Chapter 2
Earth-Based Tracking and Navigation Overview

2.1 Navigation Process

The process of spacecraft navigation is illustrated in Fig. 2-1. The two primary navigation functions are orbit determination and guidance. The orbit determination process is an iterative procedure requiring an a priori estimate of the spacecraft trajectory, referred to as the nominal orbit. Expected values of the tracking observables are calculated, based upon nominal values for the trajectory and precise models of the tracking observables. These calculated observables are differenced with the actual values obtained from the tracking system to form the data residuals.

If the trajectory and the data models were perfectly known, the residuals would exhibit a purely random, essentially Gaussian, distribution due to uncorrelated measurement errors (for example, thermal noise in the tracking receiver). However, errors in the trajectory and the observable models introduce distinctive signatures in the residuals. These signatures enable an adjustment to the model parameters through a procedure known as weighted linear least-squares estimation, in which the optimal solution is defined to be the set of parameter values that minimizes the weighted sum of squares of residuals. When the data are weighted by the inverse of their error covariance, the procedure yields a minimum variance estimator [1]. Since this procedure represents a linear solution to a nonlinear problem, the steps must be iterated, using the latest parameter estimates, until the solution converges.

The accuracy of the solutions obtained in the manner explained above may be assessed through a variety of tests. The calculated, or formal, uncertainties

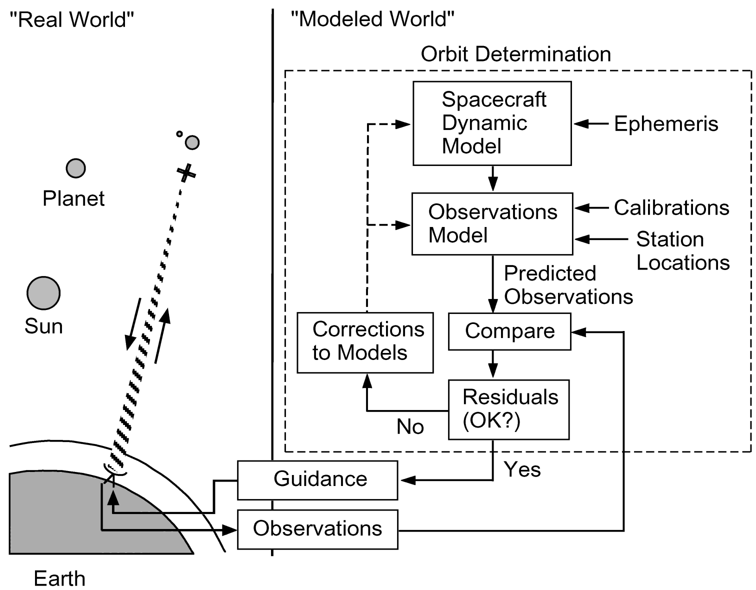

Fig. 2-1. The navigation process. Orbit determination is an iterative procedure for estimating the spacecraft trajectory and related physical parameters from a set of tracking data. Guidance involves the calculation of optimal maneuvers and commands needed to deliver the spacecraft to the desired target.

are obtained from the least-squares algorithm in the form of a parameter-error covariance matrix [1]. The postfit residuals (that is, the residuals calculated from the weighted least-squares solution) are examined for systematic trends and/or large scatter relative to the expected data noise. A more concrete test involves the subsequent acquisition of additional tracking data and an assessment of the behavior of the predicted, or unadjusted, residuals. Other tests involve comparing solutions obtained from different mixes of tracking data, model parameters, and so forth. Large variations in such solutions, relative to the calculated formal uncertainties, are strong indications of model errors, either in the tracking data or in the spacecraft dynamics.

Once the navigators are confident that the trajectory can be reliably predicted, guidance algorithms are executed to calculate any necessary retargeting maneuvers, and reoptimization of the trajectory may be performed, as necessary. The orbit-determination and guidance functions are repeated, as required, during interplanetary flight until the spacecraft is accurately delivered to the target body. Delivery accuracy requirements vary from mission to mission, but typically become increasingly more challenging as demonstrated navigation performance improves. For example, the one sigma (standard deviation) delivery requirement for the Voyager Io encounter was approximately

900 km [2]. The comparable value for the first Galileo Io encounter was about 100 km [3].

2.2 Reference Frames

As astronomical measurement accuracies have improved, understanding and definitions of reference frames have evolved from simple geometric concepts to abstract, implicitly defined constructs. What follows is an introduction to reference frames, using historical concepts. A more rigorous approach is described at the end of this section.

