Homework #2

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Problem 1

```
function oe = rv2oe_Harris_Samantha(rPCI,vPCI,mu)

%-%------------------------------------------------------------------------%-
%-%------------ Code Template for Computing Orbital Elements ------------%-
%-%----------- from Cartesian Planet-Centered Inertial (PCI) ------------%-
%-%-------- and Cartesian Planet-Centered Inertial (PCI) Velocity -------%-
%-%------------------------------------------------------------------------%-
%-%--- PRIOR TO SUBMISSION THIS FUNCTION MUST BE RENAMED AS FOLLOWS: ----%-
%-%------------------------- FUNCTION HEADER LINE -----------------------%-
%-%---------- function oe = rv2oe_LastName_FirstName(rv,vv,mu) ----------%-
%-%------------------- NAME OF ACTUAL FUNCTION FILE ---------------------%-
%-%-------------------- rv2oe_LastName_FirstName.m ----------------------%-
%-%------------------------------------------------------------------------%-
%-%------------------------------------------------------------------------%-
% Inputs:                                                                  %-
%    rPCI:  Cartesian planet-centered inertial (PCI) position (3 by 1)    %-
%    vPCI:  Cartesian planet-centered inertial (PCI) velocity (3 by 1)    %-
%    mu:    gravitational parameter of centrally attacting body.          %-
% Outputs:  orbital elements                                              %-
%    oe(1): semi-major axis.                                               %-
%    oe(2): eccentricity.                                                  %-
%    oe(3): longitude of the ascending node (rad)                         %-
%    oe(4): inclination (rad)                                              %-
%    oe(5): argument of the periapsis (rad)                               %-
%    oe(6): true anomaly (rad)                                             %-
%-%------------------------------------------------------------------------%-

%-%------------------------------------------------------------------------%-
%-% ------------------ Format of Final Line of Code -------------------- %-
%-% oe = [a; e; Omega; inc; omega; nu];                                   %-
%-%------------------------------------------------------------------------%-
%-%--- IMPORTANT: THE OUTPUT oe MUST BE A COLUMN VECTOR OF LENGTH SIX ---%-
```

```
hv = cross(rPCI,vPCI);                             %angular momentum vector
h = norm(hv);                                      %magnitude of angular momentum
p = h^2/mu;                                        %semi-latus rectum
ev = cross(vPCI, hv)/mu - rPCI/norm(rPCI);  %eccentricity vector
e = norm(ev);                                      %eccentricity
a = p/(1-e^2);                                     %semi-major axis

px = ev/norm(ev);         %Perifocal bases x direction
pz = hv/norm(hv);         %Perifocal bases y direction
py = cross(pz,px);        %Perifocal bases z direction

I = [1 0 0;0 1 0; 0 0 1]; %Inertial Bases
Ix = I(:,1);              %Inertial bases x direction
Iy = I(:,2);              %Inertial bases y direction
Iz = I(:,3);              %Inertial bases z direction

%TRUE ANOMALY
nu = atan2(dot(rPCI, py),dot(rPCI, px));%pg 42 in notes
if nu < 0 %make sure the angle is [0,2pi]
   nu  = nu + 2*pi;
end
nudeg = rad2deg(nu);%convert from radians to degrees

n = cross(Iz, hv); %pg 39 in notes

%LONGITUDE OF THE ASCENDING NODE
Omega = atan2(dot(n, Iy),dot(n,Ix));
if Omega < 0 %make sure the angle is [0,2pi]
    Omega = Omega + 2*pi;
end
Omegadeg = rad2deg(Omega);%convert from radians to degrees
%go from I bases to N basis
TNI = [cos(Omega) -sin(Omega) 0;sin(Omega) cos(Omega) 0; 0 0 1];
N = I*TNI;
Nx = N(:,1);
Ny = N(:,2);
Nz = N(:,3);

%INCLINATION
inc = atan2(dot(-hv, Ny),dot(hv, Nz));%pg 40 in notes
incdeg = rad2deg(inc); %convert from radians to degrees

%ARGUMENT OF PERIAPSIS
omega = atan2(dot(-ev,cross(hv,n)), -h*dot(ev,n)) + pi;%pg 41 in notes
omegadeg = rad2deg(omega); %convert from radians to degrees
%put all orbital elements in a column vector
```

```
oe = [a; e; Omegadeg; incdeg; omegadeg;nudeg];
```

