NumericalPropagationWithStopEvent 4.4

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```
public class NumericalPropagationWithStopEvent {

    public static void main(String[] args) throws PatriusException,
IOException, URISyntaxException {

        // Patrius Dataset initialization (needed for example to get the UTC
time
        PatriusDataset.addResourcesFromPatriusDataset() ;

        // Recovery of the UTC time scale using a "factory" (not to duplicate
such unique object)
        final TimeScale TUC = TimeScalesFactory.getUTC();

        // Date of the orbit given in UTC time scale)
        final AbsoluteDate date = new AbsoluteDate("2010-01-01T12:00:00.000",
TUC);

        // Getting the frame with wich will defined the orbit parameters
        // As for time scale, we will use also a "factory".
        final Frame GCRF = FramesFactory.getGCRF();

        // Initial orbit
        final double sma = 7200.e+3;
        final double exc = 0.02;
        final double per = sma*(1.-exc);
        final double apo = sma*(1.+exc);
        final double inc = FastMath.toRadians(98.);
        final double pa = FastMath.toRadians(0.);
        final double raan = FastMath.toRadians(0.);
        final double anm = FastMath.toRadians(180.);
        final double MU = Constants.WGS84_EARTH_MU;

        final ApsisRadiusParameters par = new ApsisRadiusParameters(per, apo,
inc, pa, raan, anm, PositionAngle.MEAN, MU);
        final Orbit iniOrbit = new ApsisOrbit(par, GCRF, date);

        // We create a spacecratftstate
        final SpacecraftState iniState = new SpacecraftState(iniOrbit);

        // Initialization of the Runge Kutta integrator with a 2 s step
        final double pasRk = 2.;
        final FirstOrderIntegrator integrator = new
ClassicalRungeKuttaIntegrator(pasRk);
```

```
        // Initialization of the propagator
        final NumericalPropagator propagator = new
NumericalPropagator(integrator);
        propagator.resetInitialState(iniState);

        // Forcing integration using cartesian equations
        propagator.setOrbitType(OrbitType.CARTESIAN);

//SPECIFIC
        // Definition of the Earth ellipsoid
        final Frame ITRF = FramesFactory.getITRF();
        final double AE = Constants.WGS84_EARTH_EQUATORIAL_RADIUS;
        final GeometricBodyShape EARTH = new ExtendedOneAxisEllipsoid(AE,
Constants.WGS84_EARTH_FLATTENING, ITRF, "EARTH");

        // Adding an altitude stop event
        final double endAlt = 750.e+3;
        final AltitudeDetector stopEvent = new AltitudeDetector(endAlt,
EARTH);
        propagator.addEventDetector(stopEvent);
//SPECIFIC

        // Propagating on one orbital period
        final double dt = iniOrbit.getKeplerianPeriod();
        final AbsoluteDate finalDate = date.shiftedBy(dt);
        final SpacecraftState finalState = propagator.propagate(finalDate);
        final Orbit finalOrbit = finalState.getOrbit();

        // Get geodetic coordinates (altitude, latitude, longitude)
        final GeodeticPoint iniGeodeticPoint =
EARTH.transform(iniOrbit.getPVCoordinates().getPosition(), ITRF, date);
        final GeodeticPoint finalGeodeticPoint =
EARTH.transform(finalOrbit.getPVCoordinates().getPosition(), ITRF, date);

        System.out.println();
        iniOrbit.getPVCoordinates(ITRF);
        System.out.println("Initial altitude =
"+iniGeodeticPoint.getAltitude()/1000.+" km");
        System.out.println("New date = "+finalOrbit.getDate().toString(TUC)+"
deg");
        System.out.println("Final altitude =
"+finalGeodeticPoint.getAltitude()/1000.+" km");

    }

}
```

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- Main differences between V4.8 and V4.7
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- Dernière modification de cette page le 3 octobre 2019 à 12:54.