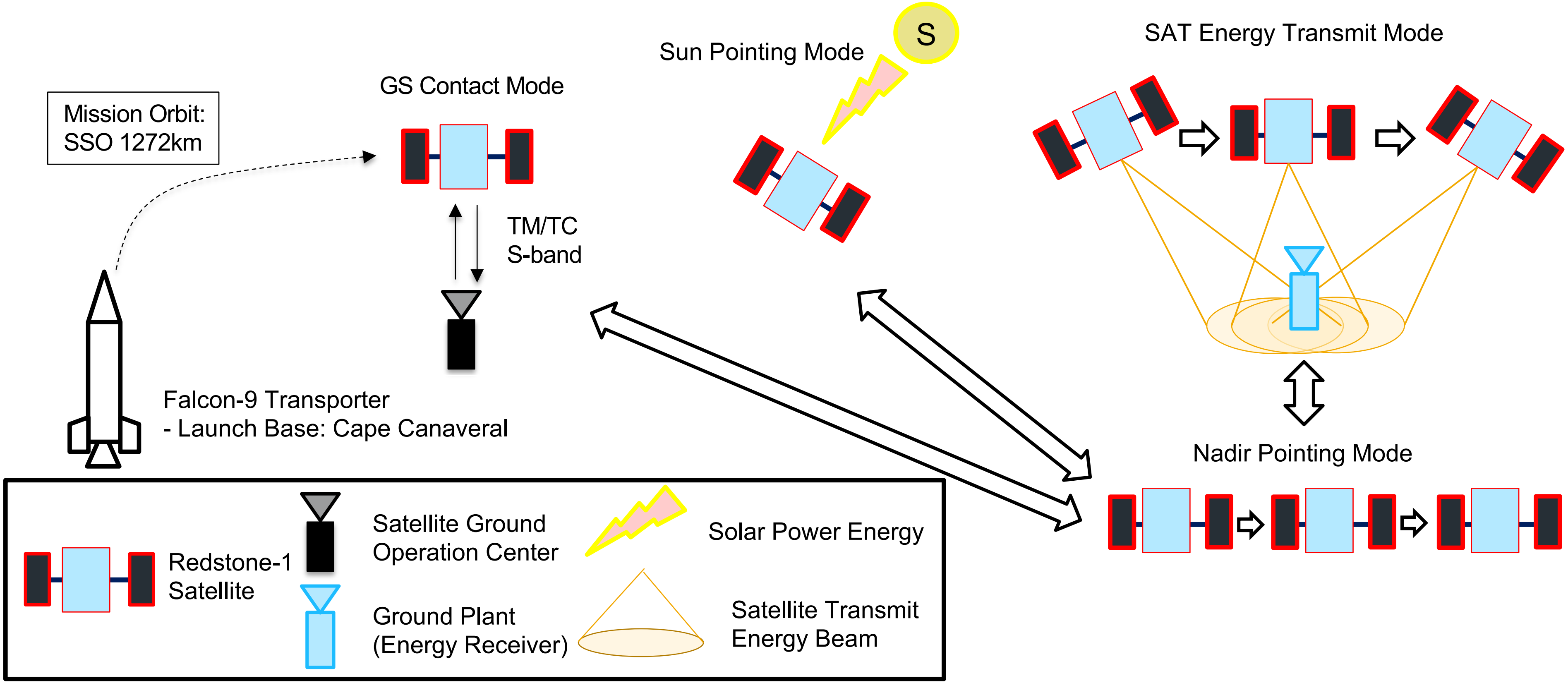
Primary Mission Objectives

- Satellite shall receive solar energy from sun more than [1,TBD] kJ per [1, TBD] orbit .
- Satellite shall transmit stored solar energy to [3, TBD] ground stations [1,TBD]kJ each per [1, TBD]week.