Problem 2

```
function [rPCI,vPCI]  = oe2rv_Harris_Samantha(oe,mu)

%-%----------------------------------------------------------------------%-
%-%-------------------- Code Template for Computing ---------------------%-
%-%--------- Cartesian Planet-Centered Inertial (PCI) Position ----------%-
%-%------- and Cartesian Planet-Centered Inertial (PCI) Velocity --------%-
%-%------------------------ from Orbital Elements -----------------------%-
%-%----------------------------------------------------------------------%-
%-%--- PRIOR TO SUBMISSION THIS FUNCTION MUST BE RENAMED AS FOLLOWS: ----%-
%-%------------------------- FUNCTION HEADER LINE -----------------------%-
%-%-------- function [rPCI,vPCI] = oe2rv_LastName_FirstName(oe,mu) ------%-
%-%------------------- NAME OF ACTUAL FUNCTION FILE ---------------------%-
%-%-------------------- oe2rv_LastName_FirstName.m ----------------------%-
%-%----------------------------------------------------------------------%-
%-%----------------------------------------------------------------------%-
%-% Input:  orbital elements             (6 by 1 column vector)         %-
%-%   oe(1): Semi-major axis.                                           %-
%-%   oe(2): Eccentricity.                                              %-
%-%   oe(3): Longitude of the ascending node Omega (rad)                       %-
%-%   oe(4): Inclination (rad)                                          %-
%-%   oe(5): Argument of the periapsis omega (rad)                             %-
%-%   oe(6): True anomaly (rad)                                         %-
%-%   mu:    Planet gravitational parameter     (scalar)                %-
%-% Outputs:                                                            %-
%-%   rPCI:  Planet-Centered Inertial (PCI) Cartesian position          %-
%-%          (3 by 1 column vector)                                     %-
%-%   vPCI:  Planet-Centered Inertial (PCI) Cartesian inertial velocity %-
%-%          (3 by 1 column vector)                                     %-
%-%----------------------------------------------------------------------%-

%-%----------------------------------------------------------------------%-
%-% ---------- Final Two Lines of Code (UNCOMMENT)                       %-
%-% rPCI = < PUT YOUR FINAL PCI POSITION HERE >                          %-
%-% vPCI = < PUT YOUR FINAL PCI INERTIAL VELOCITY HERE >                 %-
%-%----------------------------------------------------------------------%-

%assigning elements
```

```
a       = oe(1);             %semi-major axis
e       = oe(2);             %eccentricity
Omega   = deg2rad(oe(3));    %longitude of ascending node
inc     = deg2rad(oe(4));    %orbital inclination
omega   = deg2rad(oe(5));    %argument of the periapsis (rad)
nu      = deg2rad(oe(6));    %true anomaly

%transformation matrix tranforming from Reference Frame N to I
TNI = [cos(Omega) -sin(Omega) 0;sin(Omega) cos(Omega) 0; 0 0 1];
%transformation matrix tranforming from Reference Frame Q to N
TQN = [1 0 0; 0 cos(inc) -sin(inc); 0 sin(inc) cos(inc)];
%transformation matrix tranforming from Reference Frame P to Q
TPQ = [cos(omega) -sin(omega) 0; sin(omega) cos(omega) 0; 0 0 1];
%transformation from P to I
TPI = TNI*TQN*TPQ;

r = a*(1-e^2)/(1+e*cos(nu));        %orbit equation
rvp = [r*cos(nu); r*sin(nu);0];     %position of spacecraft in perifocal frame
p = a*(1-e^2);                      %semi-latus rectum
vvp = sqrt(mu/p)*[-sin(nu); e + cos(nu); 0]; %velocity of spacecraft in p frame

rPCI = TPI*rvp; %Planet-Centered Inertial (PCI) Cartesian position
vPCI = TPI*vvp; %Planet-Centered Inertial (PCI) Cartesian velocity
```

