EPHEMERIS OF A HIGHLY ECCENTRIC ORBIT:
EXPLORER 28

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Abstract. An ephemeris has been obtained for Explorer 28 (IMP 3) which agrees well with 2 years of radio observations and with SAO observations a year later. This ephemeris is generated over the 3 year lifetime by a numerical integration method utilizing a set of initial conditions at launch and without requiring further differential correction. Because highly eccentric orbits are difficult to compute with acceptable accuracy and because a long continuous arc has been obtained which compares with actual data to a known precision, this ephemeris may be used as a standard for computing highly eccentric orbits in the Earth-Moon system.

Orbit improvement was used to obtain the initial conditions which generated the ephemeris. This improvement was based on correcting the energy by adjusting the semimajor axis to match computed times of perigee passage with the observed. This procedure may generate errors in semimajor axis to compensate for model errors in the energy; however this compensation error is also implicit in orbit determination itself.

1. Introduction

Highly eccentric orbits offer unusual opportunities for fitting long arcs with high precision. One major advantage is that a high eccentricity orbit is seldom, or never, affected by the earth’s atmosphere. Since the atmospheric drag force is poorly predicted, the uncertainties in close Earth circular orbits have to be assumed to arise from this source, and indeed, close Earth orbits are often used to derive the density of the atmosphere. In the case of high eccentricity orbits, however, precise determinations can be obtained and extrapolated forward with good accuracy.

A second major advantage is provided by the good resolution of the perigee position of the orbit. Because the raw data will yield a good definition of the position of perigee, the time of perigee passage becomes an important parameter in the orbit improvement. This parameter has the additional benefit of being readily derived from both the reduced observational data and the numerical computation, thus facilitating comparisons between the two. Further, the investigation of the long term behavior of this parameter yields some insights which may not be clearly visible in short arc fitting.

IMP 3 (Explorer 28) satellite is a good example of a high eccentricity orbit. Radio tracking has been obtained for over 120 orbits, that is, 2 years of the 3 year lifetime. This paper discusses a technique of improving the orbit by finding a long arc which is consistent with the available short arc data. An ephemeris has been obtained which gives a good fit to the available data (time of perigee passage). This ephemeris, when extrapolated ahead one year, proved capable of making predictions of accuracy such that SAO’s Baker-Nunn network was able to acquire the satellite. The technique of obtaining a good fitting ephemeris is discussed in this paper.

Earth satellites in highly eccentric orbits (e > 0.9) are subject to large and erratic
perturbations by the Moon, as well as a strong solar perturbation. Also, the Earth's oblateness term may cause a serious computational error, even though the secular effect of the oblateness is quite small. It can be difficult to distinguish between a legitimate perturbation and a numerical error introduced by improper selection of integration interval or a programming error.

The IMP 3 (Explorer 28) orbit provides a good sample of the major perturbational effects in the Earth-Moon system. Since there is a total of 2 years data with which a numerical integration has been compared, it is suggested that the ephemeris of IMP 3 serve as a computational standard. For most applications the full accuracy of the ephemeris will not be required; however, a program comparison with the IMP 3 ephemeris will test the ability of the program under test to compute an orbit with high eccentricity and large semimajor axis in the Earth-Moon system within desired accuracy requirements.

The general history of the satellite is discussed in Section 2. Section 3 gives a summary of the perturbations, the characteristic changes of specific orbital elements and the origins of the changes. Section 4 describes the method of computation and discusses the short period effect of the earth's oblateness, which is the major source of computational errors and consumption of machine time. Section 5 explains the data which is employed in the comparison and indicates the technique involved in obtaining initial conditions which will give a satisfactory fit over the entire arc. Section 6 gives the comparison between the data and the numerical integration. Section 7 presents the ephemeris. The final section, Section 8, presents the conclusions.

2. History of Satellite

The satellite was launched on May 29, 1965. The initial orbital elements, as assumed in this report, are

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\begin{align*}
a_0 &= 21.726138 \text{ earth radii} \\
e_0 &= 0.95251083 \\
i_0 &= 33\degree828446 \\
w_0 &= 135\degree \\
\Omega_0 &= 138\degree45267 \\
M_0 &= 0\degree023663
\end{align*}
\]

or, in vector form,

\[
\begin{align*}
x_0 &= 6099.5844 \text{ km} \\
y_0 &= 602.05128 \\
z_0 &= 2409.1608 \\
x_0 &= 1.1047527 \text{ km s}^{-1} \\
y_0 &= 9.8556127 \\
z_0 &= -4.4520836.
\end{align*}
\]

Epoch time is May 29, 1965 at 1207 h UT \((a=\text{semimajor axis}, e=\text{eccentricity}, i=\text{inclination}, w=\text{argument of perigee}, \Omega=\text{right ascension of the ascending node}, M=\text{mean anomaly}). The angular elements are taken with respect to the Earth's