

NumericalPropagationWithMass 4.4

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```
public class NumericalPropagationWithMass {

    public static void main(String[] args) throws PatriusException,
    IOException, ParseException, 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
        // Mass model using an Assembly
        final AssemblyBuilder builder = new AssemblyBuilder();

        // Initial mass
        final double dryMass = 1000.;
        builder.addMainPart("MAIN");
        builder.addProperty(new MassProperty(dryMass), "MAIN");
```

```

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

    // We create a spacecraftstate
    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);

//SPECIFIC
    // Adding additional state (change name add to set for V3.3)
    propagator.setMassProviderEquation(mm);
//SPECIFIC

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

    // Propagating 1000s
    final double dt = 1000.;
    final AbsoluteDate finalDate = date.shiftedBy(dt);
    final SpacecraftState finalState = propagator.propagate(finalDate);
    final Orbit finalOrbit = finalState.getOrbit();

    // Printing new date 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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