Problem 3

It is true that $\mathbf{n} = \mathbf{0}$ when the equatorial plane and the orbital plane are in the same plane. Since $cos(\Omega) = \mathbf{n} \cdot \mathbf{I}_x / ||\mathbf{n}||$ The longitude of the ascending node will be undefined if the equatorial plane and orbit plane line up. In addition, the inclination is not defined when in rectilinear orbit (flying vertical towards the planet).

Problem 4

The rate of change of specific angular momentum:

$$\frac{^I d}{dt}(^I\mathbf{h}) = \mathbf{r} \times (-\frac{\mu}{r^3}\mathbf{r}) = -\frac{\mu}{r^3}\mathbf{r} \times \mathbf{r} = \mathbf{0}$$

The rate of change of eccentricity vector:
From equation (1.33)

$$\frac{^I\mathbf{v} \times {}^I\mathbf{h}}{\mu} - \frac{\mathbf{r}}{r} = \mathbf{e}$$

multiplying by μ

$$^{I}\mathbf{v} \times {}^{I}\mathbf{h} - \frac{\mu \mathbf{r}}{r} = \mu \mathbf{e}$$

$$\frac{^{I}d}{dt}(^{I}\mathbf{v} \times {}^{I}\mathbf{h} - \frac{\mu \mathbf{r}}{r}) = \frac{^{I}d}{dt}\mu \mathbf{e}$$

From equation (1.31)

$$\frac{^{I}d}{dt}(^{I}\mathbf{v} \times {}^{I}\mathbf{h} - \frac{\mu \mathbf{r}}{r}) = \mathbf{0}$$

$$\mathbf{0} = \frac{^{I}d}{dt}\mu \mathbf{e}$$

$$\mathbf{0} = \frac{^{I}d}{dt}\mathbf{e}$$

The rate of change of the line of nodes:

$$\mathbf{n} = \mathbf{I}_z \times {}^{I}\mathbf{h}$$

$$\frac{^{I}d}{dt}\mathbf{n} = \frac{^{I}d}{dt}(\mathbf{I}_z \times {}^{I}\mathbf{h})$$

$$\frac{^{I}d}{dt}\mathbf{n} = (\frac{^{I}d}{dt}\mathbf{I}_z \times {}^{I}\mathbf{h}) + (\mathbf{I}_z \times \frac{^{I}d}{dt}{}^{I}\mathbf{h})$$

$$\frac{^{I}d}{dt}\mathbf{n} = (\mathbf{0} \times {}^{I}\mathbf{h}) + (\mathbf{I}_z \times \mathbf{0})$$

