The Consultative Committee for Space Data Systems

Report Concerning Space Data System Standards

NAVIGATION DATA— DEFINITIONS AND CONVENTIONS

INFORMATIONAL REPORT

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FOREWORD

This Report contains technical material to supplement the CCSDS Recommended Standards for the standardization of spacecraft navigation data generated by CCSDS Member Agencies. The topics covered herein include radiometric data content, spacecraft ephemeris, planetary ephemeris, tracking station locations, coordinate systems, and attitude data. This Report deals explicitly with the technical definitions and conventions associated with inter-Agency cross-support situations involving the transfer of ephemeris, tracking, and attitude data.

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1 INTRODUCTION

1.1 PURPOSE AND SCOPE

Spacecraft navigation data is exchanged between Consultative Committee for Space Data Systems (CCSDS) Member Agencies during cross support of space missions. The purpose of this document is to establish a common understanding for the exchange of spacecraft navigation data. For purposes of this document, orientation and maneuver information are included as part of the spacecraft navigation process.

1.2 APPLICABILITY

This document applies to navigation and attitude data exchanged in the following cases:

– flight-to-ground;

– ground-to-flight;

– ground-to-ground;

– flight-to-flight.

This document serves as a guideline for the development of compatible, inter-Agency standards for the exchange of spacecraft navigation and attitude data.

1.3 STRUCTURE OF THIS DOCUMENT

a) Section 2 provides a brief overview of spacecraft navigation.

b) Section 3 provides foundational information on the data types and units served by the navigation messages.

c) Section 4 provides details about coordinate frames, time systems, astrodynamics constants, environmental models, and other ancillary concepts important in spacecraft navigation.

d) Section 5 discusses properties and processes of the entities that participate in a navigation data exchange.

e) Section 6 discusses the types and associated attributes of measurements that may be made during a navigation session.

f) Annexes A and B constitute a Glossary of Terms and a listing of Acronyms, respectively.

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1.4 REFERENCES

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2 SCOPE OF NAVIGATION

2.1 GENERAL

This section briefly describes the spacecraft navigation process, and defines terms relevant to this process.

2.2 NAVIGATION

2.2.1 DEFINITION

The word ‘navigate’ is derived from the Latin words *navis*, meaning ship, and *agere*, meaning to move or direct. The common definition of navigation establishes that it is the science of getting a craft or person from one place to another. In this document, ‘navigation’ is the process used to find the present and future orbit and orientation of a spacecraft using a series of measurements. For purposes of this document, orientation and maneuver information are included as part of the spacecraft navigation process.

2.2.2 SPACECRAFT NAVIGATION PROCESS

In its simplest form, navigation is the determination of the position and/or orientation of an object. The position problem is generally called orbit determination and the orientation problem is called attitude determination. Orbit and attitude determination, although related, affect each other only weakly, hence they can generally be performed separately. For example, a nominal attitude can generally be used in drag models that affect orbit determination, and a predetermined orbital ephemeris can generally be used in attitude determination.

2.3 DEFINITIONS OF SPACECRAFT NAVIGATION TERMS

In order to establish a solid standard for the exchange of spacecraft navigation data among agencies, it is important to clearly define terms relevant to this process. These terms are as follows:

Navigation is the process used to find the present and future orbit and orientation of a spacecraft using a series of measurements.

Guidance is the process of defining a path to move a spacecraft from one point to another or from one orientation to another.

Control is the process to maintain a spacecraft within the prescribed orbital and attitude path.

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Orbit is the translational motion of a spacecraft resulting from the gravitational forces of larger mass bodies acting on the spacecraft. The orbit can be represented in state vector format as position and velocity or as orbital elements.

Attitude is the orientation and/or pointing of a spacecraft and can be represented as a transformation from a reference coordinate frame to a body fixed coordinate frame, or it can be represented by a spin axis direction in a reference frame and spin rate about the spin axis.

Definitive is when the state estimate spans the period for which there are measurements.

Predictive is when the state estimate spans a time for which there are no available measurements, often used for planning purposes.

Absolute State is ~~a state solution~~ defined with respect to a fixed reference frame.

Relative State is ~~a state solution~~ defined with respect to another spacecraft or moving element.

The responsibilities for guidance and control are outside of the scope of this document.

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3 NAVIGATION MESSAGE DATA TYPES AND UNITS

3.1 GENERAL

This section describes the data types involved in the exchange of navigation data messages.