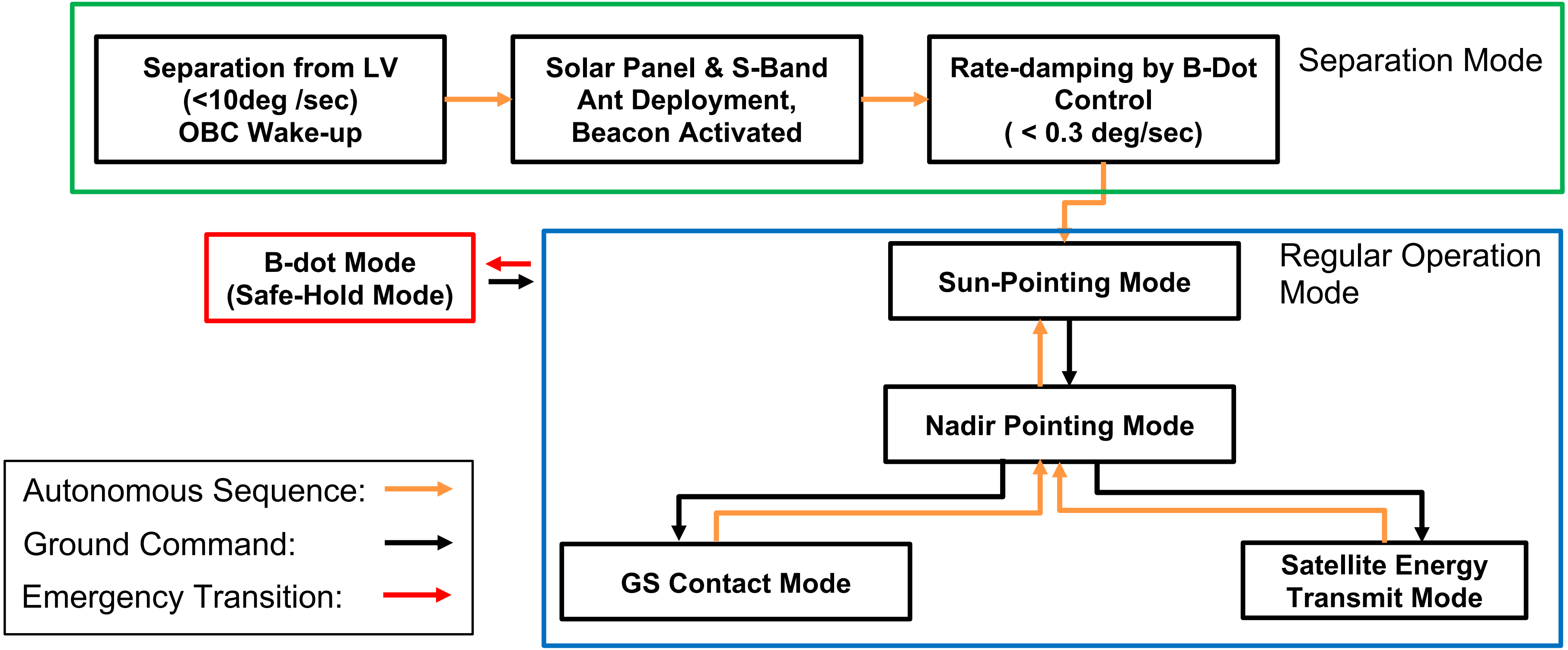
Secondary Mission Objective

- Satellite can transmit stored solar energy to cislunar orbit satellite or other satellites. [Concept Study]
- Propose constellation strategy for continuous solar power reception and transmission. [Concept Study]

II. Satellite Concept of Operations



III. Operation Mode Definition (1/4)



III. Operation Mode Definition (2/4)

Separation Mode

- Pre-Launch Condition: No power will be provided before separation from launch vehicle
- After separation from launch vehicle, the max tip-off rate of 10deg/sec is anticipated and satellite OBC will wake up and start to send beacon signal
- Solar array will be deployed for the generation of solar power
- B-dot Controller will autonomously start to run to damp out high tip-off rate of 10deg/sec to 0.3deg/sec: predicted time duration for detumbling is less than about 4.2 hrs.
- After settling down to satellite rate of 0.3deg/sec, the sun pointing controller will autonomously start to run to make solar array face to the sun for taking solar power by three-axis control method

Sun Pointing Mode

- The sun pointing mode requires aligning –y body axis to sun vector
- The sun-pointing controller will autonomously start from the “Post-Launch Mode” to make solar array face to the sun for taking solar power by three-axis control method.
- If no anomaly, most of the lifetime of satellite will be operated with “Sun Pointing Mode”
- Sun sensors, star tracker, gyro, three-axis magnetometer as sensors and RWAs, magnetic torque rod will be utilized for three axis control and momentum dumping
- Estimation algorithm will run to estimate to attitude and rate information

III. Operation Mode Definition (3/4)

Nadir Pointing Mode

- The Nadir Pointing Mode is aligning z vector to center of earth, and x vector to velocity of spacecraft
- Between transition to any normal operation mode (solar pointing mode, Nadir Pointing Mode, GS contact mode, Satellite Energy transmission mode), the satellite should transit to Nadir Pointing Mode
- The RWA shall comply performance requirements to transit from Nadir Point Mode to any operation mode and vice versa

GS contact Mode

- Since we are using S-band transceiver for TM/TC operation, the satellite will maintain Nadir Pointing mode while making communication with ground station.
- The S-band transceiver turns on this operation mode

Satellite Energy Transmit Mode

- Satellite z-axis will point designated ground energy plant and transmit the stored solar energy
- When this mode is planed, the satellite will point nadir direction and do roll-off and pitch-off maneuver if necessary
- Mode transition from “Sun-Pointing Mode” to “Satellite Energy Transmit Mode” will be performed by ground commands, NOT autonomously

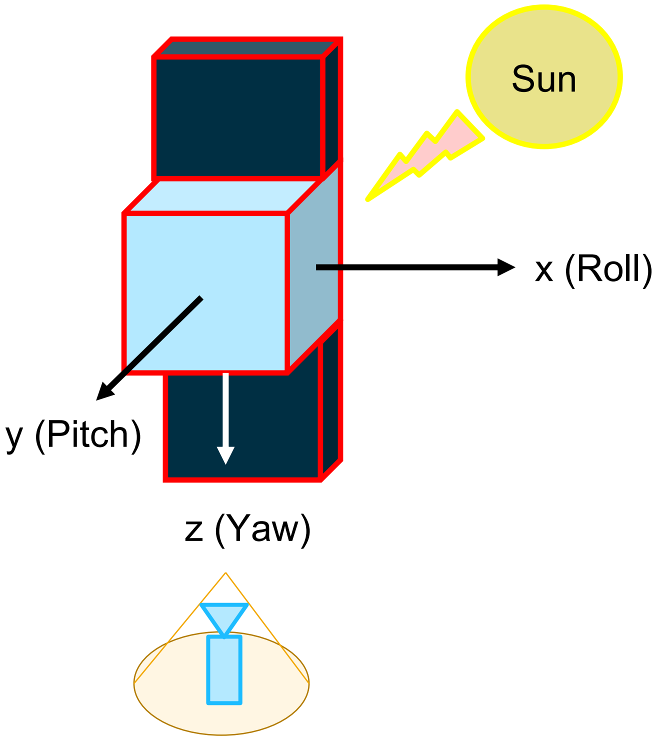
III. Operation Mode Definition (4/4)

Safe-Hold Mode

- If there are some kinds of anomaly at hardware of software by unknown reason or SEU, satellite will enter from any mode to "Safe-Hold Mode" autonomously
- Once abnormal condition is detected, the satellite will be transition to the "Safe-Hold Mode" for survival of satellite, which makes first OBC power-reset and power-off unnecessary hardware for power-saving and satellite survival.
- Magnetic torque rods, sun sensors and magnetometer will used for two axis control to save power. [TBD]
- Estimation algorithm will run to estimate the attitude and rate information. [TBD]
- Recovery from "Safe-Hold Mode" to "Sun-Pointing Mode" will be done by ground commands.