$$\frac{^{I}d}{dt}\mathbf{n} = \mathbf{0}$$

Problems 5-10

Solutions

```
--------------------------------------------------------------
--------------------------------------------------------------
                       Question (2-5):
--------------------------------------------------------------
                          Part a):
--------------------------------------------------------------
```

```
Orbital elements:
semi-major axis (a)   =   1.277396 AU
eccentricity (e)   =   0.251185
longitude of the ascending node (Omega)  = 315.000000 degrees
inclination (inc)   =  18.074455 degrees
argument of the periapsis (omega)   = 106.879105 degrees
true anomaly (nu)   = 338.938457 degrees
-------------------------------------------------------------
                            Part b):
-------------------------------------------------------------
orbital period (tao)   =   9.071274 TU
-------------------------------------------------------------
                            Part c):
-------------------------------------------------------------
semi-latus rectum (p)   =   1.196800 AU
-------------------------------------------------------------
                            Part d):
-------------------------------------------------------------
Specific Angular Momentum (hv) [AU^2/TU]:
   [     -0.24000000,     -0.24000000,      1.04000000]
Magnitude of the Specific Angular Momentum (h) [AU^2/TU]:
                                           1.09398355
-------------------------------------------------------------
                            Part e):
-------------------------------------------------------------
Specific Mechanical Energy (E)   =  -0.638698 AU^2/s^2
-------------------------------------------------------------
The periapsis radius of this asteroid is   0.956533 AU and the
radius of Earth is 1 AU. Therefore, the asteroid is potentially
hazardous to Earth.
-------------------------------------------------------------
-------------------------------------------------------------
-------------------------------------------------------------
                        Question (2-6):
-------------------------------------------------------------
Position of Spacecraft in SCI coordinates [AU]:
   [      0.70000000,      0.60000000,      0.30000000]
Inertial Velocity of Spacecraft in SCI coordinates [AU/TU]:
   [     -0.80000000,      0.80000000,      0.00000000]
-------------------------------------------------------------
-------------------------------------------------------------
-------------------------------------------------------------
                        Question (2-7):
-------------------------------------------------------------
GIVEN:
Orbital elements:
```

```
semi-major axis (a)   = 15307.548000 km
eccentricity (e)   =   0.700000
longitude of the ascending node (Omega)  = 194.000000 degrees
inclination (inc)   =  39.000000 degrees
argument of the periapsis (omega)   =  85.000000 degrees
true anomaly (nu)   =  48.000000 degrees
-------------------------------------------------------------
ANSWER:
Position of Spacecraft in ECI coordinates [km]:
   [   4249.24395473,  -2054.84062287,   2446.99585787]
Inertial Velocity of Spacecraft in ECI coordinates [km/s]:
   [      9.07117614,      5.81566502,     -2.79245828]
-------------------------------------------------------------
CHECK:
Orbital elements:
semi-major axis (a)   = 15307.548000 km
eccentricity (e)   =   0.700000
longitude of the ascending node (Omega)  = 194.000000 degrees
inclination (inc)   =  39.000000 degrees
argument of the periapsis (omega)   =  85.000000 degrees
true anomaly (nu)   =  48.000000 degrees
-------------------------------------------------------------