3.2 NAVIGATION EXCHANGE DATA TYPES

Tables 3-1 through 3-3 contain lists of the most typical measurement, property, and ancillary information data types currently exchanged. For current and future Recommended Standards, it is preferable to use the units from these tables; in most cases, the International System of units (SI) will be used. However, there are cases where it is not possible to convert raw measurements from hardware-specific units to SI units without risking some degradation of measurement quality. For example, for Doppler and range measurements collected during periods when the transmitter frequency is time varying, a conversion to SI units is only possible with accurate trajectory information. In this case, the recommendation is for the participating agencies to agree upon a hardware-specific unit (e.g., 'range units', a function of the uplink frequency).

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Table 3-1: Typical Measurement Data Types

Type	Typical Units
Range	km, s, RU
Variable Transmitter Range	RU
Range rate	km/s
Light Time	s
Angles (antenna tracking, sun sensor, star sensor, gyro package, horizon sensor, videometers, etc.)	deg, rad
Doppler (coherent)	Hz or counts
Doppler (non-coherent)	Hz or counts
Variable Transmitter Doppler	Hz
Integrated Doppler count	counts
Rate sensors (IRU, gyros)	deg/s
Accelerometer output	deg/s^2 or m/s^2
Magnetometer output	μT

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Table 3-2: Typical Property Data Types

Type	Units	Alternate Units
Position	km	
Angular Velocity	deg/s	rad/s
Linear Velocity	km/s	
Angular acceleration	deg/s^2	rad/s^2
Acceleration	km/s^2	
Length	m	
Moment of Inertia	kgm^2	
Force	kgm/s^2	N
Torque	kgm^2/s^2	Nm, J/rad
Mass	kg	
Linear momentum	kgm/s	Ns
Angular momentum	kgm^2/s	Nms
Energy, work, or heat	kgm^2/s^2	Nm, J
Power	kgm^2/s^3	W
Pressure	kg/ms^2	hPa
Temperature	K	
Transmitter delay	s	
Receiver delay	s	
Surface	m^2	
Antenna angles	deg	rad
Oscillator frequency	MHz, GHz	Hz
Ballistic coefficient	m^2/kg	
Aerodynamic coefficient	1	
Reflectivity	1	
Angles	deg	rad
Angular drift	deg/s^2	rad/s^2
Magnetic field components	μT	

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Table 3-3: Typical Ancillary Information Data Types

Type	Units
Physical constants	Depends on constant
Transmitter ID	N/A
Receiver ID	N/A
Epoch	N/A
Co-ordinate system description	N/A
Time system description	N/A
Antenna Type	N/A
Quality of property	N/A

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4 ANCILLARY DATA

4.1 GENERAL

This section describes ancillary data types, which are pieces of information needed to interpret measurements and properties of navigation participants. In general, ancillary information makes it possible to take properties or measurements and incorporate them correctly into numerical computations. In some cases, very detailed modeling information is passed along so that measurements and properties can be used in state-of-the-art, high-fidelity computations.

4.2 PHYSICAL CONSTANTS AND DATA QUALITY

Navigation uses many physical constants having uncertainty values that are agreed upon in the international community; the user should refer to the relevant governing document for the information. For some exchanges, it may be desirable to include the uncertainty of the transmitted data. This uncertainty information is referred to as data quality. Quality specifications are included in the individual Recommended Standards, as applicable.

4.3 COORDINATE FRAME IDENTIFICATION

4.3.1 GENERAL

This subsection defines coordinate system terms and describes commonly used specifications. The SANA registry, https://sanaregistry.org/r/reference_frames, contains a complete definition of coordinate reference frames and https://sanaregistry.org/r/orbit_centers contains a complete definition of object-centric orbit reference frames.

4.3.2 COORDINATE SYSTEM DEFINITIONS

A **coordinate frame** is defined as an associated set of mutually orthogonal Cartesian axes. For the following definitions, the axes are referred to as x, y, and z).

The **frame origin** is the common origin of the Cartesian axes, also called ‘center name’.

The **reference plane** is the xy plane in a coordinate frame.

The **reference direction** is the direction of the x axis.