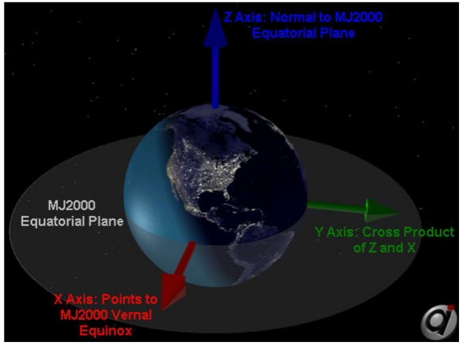
IV. Satellite Dynamics Modeling [1/4]

Satellite Body Frame Definition

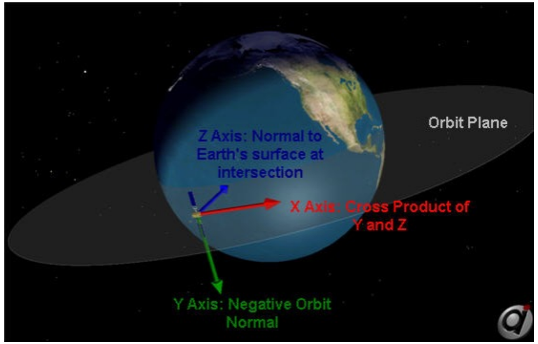


Satellite Orbit Frame Definition

MJ2000 Frame



LVLH Frame



https://ai-solutions.com/_freeflyeruniversityguide/attitude_reference_frames.htm

- We are using Euler Angle 3-1-2 Rotation System for generating Direction-Cosine-Matrix (DCM)
- We are using ECI Frame as MJ2000 Equatorial Frame to get satellite, sun, star locations
- We are using LVLH frame to get Nadir direction
- Since our proposed orbit is circular orbit x axis of LVLH frame = velocity vector of satellite

IV. Satellite Dynamics Modeling [2/4]

Important Assumptions

J2 Perturbation

- J2 Perturbation by earth oblateness is the single perturbation factor calculating satellite orbit trajectory
- We assume the Satellite maintain two-body dynamics except J2 Perturbation

Low-Earth-Orbit Circular Orbit

- We assume satellite's orbit is LEO perfect Circular Orbit
- We assume Geodetic frame of satellite is exactly same as LVLH frame, and x axis of LVLH frame is same as velocity vector of satellite since it is circular orbit

Perfect Sphere Earth

- We assume Earth is perfect sphere with 6378km radius, but only assume oblateness for J2 perturbation
- Nadir pointing of the Earth is same as Satellite pointing to center of earth, and z body axis is normal to tangent plane of nadir pointing ground location

Ignorance of Precession and Nutation of Satellite

- We assume satellite's MOI is diagonal matrix which indicate one axis of rotation does not impact another axis rotation

IV. Satellite Dynamics Modeling [3/4]

Satellite State Vector

$$X_{\mathrm{SAT}} = \begin{bmatrix} \vec{r}_I \\ \vec{v}_I \\ R_{\mathrm{BI}} \\ \vec{\omega}_B \\ \vec{\omega}_{\mathrm{RWA}} \end{bmatrix}$$

$\vec{r}_I$: Satellite Position Vector in ECI frame

$\vec{v}_I$: Setellite Velocity Vector in ECI frame

R_{BI} : Rotation Matrix from ECI frame to Body Frame

$\vec{\omega}_B$: Satellite Angular Velocity Vector $w.r.t$ Body Frame

$\vec{\omega}_{\mathrm{RWA}}$: Satellite Reaction Wheels Angular Velocity vector

Satellite Dynamic Model

$$\boxed{\dot{X}_{\mathrm{SAT}} = f_{\mathrm{SAT}}\left(t, X_{\mathrm{SAT}}, \vec{e}_a, \vec{d}_I, P\right) : \text{Non} - \text{linear dynamic Model of } \dot{X}_{\mathrm{sat}}}$$

t : Time

X_{SAT} : Satellite State Vector

$\vec{e}_a$: Satellite Reaction Wheels Motor Voltage (Control input)

$\vec{d}_I$: Satellite Disturbance Force Vector (Disturbance)

P : Satellite Paramter Structure

- m : Mass of Satellite
- J : Satellite's Moment of Inertia
- τ_m : Time constant for Motor Dynamics
- c_m : Steady – State Paramter : voltage to angular rate ω (rad/s)
- I_{RWA} : Momentum of Inertia of Reaction Wheel

IV. Satellite Dynamics Modeling [4/4]

Satellite Dynamic Model

$$\dot{X}_{\mathrm{SAT}} = f_{\mathrm{SAT}}\left(t, X_{\mathrm{SAT}}, \vec{e}_a, \vec{d}_I, P\right) : \text{Non} - \text{linear dynamic Model of } \dot{X}_{\mathrm{sat}}$$

$$X_{\mathrm{SAT}} = \begin{bmatrix} \vec{r}_I \\ \vec{v}_I \\ R_{\mathrm{BI}} \\ \vec{\omega}_B \\ \vec{\omega}_{\mathrm{RWA}} \end{bmatrix}$$

$\vec{r}_I$: Satellite Position Vector in ECI frame

$\vec{v}_I$: Setellite Velocity Vector in ECI frame

R_{BI} : Rotation Matrix from ECI frame to Body Frame

$\vec{\omega}_B$: Satellite Angular Velocity Vector $w.r.t$ Body Frame

$\vec{\omega}_{\mathrm{RWA}}$: Satellite Reaction Wheels Angular Velocity vector

$$\dot{r}_I = v_I$$

$$\dot{v}_I = \begin{bmatrix} -\mu \frac{r_{\mathrm{xI}}}{r^3}\left[1 - \frac{3}{2}J_2\left(\frac{R_{\mathrm{Earth}}}{r}\right)^2\left(5\left(\frac{r_{\mathrm{zI}}}{r}\right)^2 - 1\right)\right] \\ -\mu \frac{r_{\mathrm{yI}}}{r^3}\left[1 - \frac{3}{2}J_2\left(\frac{R_{\mathrm{Earth}}}{r}\right)^2\left(5\left(\frac{r_{\mathrm{zI}}}{r}\right)^2 - 1\right)\right] \\ -\mu \frac{r_{\mathrm{zI}}}{r^3}\left[1 - \frac{3}{2}J_2\left(\frac{R_{\mathrm{Earth}}}{r}\right)^2\left(5\left(\frac{r_{\mathrm{zI}}}{r}\right)^2 - 3\right)\right] \end{bmatrix}$$

$$\dot{R}_{\mathrm{BI}} = -\omega_B \times R_{\mathrm{BI}}$$

$$\dot{\omega}_B = J^{-1}\left(\vec{N}_B - [\omega_B \times]J\omega_B\right)$$

$$\dot{\omega}_{\mathrm{RWA}} = -\frac{\vec{\omega}_{\mathrm{RWA}}}{\tau_m} + \frac{c_m}{\tau_m}\vec{e}_a$$