-------------------------------------------------------------
-------------------------------------------------------------
                        Question (2-8):
-------------------------------------------------------------
GIVEN:
Orbital elements:
semi-major axis (a)   = 19133.333000 km
eccentricity (e)   =   0.500000
longitude of the ascending node (Omega)  =  30.000000 degrees
inclination (inc)   =  45.000000 degrees
argument of the periapsis (omega)   =  45.000000 degrees
true anomaly (nu)   =   0.000000 degrees
-------------------------------------------------------------
ANSWER:
Position of Spacecraft in ECI coordinates [km]:
   [   3466.69624109,   7524.81548702,   4783.33325000]
Inertial Velocity of Spacecraft in ECI coordinates [km/s]:
   [     -6.81755776,      0.62817226,      3.95279202]
-------------------------------------------------------------
CHECK:
Orbital elements:
semi-major axis (a)   = 19133.333000 km
eccentricity (e)   =   0.500000
longitude of the ascending node (Omega)  =  30.000000 degrees
```

```
inclination (inc)   =  45.000000 degrees
argument of the periapsis (omega)   =  45.000000 degrees
true anomaly (nu)   =   0.000000 degrees
-------------------------------------------------------------
-------------------------------------------------------------
-------------------------------------------------------------
                        Question (2-9):
-------------------------------------------------------------
GIVEN:
Orbital elements:
semi-major axis (a)   = 20000.000000 km
eccentricity (e)   =   0.450000
longitude of the ascending node (Omega)  =  59.000000 degrees
inclination (inc)   =  27.000000 degrees
argument of the periapsis (omega)   =  94.000000 degrees
true anomaly (nu)   =  58.000000 degrees
-------------------------------------------------------------
ANSWER:
Position of Spacecraft in ECI coordinates [km]:
   [ -10474.46193267,  -6972.51466993,   2744.94389648]
Inertial Velocity of Spacecraft in ECI coordinates [km/s]:
   [      1.12638675,     -6.03283073,     -2.07511343]
-------------------------------------------------------------
CHECK:
Orbital elements:
semi-major axis (a)   = 20000.000000 km
eccentricity (e)   =   0.450000
longitude of the ascending node (Omega)  =  59.000000 degrees
inclination (inc)   =  27.000000 degrees
argument of the periapsis (omega)   =  94.000000 degrees
true anomaly (nu)   =  58.000000 degrees
-------------------------------------------------------------
-------------------------------------------------------------
-------------------------------------------------------------
                        Question (2-10):
-------------------------------------------------------------
GIVEN: (in canonical units)
Orbital elements:
semi-major axis (a)   =   1.600000
eccentricity (e)   =   0.400000
longitude of the ascending node (Omega)  = 287.000000 degrees
inclination (inc)   =  46.000000 degrees
argument of the periapsis (omega)   =  28.000000 degrees
true anomaly (nu)   = 139.000000 degrees
-------------------------------------------------------------
ANSWER:
```

```
Position of Spacecraft in PCI coordinates :
   [     -0.26075057,      1.88182968,      0.31152557]
Inertial Velocity of Spacecraft in PCI coordinates :
   [     -0.46004409,      0.23163947,     -0.38544253]
-------------------------------------------------------------
CHECK:
Orbital elements:
semi-major axis (a)   =   1.600000
eccentricity (e)   =   0.400000
longitude of the ascending node (Omega)  = 287.000000 degrees
inclination (inc)   =  46.000000 degrees
argument of the periapsis (omega)   =  28.000000 degrees
true anomaly (nu)   = 139.000000 degrees
```