Discussion of Earth-based tracking capabilities is most readily accomplished using a geocentric equatorial reference system such as the one shown in Fig. 2-2. In this system, Earth is located at the center of a celestial sphere. Earth's equator is the plane of reference, and the celestial poles are defined by an extension of Earth's axis of rotation.

The plane that contains the celestial poles and an object, for example, a spacecraft, describes a great circle on the celestial sphere. The point at which

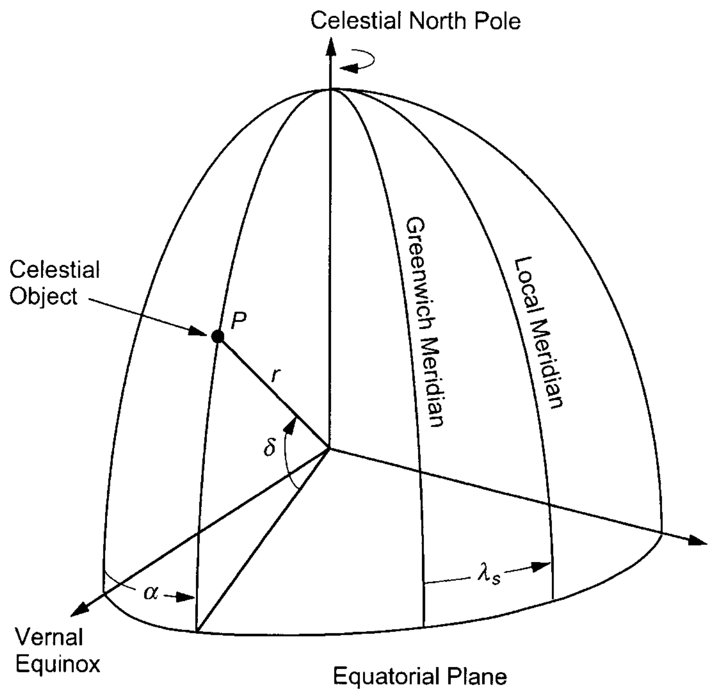

Fig. 2-2. Geocentric equatorial reference frame. The spherical coordinates of an object, *P*, are given by the geocentric range, *r*, and the angles α (right ascension) and δ (declination).

this circle intersects the celestial equator defines the right ascension of the object, α, measured easterly from the vernal equinox. The angular distance of the object from the equator measured on the great circle is termed the declination, δ. The declination varies from –90 deg to +90 deg with a positive angle indicating that the object is north of the equator.

The reference point for measurement of right ascension is defined by the intersection of Earth's equator and the ecliptic, the plane in which Earth moves about the Sun. The point where the Sun crosses the equator on its apparent path northward is termed the vernal equinox. This point, however, gradually moves with time, due to precession of Earth's axis about the pole of the ecliptic. Therefore, the vernal equinox must be defined as of a specific date. The current internationally accepted epoch is 12:00 on January 1 of the year 2000, or Julian date 2451545.0, and is referred to as J2000. This epoch has been adopted by the International Astronomical Union (IAU) and the International Earth Rotation Service (IERS) along with a set of standards for precession and nutation of Earth's pole and other physical models and constants associated with Earth-based observation systems [4–8].

Measurements from stations fixed on Earth are best described in an Earth-fixed (terrestial) reference frame (see Section 3.3.4). In this terrestial frame, points on Earth are located relative to the instantaneous Earth pole and equator, and a great circle that passes through Greenwich, known as the prime meridian (see Fig. 2-2). Spacecraft positions, on the other hand, are calculated in a space-fixed (celestial) reference frame associated with the planetary ephemeris. This celestial reference frame is typically a solar system barycentric frame aligned with the mean Earth equator and equinox of J2000 [9]. Transformations between the terrestial and celestial frames are carefully modeled and accounted for in the orbit-determination process [9]. Tracking-system calibrations that support these transformations are described in Sections 3.3.4 and 4.1.

Today, the celestial reference frame is defined by the positions of quasars in the International Celestial Reference Frame (ICRF) [10]. The origin of right ascension is a certain linear combination of catalog coordinates. The equator and the equinox are now measured quantities. Planetary ephemerides are constrained to be consistent with this definition to within current knowledge. For a more in-depth discussion of reference frames, see references 10–15 and the references therein.

2.3 Spacecraft Equations of Motion

Spacecraft trajectories are calculated by integrating the equations of motion in the celestial reference frame adopted by the project. This frame is implicitly defined by the current planetary ephemeris and is closely aligned with the ICRF [15,16].