Assumption: Reaction Wheel is Mounted parallel to Each Body Axis

J_2 Purturbation Model

J_2 : J_2 Purturbation Constant

μ : Gravity Constant

R_{Earth} : Earth Radius

$r = |\boldsymbol{r}_I|$

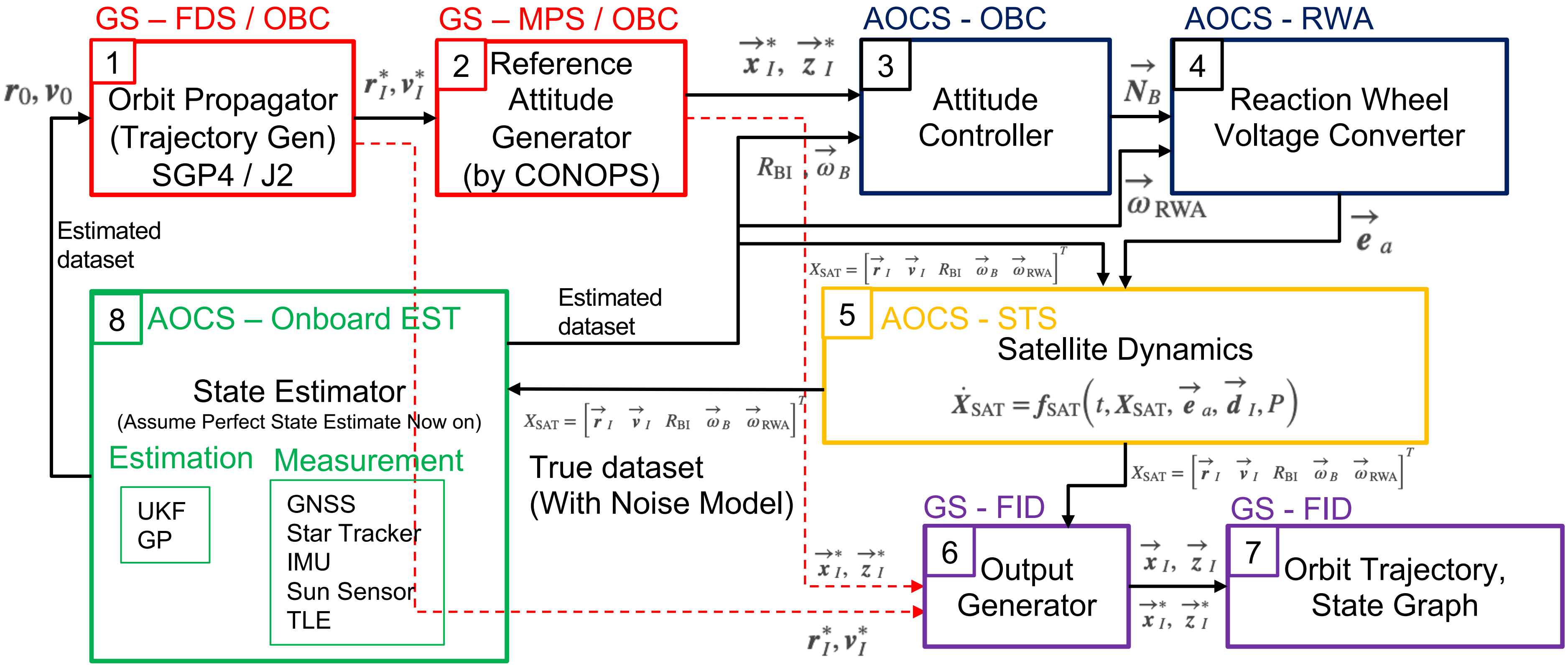
$\boldsymbol{r}_I = (r_{\mathrm{xI}}, r_{\mathrm{yI}}, r_{\mathrm{zI}})^T$

Body Torque Calculation RWA Dynamics

$\vec{N}_B$: Body Torque Vector

$$\vec{N}_B = -I_{\mathrm{RWA}}\dot{\omega}_{\mathrm{RWA}} = I_{\mathrm{RWA}}\left(-\frac{\vec{\omega}_{\mathrm{RWA}}}{\tau_m} + \frac{c_m}{\tau_m}\vec{e}_a\right)$$

V. Satellite Controller Modeling [1/6]



V. Satellite Controller Modeling [2/6]

1. On-orbit Propagator

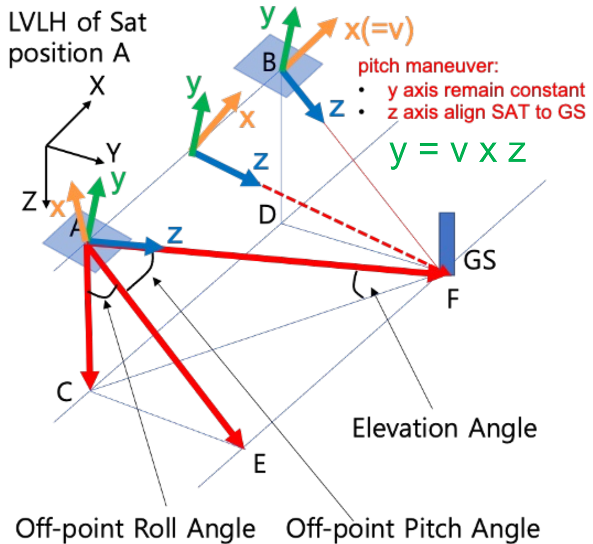
$\boldsymbol{r}_0, \boldsymbol{v}_0$ → **1** Orbit Propagator (Trajectory Gen) SGP4 / J2 → $\boldsymbol{r}_I^*, \boldsymbol{v}_I^*$