Script

```
clc
clear all

%Chapter 2 Problems 5 - 10
%2-5
mu5 = 1;
rv5 = [.7;.6;.3];
vv5 = [-.8;.8;0];

oe5 = rv2oe_Harris_Samantha(rv5,vv5,mu5);
a = oe5(1);
e = oe5(2);

%orbital period
tao = 2*pi*sqrt(a^3/mu5);
%semi-latus rectum
p = a*(1-e^2);
%specific angular momentum
hv = cross(rv5, vv5);
%magnitude of specific angular momentum
h = norm(hv);
%specific mechanial energy
E = -mu5/2*a;
%periapsis radius
rp = p/(1+e);

%2-6
[rSCI6,vSCI6]  = oe2rv_Harris_Samantha(oe5,mu5);

%2-7
```

```
muEarth = 398600;
a7 = 15307.548;
e7 = .7;
Omega7 = 194;
inc7 = 39;
omega7 = 85;
nu7 = 48;

oe7 = [a7;e7;Omega7;inc7;omega7;nu7];

[rECI7,vECI7]  = oe2rv_Harris_Samantha(oe7,muEarth);
oe7c = rv2oe_Harris_Samantha(rECI7,vECI7,muEarth);

%2-8
a8 = 19133.333;
e8 = .5;
Omega8 = 30;
inc8 = 45;
omega8 = 45;
nu8 = 0;

oe8 = [a8;e8;Omega8;inc8;omega8;nu8];

[rECI8,vECI8]  = oe2rv_Harris_Samantha(oe8,muEarth);
oe8c = rv2oe_Harris_Samantha(rECI8,vECI8,muEarth);

%2-9
a9 = 20000;
e9 = .45;
Omega9 = 59;
inc9 = 27;
omega9 = 94;
nu9 = 58;

oe9 = [a9;e9;Omega9;inc9;omega9;nu9];

[rECI9,vECI9]  = oe2rv_Harris_Samantha(oe9,muEarth);
oe9c = rv2oe_Harris_Samantha(rECI9,vECI9,muEarth);

%2-10
mu10 = 1;
a10 = 1.6;
e10 = .4;
Omega10 = 287;
inc10 = 46;
omega10 = 28;
```

```
nu10 = 139;

oe10 = [a10;e10;Omega10;inc10;omega10;nu10];

[rPCI10,vPCI10]  = oe2rv_Harris_Samantha(oe10,mu10);
oe10c = rv2oe_Harris_Samantha(rPCI10,vPCI10,mu10);

fprintf('-------------------------------------------------------------\n');
fprintf('-------------------------------------------------------------\n');
fprintf('                       Question (2-5):                        \n');
fprintf('-------------------------------------------------------------\n');
fprintf('                           Part a):                         \n');
fprintf('-------------------------------------------------------------\n');
fprintf('Orbital elements:\n');
fprintf('semi-major axis (a) \t\t\t = %10.6f AU\n',oe5(1));
fprintf('eccentricity (e) \t\t\t = %10.6f\n',oe5(2));
fprintf('longitude of the ascending node (Omega)  = %10.6f degrees\n',oe5(3));
fprintf('inclination (inc) \t\t\t = %10.6f degrees\n',oe5(4));
fprintf('argument of the periapsis (omega) \t = %10.6f degrees\n',oe5(5));
fprintf('true anomaly (nu) \t\t\t = %10.6f degrees\n',oe5(6));
fprintf('-------------------------------------------------------------\n');
fprintf('                           Part b):                         \n');
fprintf('-------------------------------------------------------------\n');
fprintf('orbital period (tao) \t\t\t = %10.6f TU\n',tao);
fprintf('-------------------------------------------------------------\n');
fprintf('                           Part c):                         \n');
fprintf('-------------------------------------------------------------\n');
fprintf('semi-latus rectum (p) \t\t\t = %10.6f AU\n',p);
fprintf('-------------------------------------------------------------\n');
fprintf('                           Part d):                         \n');
fprintf('-------------------------------------------------------------\n');
fprintf('Specific Angular Momentum (hv) [AU^2/TU]:\n   ');
fprintf('[%16.8f,%16.8f,%16.8f]\n',hv);
fprintf('Magnitude of the Specific Angular Momentum (h) [AU^2/TU]:\n');
fprintf('                                  \t %16.8f\n',h);
fprintf('-------------------------------------------------------------\n');
fprintf('                           Part e):                         \n');
fprintf('-------------------------------------------------------------\n');
fprintf('Specific Mechanical Energy (E) \t = %10.6f AU^2/s^2\n',E);
fprintf('-------------------------------------------------------------\n');
fprintf('The periapsis radius of this asteroid is %10.6f AU and the\n', rp);