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4.3.3 COORDINATE SYSTEM SPECIFICATIONS

4.3.3.1 Specifying (1) the frame origin of the Cartesian axes, (2) the reference direction, and (3) the reference plane (or its normal direction, the z-axis direction) is sufficient to define a coordinate frame unambiguously. Frame origins are generally either (1) a point defined on a participant (examples are the center of mass of a spacecraft, the location of a ground station, etc.), (2) the center of mass of a natural body, (3) the center of mass of a set of bodies (referred to as the barycenter), or (4) the object position on orbit at each given time for local frames. Other frame origins are possible for special purposes; for some applications, the frame origin is unnecessary. For the purposes of this discussion, the origin will remain constant over time. The reference direction and reference plane normal are vectors that can be defined in one of these ways:

– point to a fixed direction in inertial space (e.g., toward a quasar);

– be parallel to the distance vector between one object and another;

– be parallel to an object's velocity vector;

– point from the origin through the intersection of two defined planes;

– be parallel to an object's spin axis; or

– be normal to an object's orbit.

4.3.3.2 In many cases, the reference plane is the equator or orbit plane of a natural body or orbit of the spacecraft. In those cases, the motions of the equator or orbit plane are explicitly computed. A natural body's equatorial bulges can be perturbed by the gravitational attraction of other natural bodies; this causes variations in the orientation of the equatorial plane. Also, the perturbative effects of natural bodies on each other cause variations in the orientation of their orbit plane. Long term motions that can be treated as though they are secular are known as **precession** motion; short periodic motions are referred to as **nutation**. When the natural body's equator and ecliptic are defined as being represented by the precession motions only, these are referred to as **mean** directions. Those affected by both precession and nutation are referred to as **true** directions. Directions fixed at the time corresponding to a fundamental reference are referred to as values at the **epoch**, while those referring to instantaneous moments are referred to as **values of date**.

4.3.3.3 With the information contained in 4.3.3.2, it is possible to specify any form of coordinate reference system about any form of participant. In current practice, however, the vast majority of navigation messages between agencies use only a small subset of the possibilities, some of which are described in the following subsections.

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4.3.4 INTERNATIONAL CELESTIAL REFERENCE SYSTEM (ICRS)

4.3.4.1 General

In 1991 the International Astronomical Union (IAU) established the International Celestial Reference System (ICRS) as the fundamental inertial coordinate system (reference [1]). The origin of the ICRS is defined as the solar system barycenter within a relativistic framework, and its axes are fixed with respect to distant extragalactic radio objects. The ICRS is a kinematical concept. The IAU has also defined dynamical reference systems that are based on general relativity; the Barycentric Celestial Reference System (BCRS) defined with the origin at the solar system barycenter, which can be considered to provide a way to prolong the coordinate lines of the ICRS from distant quasars to any point in the solar system, and the Geocentric Celestial Reference System (GCRS) which is a version of the local geocentric reference system for the Earth. Due to the Earth's motion around the solar system barycenter, there are small relativistic precession and nutation terms in the GCRS.

4.3.4.2 International Celestial Reference Frame (ICRF)

The practical realization of the ICRS is designated the International Celestial Reference Frame (ICRF), which is jointly maintained by the International Earth Rotation Service (IERS) and the IAU Working Group on Reference Frames. The fundamental plane of the ICRF is closely aligned with the mean Earth equator at J2000, and the origin of right ascension is defined by an adopted right ascension of the quasar 3C273 to closely match the vernal equinox at J2000 (reference [9]). The difference between the dynamical J2000 reference frame and the ICRF is at a level of 0.01 arc second, and determined with an accuracy of 0.003 arc second (reference [5]). The Hipparcos star catalogue is an optical realization of the ICRS (reference [11]).

4.3.4.3 International Terrestrial Reference System (ITRS)

Complementary to the ICRS, the International Terrestrial Reference System (ITRS) provides the conceptual definition of an Earth-fixed reference system (reference [1]). Its origin is located at the Earth's center of mass (including oceans and atmosphere), and its unit of length is the SI meter. The orientation of the IERS Reference Pole (IRP) and IERS Meridian (IRM) are consistent with the previously adopted Bureau International de l'Heure (BIH) system at epoch 1984.0, as well as the former Conventional International Origin (CIO). The time evolution of the ITRS is such that it exhibits no net rotation with respect to the Earth's crust. The International Terrestrial Reference Frame (ITRF) is a realization of the ITRS. New versions of the ITRF are published occasionally and exhibit global differences on the centimeter level (reference [3]).

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4.3.5 TRUE OF DATE AND MEAN OF DATE COORDINATE SYSTEMS

The True of Date (TOD) coordinate system is frequently used for astrodynamical applications. The mean equator and equinox of a given date define a Mean-of-Date (MOD) coordinate system, which includes the effects of precession but not the effects of nutation. A given date's true equator and equinox, which can be obtained by applying nutation to the mean values, define a True Of Date (TOD) coordinate system.