Input : Current Estimated r,v
Output: Propagated Trajectory
Using Existing J2 Model

- Can be substituted by SGP4 Model with TLE telemetry information
- Or can use more precise Propagation Model

2. Reference Attitude Generator

$\boldsymbol{r}_I^*, \boldsymbol{v}_I^*$ → **2** Reference Attitude Generator (by CONOPS) → $\vec{x}^*_I,\ \vec{z}^*_I$



***I/O* Structure**

Input : Propagated Trajectory $\boldsymbol{r}_I^*, \boldsymbol{v}_I^*$

Output : Reference Attitude Body Axis $\vec{z}^*_I, \vec{x}^*_I$

cf : $\vec{z}^*_I = R_{\mathrm{BI}}^T[0 \quad 0 \quad 1]^T, \vec{x}^*_I = R_{\mathrm{BI}}^T[1 \quad 0 \quad 0]^T$

Different Attitude Generation by Operating Mode

1) **Nadir Pointing Mode**

$$\Rightarrow \vec{x}^*_I = \frac{\vec{v}^*_I}{|\vec{v}^*_I|}, \quad \vec{z}^*_I = -\frac{\vec{r}^*_I}{|\vec{r}^*_I|}$$

2) **Energy Transmitting Mode**

$\Rightarrow$ Ground Location Vector $\vec{r}_{\mathrm{GS}}$

$$\Rightarrow \vec{z}^*_I = \frac{\vec{r}_{\mathrm{GS}} - \vec{r}^*_I}{|\vec{r}_{\mathrm{GS}} - \vec{r}^*_I|}, \quad \vec{y}^*_I = \frac{\vec{z}^*_I \times \vec{v}^*_I}{|\vec{z}^*_I \times \vec{v}^*_I|}$$

$$\Rightarrow \vec{x}^*_I = \vec{y}^*_I \times \vec{z}^*_I$$

3) **Sun Pointing Mode**

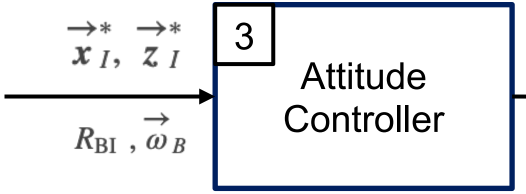
$\Rightarrow$ Sun direction vector $\hat{\boldsymbol{r}}_{\mathrm{sun}}$

$$\Rightarrow \vec{y}^*_I = -\hat{\boldsymbol{r}}_{\mathrm{sun}}, \quad \vec{z}^*_I = \frac{\vec{v}^*_I \times \vec{y}^*_I}{|\vec{v}^*_I \times \vec{y}^*_I|}$$

$$\Rightarrow \vec{x}^*_I = \vec{y}^*_I \times \vec{z}^*_I$$

V. Satellite Controller Modeling [3/6]

3. Attitude Controller



Use TRIAD Method to Determine R^*_{BI}

$\vec{x}^*_I, \vec{z}^*_I$: Reference AttiudeInput

$$\vec{b} = \frac{\vec{z}^*_I \times \vec{x}^*_I}{\left|\vec{z}^*_I \times \vec{x}^*_I\right|}$$

if $\vec{z}^*_I \times \vec{x}^*_I = 0 \Rightarrow \vec{b} = \vec{0}$

$$\vec{a} = \vec{b} \times \vec{z}^*_I$$

$$R^*_{BI} = \begin{bmatrix} | & | & | \\ \vec{a} & \vec{b} & \vec{z}^*_I \\ | & | & | \end{bmatrix}^T$$

Calculate error vector $\vec{e}$

R^*_{BI} : Refernce Direction Cosine Matrix

R_E : Error Direction Cosine Matrix

$$R^*_{BI} = R_E R_{BI} \Leftrightarrow R_E = R^*_{BI} R^T_{BI}$$

$$R_E = \begin{bmatrix} e_{11} & e_{12} & e_{13} \\ e_{21} & e_{22} & e_{23} \\ e_{31} & e_{32} & e_{33} \end{bmatrix} \Rightarrow \vec{e} = \begin{bmatrix} e_{23} - e_{32} \\ e_{31} - e_{13} \\ e_{12} - e_{21} \end{bmatrix}$$

We assume $\dot{\vec{e}} = -\vec{\omega}_B$

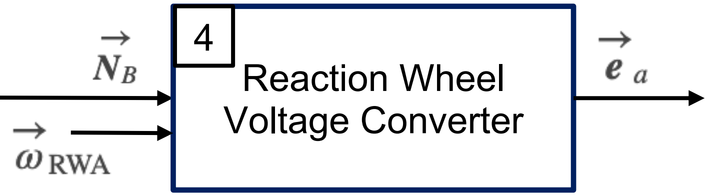
Design PD Controller

$$\vec{N}_B = K_p \vec{e} - K_d \vec{\omega}_B + \left[\vec{\omega}_B \times\right] J \vec{\omega}_B$$

K_p, K_d : PD gain matrix

$\vec{N}_B$: Required body torque (Control Input)

4. Reaction Wheel Voltage Converter



Reaction Wheel Motor Dynamics

$$\frac{\Omega(s)}{E_a(s)} = \frac{c_m}{\tau_m s + 1} \text{ : 1st order system}$$

$\Omega(s)$: RWA Angular Rate in s domain

$E_a(s)$: RWA Input Voltage in s domain

$$\Omega(s)(\tau_m s + 1) = c_m E_a(s)$$

$$s\Omega(s) = \frac{1}{\tau_m}(c_m E_a(s) - \Omega(s))$$

$$\therefore \dot{\omega}_{RWA}(t) = \frac{1}{\tau_m}(c_m e_a(t) - \omega_{RWA}(t))$$

Reaction Wheel Voltage Converter Design

Input : $\vec{N}_B$: Required body torque (Attitude Control output)

$\vec{\omega}_{RWA}$: Reaction wheel angular rate (From Estimator)