fprintf('radius of Earth is 1 AU. Therefore, the asteroid is potentially\n');
fprintf('hazardous to Earth.\n');
fprintf('-------------------------------------------------------------\n');
fprintf('-------------------------------------------------------------\n');
fprintf('-------------------------------------------------------------\n');
```

```
fprintf('                             Question (2-6):                             \n');
fprintf('------------------------------------------------------------\n');
fprintf('Position of Spacecraft in SCI coordinates [AU]:\n   ');
fprintf('[%16.8f,%16.8f,%16.8f]\n',rSCI6);
fprintf('Inertial Velocity of Spacecraft in SCI coordinates [AU/TU]:\n   ');
fprintf('[%16.8f,%16.8f,%16.8f]\n',vSCI6);
fprintf('------------------------------------------------------------\n');
fprintf('------------------------------------------------------------\n');
fprintf('------------------------------------------------------------\n');
fprintf('                             Question (2-7):                             \n');
fprintf('------------------------------------------------------------\n');
fprintf('GIVEN:\n');
fprintf('Orbital elements:\n');
fprintf('semi-major axis (a) \t\t\t = %10.6f km \n',oe7(1));
fprintf('eccentricity (e) \t\t\t = %10.6f\n',oe7(2));
fprintf('longitude of the ascending node (Omega)  = %10.6f degrees\n',oe7(3));
fprintf('inclination (inc) \t\t\t = %10.6f degrees\n',oe7(4));
fprintf('argument of the periapsis (omega) \t = %10.6f degrees\n',oe7(5));
fprintf('true anomaly (nu) \t\t\t = %10.6f degrees\n',oe7(6));
fprintf('------------------------------------------------------------\n');
fprintf('ANSWER:\n');
fprintf('Position of Spacecraft in ECI coordinates [km]:\n   ');
fprintf('[%16.8f,%16.8f,%16.8f]\n',rECI7);
fprintf('Inertial Velocity of Spacecraft in ECI coordinates [km/s]:\n   ');
fprintf('[%16.8f,%16.8f,%16.8f]\n',vECI7);
fprintf('------------------------------------------------------------\n');
fprintf('CHECK:\n');
fprintf('Orbital elements:\n');
fprintf('semi-major axis (a) \t\t\t = %10.6f km \n',oe7c(1));
fprintf('eccentricity (e) \t\t\t = %10.6f\n',oe7c(2));
fprintf('longitude of the ascending node (Omega)  = %10.6f degrees\n',oe7c(3));
fprintf('inclination (inc) \t\t\t = %10.6f degrees\n',oe7c(4));
fprintf('argument of the periapsis (omega) \t = %10.6f degrees\n',oe7c(5));
fprintf('true anomaly (nu) \t\t\t = %10.6f degrees\n',oe7c(6));
fprintf('------------------------------------------------------------\n');
fprintf('------------------------------------------------------------\n');
fprintf('------------------------------------------------------------\n');
fprintf('                             Question (2-8):                             \n');
fprintf('------------------------------------------------------------\n');
fprintf('GIVEN:\n');
fprintf('Orbital elements:\n');
fprintf('semi-major axis (a) \t\t\t = %10.6f km \n',oe8(1));
fprintf('eccentricity (e) \t\t\t = %10.6f\n',oe8(2));
fprintf('longitude of the ascending node (Omega)  = %10.6f degrees\n',oe8(3));
fprintf('inclination (inc) \t\t\t = %10.6f degrees\n',oe8(4));
fprintf('argument of the periapsis (omega) \t = %10.6f degrees\n',oe8(5));
```

```
fprintf('true anomaly (nu) \t\t\t = %10.6f degrees\n',oe8(6));
fprintf('-------------------------------------------------------------\n');
fprintf('ANSWER:\n');
fprintf('Position of Spacecraft in ECI coordinates [km]:\n   ');
fprintf('[%16.8f,%16.8f,%16.8f]\n',rECI8);
fprintf('Inertial Velocity of Spacecraft in ECI coordinates [km/s]:\n   ');
fprintf('[%16.8f,%16.8f,%16.8f]\n',vECI8);
fprintf('-------------------------------------------------------------\n');
fprintf('CHECK:\n');
fprintf('Orbital elements:\n');
fprintf('semi-major axis (a) \t\t\t = %10.6f km \n',oe8c(1));
