

NumericalPropagationWithImpulsiveManeuver 4.5.1

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```
public class NumericalPropagationWithImpulsiveManeuver {

    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 = 0.01;
        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(0.);
        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);

//SPECIFIC
        // Creating a mass model (see also specific example)
        final AssemblyBuilder builder = new AssemblyBuilder();
        final double iniMass = 1000.;
        builder.addMainPart("MAIN");
        builder.addProperty(new MassProperty(iniMass), "MAIN");
```

```

        final Assembly assembly = builder.returnAssembly();
        final MassProvider mm = new MassModel(assembly);

        // We create a spacecraft state
        final SpacecraftState iniState = new SpacecraftState(iniOrbit, mm);
//SPECIFIC

        // 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
        // Event corresponding to the criteria to trigger the impulsive
maneuver
        final EventDetector event = new DateDetector(date.shiftedBy(10.));
        // Creation of the impulsive maneuver
        final Vector3D deltaV = new Vector3D(20., 0., 0.);
        final double isp = 300.;
        final ImpulseManeuver imp = new ImpulseManeuver(event, deltaV, isp,
mm, "MAIN", LOFType.TNW);
        // Adding the impulsive maneuver
        propagator.addEventDetector(imp);
        // Adding additional state (change name add to set for V3.3)
        propagator.setMassProviderEquation(mm);

//SPECIFIC

        // Propagating 100s
        final double dt = 100.;
        final AbsoluteDate finalDate = date.shiftedBy(dt);
        final SpacecraftState finalState = propagator.propagate(finalDate);
        final Orbit finalOrbit = finalState.getOrbit();

        // Printing new date and semi major axis
        System.out.println();
        System.out.println("Initial semi major axis =
"+iniOrbit.getA()/1000.+ " km");
        System.out.println("New date = "+finalOrbit.getDate().toString(TUC)+
deg");
        System.out.println("Final semi major axis =
"+finalOrbit.getA()/1000.+ " km");
        // Printing mass

```

```
        System.out.println();
        System.out.println("Mass = "+finalState.getMass("MAIN")+ " kg");

    }

}
```

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