Output : $\vec{e}_a$ Required motor voltage (Control Input to SAT)

$$\vec{N}_B = -I_{RWA}\dot{\omega}_{RWA} \Rightarrow -I^{-1}_{RWA}\vec{N}_B = \dot{\omega}_{RWA}$$

$$-I^{-1}_{RWA}\vec{N}_B = \frac{1}{\tau_m}\left(c_m e_a - \vec{\omega}_{RWA}\right)$$

$$\therefore \vec{e}_a = \frac{1}{c_m}\left[-I^{-1}_{RWA}\vec{N}_B \tau_m + \vec{\omega}_{RWA}\right]$$

V. Satellite Controller Modeling [4/6]

5. Satellite Dynamics

$\vec{e}_a$ Xk, tspan → **5** Satellite Dynamics → [tVeck, xMatk]

$$\dot{X}_{\mathrm{SAT}} = f_{\mathrm{SAT}}\left(t, X_{\mathrm{SAT}}, \vec{e}_a, \vec{d}_I, P\right)$$

$$X_{\mathrm{SAT}} = \begin{bmatrix} \vec{r}_I \\ \vec{v}_I \\ R_{\mathrm{BI}} \\ \vec{\omega}_B \\ \vec{\omega}_{\mathrm{RWA}} \end{bmatrix}$$

refer III. Satellite Dynamics Modeling for detailed explanation

$$X_{\mathrm{SAT}} = \begin{bmatrix} \vec{r}_I \\ \vec{v}_I \\ R_{\mathrm{BI}} \\ \vec{\omega}_B \\ \vec{\omega}_{\mathrm{RWA}} \end{bmatrix}$$

```
function [Xdot] = SATOdeFunctionHF(t,X,eaVec,distVec,P)
  INPUTS

  t ---------- Scalar time input, as required by Matlab's ODE function
               format.

  X ---------- Nx-by-1 SAT state, arranged as

               X = [rI',vI',RBI(1,1),RBI(2,1),...,RBI(2,3),RBI(3,3),...
                    omegaB',omegaVec']'

                rI = 3x1 position vector in I in meters
                vI = 3x1 velocity vector wrt I and in I, in meters/sec
               RBI = 3x3 attitude matrix from I to B frame
            omegaB = 3x1 angular rate vector of body wrt I, expressed in B
                     in rad/sec
          omegaVec = 4x1 vector of rotor angular rates, in rad/sec.
                     omegaVec(i) is the angular rate of the ith rotor.

     eaVec --- 4x1 vector of voltages applied to motors, in volts.  eaVec(i)
               is the constant voltage setpoint for the ith rotor.

   distVec --- 3x1 vector of constant disturbance forces acting on the SAT's
               center of mass, expressed in Newtons in I.

  OUTPUTS

  Xdot ------- Nx-by-1 time derivative of the input vector X
```

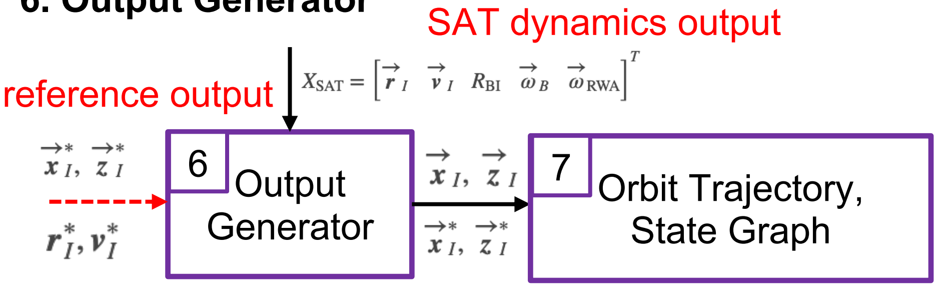
Generate oversampled, updated satellite dynamics using integrator function (ODE45 or RK4)

```
eak = voltageConverterSAT(NBk,omegaVec_k,P_voltage);

Xk = [r_k;v_k;RBI_k(:);omegaB_k;omegaVec_k];

[tVeck, xMatk] = ode45(@(t,X) SATOdeFunctionHF(t,X,eak,distMat_k,Parameters),tspan,Xk);
```

Then Store the simulation output `[tVeck, xMatk]`

V. Satellite Controller Modeling [5/6]

6. Output Generator



List of Available Output

1. **Satellite Trajectory with Body Axis Vector display**

$\Rightarrow$ Reference Trajectory and attitude : $r^*, \vec{x}_I^*, \vec{z}_I^*, \vec{y}_I^* (= \vec{z}_I^* \times \vec{x}_I^*)$

$\Rightarrow$ Output Trajectory and attitude : $r, \vec{x}_I = R_{BI}^T \begin{bmatrix} 1 \\ 0 \\ 0 \end{bmatrix}, \vec{y}_I = R_{BI}^T \begin{bmatrix} 0 \\ 1 \\ 0 \end{bmatrix}, \vec{z}_I = R_{BI}^T \begin{bmatrix} 0 \\ 0 \\ 1 \end{bmatrix}$

$\Rightarrow$ We can use **quiver** function for display trajectory and attitude vector

2. **Satellite Euler angle (Roll, Pitch, Yaw) display**

$\Rightarrow$ Reference Euler Angle : $R_{BI}^* = \begin{bmatrix} | & | & | \\ \vec{a} & \vec{b} & \vec{z}_I^* \\ | & | & | \end{bmatrix}^T, \begin{cases} \vec{b} = \dfrac{\vec{z}_I^* \times \vec{x}_I^*}{\left|\vec{z}_I^* \times \vec{x}_I^*\right|} & \text{, if } \vec{z}_I^* \times \vec{x}_I^* = 0 \Rightarrow \vec{b} = \vec{0} \\ \vec{a} = \vec{b} \times \vec{z}_I^* \end{cases}$

$$\phi^* = \sin^{-1}\left(R_{BI}^*(2,3)\right),\ \theta^* = \tan^{-1}\left(-\frac{R_{BI}^*(1,3)}{R_{BI}^*(3,3)}\right), \psi^* = \tan^{-1}\left(-\frac{R_{BI}^*(2,1)}{R_{BI}^*(2,2)}\right)$$