fprintf('eccentricity (e) \t\t\t = %10.6f\n',oe8c(2));
fprintf('longitude of the ascending node (Omega)  = %10.6f degrees\n',oe8c(3));
fprintf('inclination (inc) \t\t\t = %10.6f degrees\n',oe8c(4));
fprintf('argument of the periapsis (omega) \t = %10.6f degrees\n',oe8c(5));
fprintf('true anomaly (nu) \t\t\t = %10.6f degrees\n',oe8c(6));
fprintf('-------------------------------------------------------------\n');
fprintf('-------------------------------------------------------------\n');
fprintf('-------------------------------------------------------------\n');
fprintf('                        Question (2-9):                         \n');
fprintf('-------------------------------------------------------------\n');
fprintf('GIVEN:\n');
fprintf('Orbital elements:\n');
fprintf('semi-major axis (a) \t\t\t = %10.6f km \n',oe9(1));
fprintf('eccentricity (e) \t\t\t = %10.6f\n',oe9(2));
fprintf('longitude of the ascending node (Omega)  = %10.6f degrees\n',oe9(3));
fprintf('inclination (inc) \t\t\t = %10.6f degrees\n',oe9(4));
fprintf('argument of the periapsis (omega) \t = %10.6f degrees\n',oe9(5));
fprintf('true anomaly (nu) \t\t\t = %10.6f degrees\n',oe9(6));
fprintf('-------------------------------------------------------------\n');
fprintf('ANSWER:\n');
fprintf('Position of Spacecraft in ECI coordinates [km]:\n   ');
fprintf('[%16.8f,%16.8f,%16.8f]\n',rECI9);
fprintf('Inertial Velocity of Spacecraft in ECI coordinates [km/s]:\n   ');
fprintf('[%16.8f,%16.8f,%16.8f]\n',vECI9);
fprintf('-------------------------------------------------------------\n');
fprintf('CHECK:\n');
fprintf('Orbital elements:\n');
fprintf('semi-major axis (a) \t\t\t = %10.6f km \n',oe9c(1));
fprintf('eccentricity (e) \t\t\t = %10.6f\n',oe9c(2));
fprintf('longitude of the ascending node (Omega)  = %10.6f degrees\n',oe9c(3));
fprintf('inclination (inc) \t\t\t = %10.6f degrees\n',oe9c(4));
fprintf('argument of the periapsis (omega) \t = %10.6f degrees\n',oe9c(5));
fprintf('true anomaly (nu) \t\t\t = %10.6f degrees\n',oe9c(6));
fprintf('-------------------------------------------------------------\n');
fprintf('-------------------------------------------------------------\n');
```

```
fprintf('--------------------------------------------------------------\n');
fprintf('                          Question (2-10):                        \n');
fprintf('--------------------------------------------------------------\n');
fprintf('GIVEN: (in canonical units)\n');
fprintf('Orbital elements:\n');
fprintf('semi-major axis (a) \t\t\t = %10.6f \n',oe10(1));
fprintf('eccentricity (e) \t\t\t = %10.6f\n',oe10(2));
fprintf('longitude of the ascending node (Omega)  = %10.6f degrees\n',oe10(3));
fprintf('inclination (inc) \t\t\t = %10.6f degrees\n',oe10(4));
fprintf('argument of the periapsis (omega) \t = %10.6f degrees\n',oe10(5));
fprintf('true anomaly (nu) \t\t\t = %10.6f degrees\n',oe10(6));
fprintf('--------------------------------------------------------------\n');
fprintf('ANSWER:\n');
fprintf('Position of Spacecraft in PCI coordinates :\n   ');
fprintf('[%16.8f,%16.8f,%16.8f]\n',rPCI10);
fprintf('Inertial Velocity of Spacecraft in PCI coordinates :\n   ');
fprintf('[%16.8f,%16.8f,%16.8f]\n',vPCI10);
fprintf('--------------------------------------------------------------\n');
fprintf('CHECK:\n');
fprintf('Orbital elements:\n');
fprintf('semi-major axis (a) \t\t\t = %10.6f \n',oe10c(1));
fprintf('eccentricity (e) \t\t\t = %10.6f\n',oe10c(2));
fprintf('longitude of the ascending node (Omega)  = %10.6f degrees\n',oe10c(3));
fprintf('inclination (inc) \t\t\t = %10.6f degrees\n',oe10c(4));
fprintf('argument of the periapsis (omega) \t = %10.6f degrees\n',oe10c(5));
fprintf('true anomaly (nu) \t\t\t = %10.6f degrees\n',oe10c(6));
```