In the trajectory computations, all known dynamical influences on the spacecraft are accounted for. These include all solar system gravitational accelerations and all nongravitational forces such as solar radiation pressure, attitude control thrusts, and gas leaks from the spacecraft control jets. Accurate representation of these forces can require detailed modeling of spacecraft features such as reflective surfaces, heat radiation characteristics, and thruster units. Initial conditions for the equations of motion are the six parameters representing spacecraft position and velocity at a specific epoch. This initial state may be expressed in a variety of forms within the adopted reference frame, for example, using Cartesian or spherical coordinates, or in terms of classical Keplerian elements. It is convenient in the ensuing discussion of Earth-based tracking to refer to the spherical coordinates (r, α, δ, $\dot{r}$, $\dot{\alpha}$, $\dot{\delta}$) in the geocentric Earth equatorial system of Fig. 2-2.

Given perfect knowledge of all forces and purely random measurement errors, a navigator must estimate only six parameters (r, α, δ, $\dot{r}$, $\dot{\alpha}$, $\dot{\delta}$) to determine a spacecraft orbit. The real world is not so kind, however. A more typical scenario requires simultaneous determination of spacecraft state and selected parameters of the force models. It is also usually necessary to estimate a number of model parameters associated with calculation of the tracking observables. These models are the focus of the next chapter.

References

[1] G. J. Bierman, *Factorization Methods for Discrete Sequential Estimation,* Mathematics in Science and Engineering, vol. 128, New York: Academic Press, 1977.

[2] J. P. McDanell et al., *Voyager Orbit Determination Strategy and Accuracy,* Voyager Project Document 618-118 (internal document), Jet Propulsion Laboratory, Pasadena, California, 1977.

[3] *Galileo Navigation Plan,* Galileo Project Document 625-566, Rev. A, JPL D-649, Rev. A (internal document), Jet Propulsion Laboratory, Pasadena, California, October 1989.

[4] D. D. McCarthy, editor, *IERS Standards (1992): IERS Technical Note 3,* Paris: Observatoire de Paris, 1992.

[5] D. D. McCarthy, editor, *IERS Standards (1996): IERS Technical Note 21,* Paris: Observatoire de Paris, 1996.

[6] P. K. Seidelmann, “1980 IAU Theory of Nutation: The Final Report of the IAU Working Group on Nutation,” *Celestial Mechanics*, vol. 27, no. 1, pp. 79–106, May 1982.

[7] J. H. Lieske et al., “Expressions for the Precession Quantities Based upon the IAU (1976) System of Astronomical Constants,” *Astronomy and Astrophysics*, vol. 58, no. 1–2, pp.1–16, June 1977.

[8] J. H. Lieske, “Precession Matrix Based upon IAU (1976) System of Astronomical Constants,” *Astronomy and Astrophysics*, vol. 73, no. 3, pp. 282–284, March 1979.

[9] T. D. Moyer, *Formulation for Observed and Computed Values of Deep Space Network Data Types*, JPL Publication 00-7, Jet Propulsion Laboratory, Pasadena, California, October 2000.

[10] C. Ma et al., “The International Celestial Reference Frame as Realized by Very Long Baseline Interferometry,” *The Astronomical Journal*, vol. 116, no. 1, pp. 516–546, July 1998.

[11] O. J. Sovers and C. S. Jacobs, *Observation Model and Parameter Partials for the JPL VLBI Parameter Estimation Software MODEST—1996*, JPL Publication 83-39, Rev. 6, Jet Propulsion Laboratory, Pasadena, California, August 1996.

[12] *1998 IERS Annual Report*, Paris: Observatoire de Paris, 1999.

[13] E. M. Standish, “Celestial Reference Frames: Definitions and Accuracies,” chapter in *The Impact of VLBI on Astrophysics and Geophysics* (M. J. Reid and J. M. Moran, editors), proceedings of the 129th IAU Symposium, Cambridge, Massachusetts, May 10–15, 1987, Dordrecht, The Netherlands: D. Reidel, 1988.

[14] J. G. Williams and E. M. Standish, “Dynamical Reference Frames in the Planetary and Earth-Moon Systems,” chapter in *Reference Frames in Astronomy and Geophysics* (J. Kovalevsky et al., editors), Dordrecht, The Netherlands: Kluwer, 1989.

[15] C. S. Jacobs et al., “The Extragalactic and Solar System Celestial Frames: Accuracy, Stability, and Interconnection,” *Advances in Space Research*, vol. 13, no. 11, pp. (11)161–(11)174, 1993.

[16] E. M. Standish, “JPL Planetary and Lunar Ephemerides, DE405/LE405,” JPL Interoffice Memorandum 312.F-98-048 (internal document), Jet Propulsion Laboratory, Pasadena, California, August 26, 1998.