$\Rightarrow$ Output Euler Angle : Using R_{BI} from X_{SAT}

$$\phi = \sin^{-1}\left(R_{BI}(2,3)\right),\ \theta = \tan^{-1}\left(-\frac{R_{BI}(1,3)}{R_{BI}(3,3)}\right), \psi = \tan^{-1}\left(-\frac{R_{BI}(2,1)}{R_{BI}(2,2)}\right)$$

3. **Reaction Wheel Angular Rate display**

$\Rightarrow$ Use $\vec{\omega}_{RWA}$ from X_{SAT}

- We can make more visualized output from given dataset
- ECI to LLA map display, Pointing Display for GS Contact
- Sun Pointing Display, Nadir Pointing Display

V. Satellite Controller Modeling [6/6]

8. State Estimator

$$X_{SAT} = \begin{bmatrix} \vec{r}_I & \vec{v}_I & R_{BI} & \vec{\omega}_B & \vec{\omega}_{RWA} \end{bmatrix}^T$$

True dataset
(With Noise Model)

8 AOCS – Onboard EST

State Estimator

(Assume Perfect State Estimate Now on)
Function: Sensor Measurement & State Estimation Algorithm

Measurement

- GNSS
- Star Tracker
- IMU
- Sun Sensor
- Magnetometer
- TLE

Estimation

- UKF
- GP

$$X_{SAT} = \begin{bmatrix} \vec{r}_I & \vec{v}_I & R_{BI} & \vec{\omega}_B & \vec{\omega}_{RWA} \end{bmatrix}^T$$

Estimated dataset

- Currently, assuming we have perfect estimation, so estimated dataset is same with True Dataset with no noise model
- In the future research we will develop the onboard sensor measurement model and estimation model

- Since the satellite does not receive the perfectly accurate dataset, for high-fidelity simulation, we should model the noised / biased state signal and sensor dynamics model
- The satellite uses multiple sensors for trajectory and attitude (GNSS, Star Tracker, IMU, Sun Sensor, TLE Telemetry, Magnetometer, etc.)
- By sensor fusion technique like Unscented Kalman Filter, satellite optimizes the state signal
- By optimization methodology such as Gaussian Process or Bayesian Optimization, Satellite Develops innate dynamic model for advanced adaptive control

VI. Simulation Parameters Modeling

Epoch :

Mar 21, 2024

Satellite Initial Location :

SMA = 7640km

eccentricity $e = 1.6e-5$

inclination $i = 100.7$ deg

RAAN $\Omega = 90$ deg

AOP $\omega = 0$ deg

True Anomaly $\theta = 0$ deg

Designated Ground Station :

Austin, Texas :

Latitude = 30.285 deg

Longitude = −97.735 deg

Altitude = $70m$

Satellite Initial Operation Mode : Nadir

Nadir Mode Simulation Specification :

Total Elapsed Time : 3600 sec

Energy Transmission Mode Simulation Specification :

Total Elapsed Time : 1000 sec

Slant Range Calculation Include

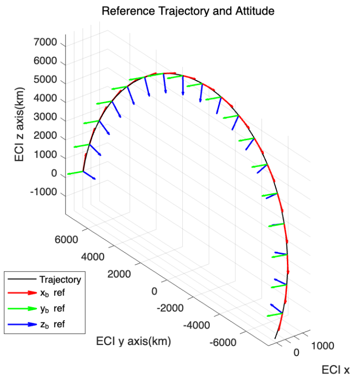
Sun Pointing Mode Simulation Specification

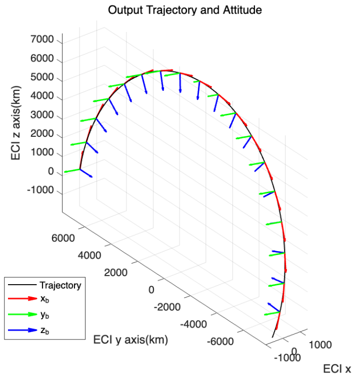
Total Elapsed Time : 3600 sec

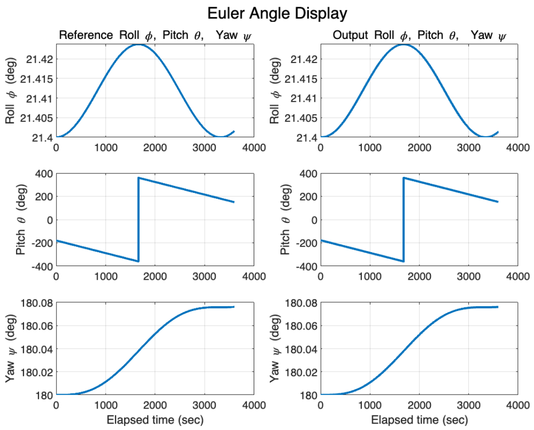
Sun vector on 3/21/2024 : $(1, 0, 0) \Rightarrow$ Changed by Day

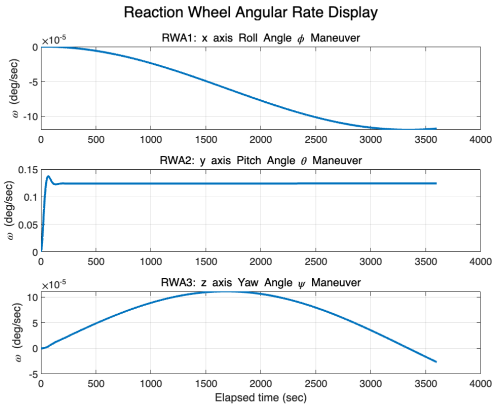
VII. Simulation Results and Discussion [1/3]

Result 1: Nadir Pointing Mode





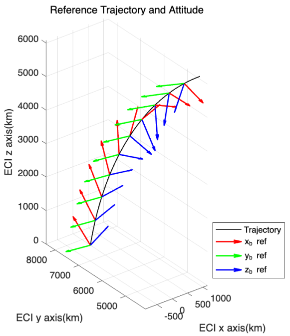


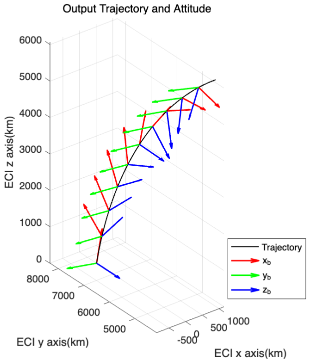


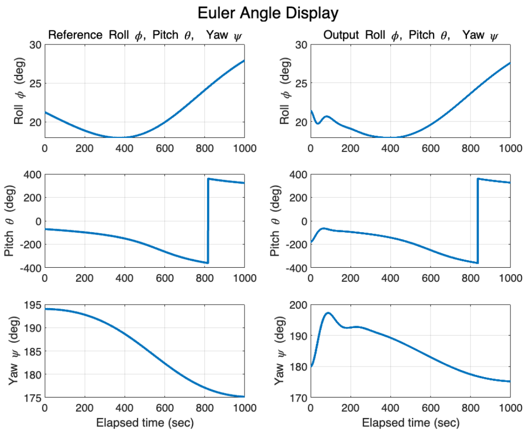
- The satellite has good tracking performance for Nadir Pointing Mode
- RWA #2 maintains steady-state non-zero angular velocity, which indicates there exist constant momentum for Nadir pointing mode
- RWA #1 and #3 almost remains zero rotational speed

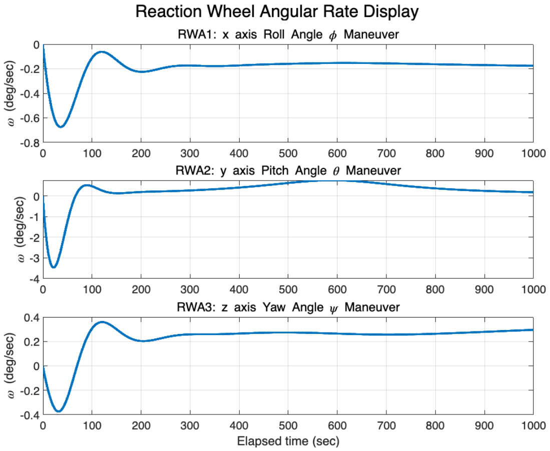
VII. Simulation Results and Discussion [2/3]

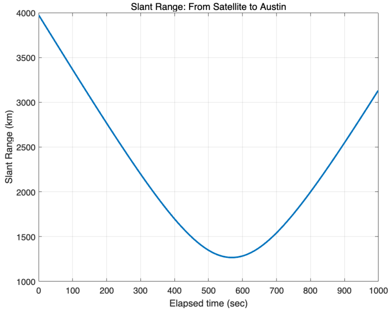
Result 2: Energy Transmission Mode







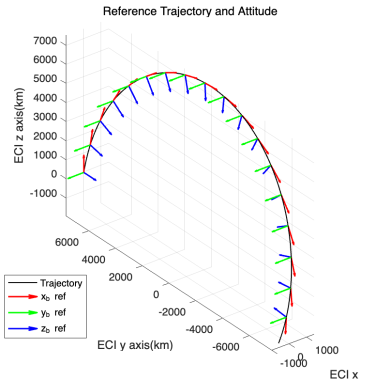


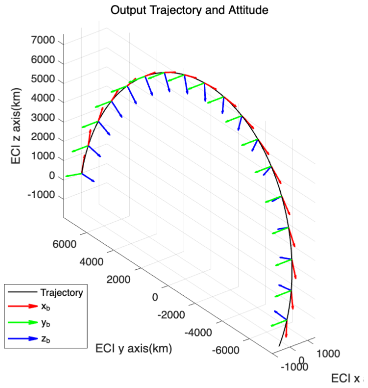


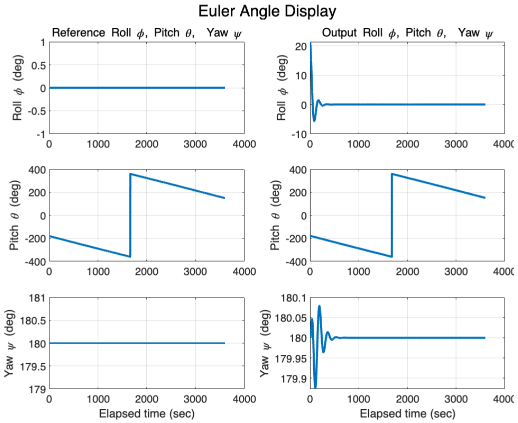
- The satellite requires 100 seconds to transit from Nadir pointing mode for Energy Transmission mode in given condition
- The satellite has good ground plant tracking performance after transition to Energy Transmission mode
- The Yaw-axis maneuver exists in Energy transmission mode
- The shortest slant range occurs around elapsed time 550 seconds

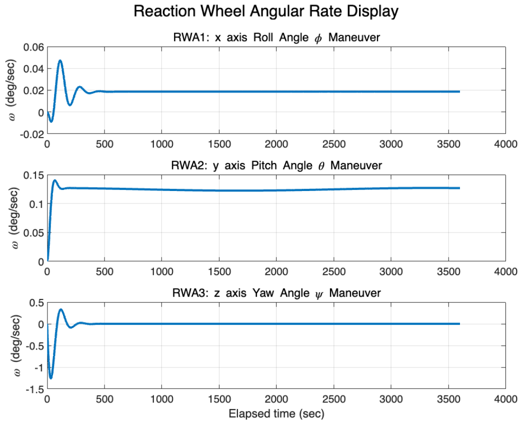
VII. Simulation Results and Discussion [3/3]

Result 3: Sun Pointing Mode









- The tracking performance of Sun-Pointing mode is good, maintaining constant roll and yaw angle.
- The satellite requires around 250 seconds for transition from nadir pointing mode to sun pointing mode in given simulation environment
- The RWA #1 and #2 maintains non-zero steady state angular rate, which indicates there exist steady-state angular momentum

VIII. Conclusion and Future Research

- In this session, we have discussed the Redstone-1's concept of operations and defined operation mode
- We have discussed the Redstone-1's attitude control mechanism
- We have discussed simulation of transition between normal operation modes in given initial condition
- In future research, we will investigate the satellite Sensor Measurement and Estimation Method

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