4.3.5.1 Greenwich True of Date (GTOD) Coordinate System

The Greenwich True of Date (GTOD) (geographic) coordinate system is a rotating, right-handed, Cartesian system with the origin at the center of the Earth. The orientation of this system is specified with:

– The xy plane is the Earth's true of date Equator.

– The z axis is directed along the Earth's true of date rotational axis and is positive north.

– The positive x axis is directed toward the prime meridian.

– The y axis completes a right-handed system.

Greenwich True of Date is also referred to as 'True of Date Rotating (TDR)' or 'Greenwich Rotating Coordinate Frame'.

4.3.5.2 Relationships among Common Reference Frames

The transformations for position vectors (**r**) among the ICRS, the ITRF, and the TOD coordinate systems are performed in figure 4-1 **Error! Reference source not found.**. With the simplification

$$\dot{\Pi}(\mathrm{t}) \approx \dot{\mathrm{N}}(t) \approx \dot{\mathrm{P}}(t) \approx 0,$$

which is applicable in the framework of navigation data exchange, the transformations of velocity vectors (**v**) among these frames are also shown.

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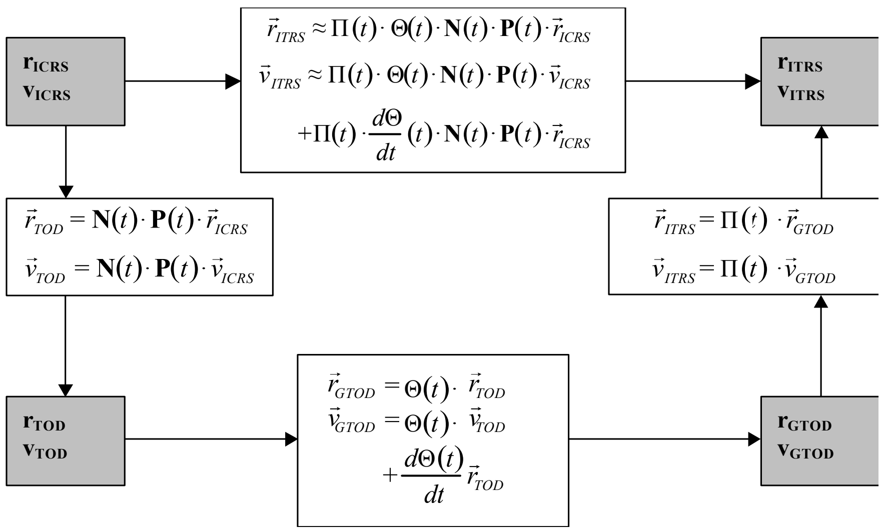

where:

- **Π**: Polar Motion Transformation Matrix
- **Θ**: Earth Rotation Transformation Matrix
- **N**: Nutation Transformation Matrix
- **P**: Precession Transformation Matrix

Figure 4-1: Relationships among Common Reference Frames

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where ω_E is the time dependent Earth angular velocity.

Details for these transformations are contained in reference [1].

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4.3.5.3 TRUE EQUATOR MEAN EQUINOX (TEME)

The True Equator Mean Equinox (TEME) reference frame, used in the Simplified General Perturbations Satellite Orbit Model 4 (SGP4), has both 'of date' and 'of epoch' variants. In TEME, the x-axis points along the mean vernal equinox and the z-axis points along the true rotation axis of the Earth at the specified date or coordinate epoch. The TEME reference frame was developed during early satellite operations because it was computationally convenient. For various reasons, there is no consensus or authoritative definition for the reference frame. Consequently, the use of TEME is not recommended, though it still forms the basis for NORAD Two Line Elements (TLEs), which are widely used in Earth orbit operations.

4.3.6 BODY FRAME SPECIFICATIONS

Just as there are various coordinate frames associated with an orbit for convenience, a coordinate frame is established to define the physical geometry of the spacecraft body. There is no restriction on this frame (except for orthogonality) as its purpose is to make convenient the definition of mechanical components, science instruments, forces, and torques. Some examples of these coordinate frames are as follows:

a) center of frame placed at the center of mass or center of gravity and aligned along the principal axes of inertia;

b) center of frame placed at an arbitrary location on or near the spacecraft body and oriented to conveniently describe the location of mechanical equipment and/or science instruments;

c) center of frame placed at center of mass or center of gravity and oriented to define roll, pitch and yaw similar to the definition for aircraft.

If a reference frame is not centered on the center of gravity (usually coincident with the center of mass) then care must be taken when specifying forces and torques to be applied.

Local frames, such as individual instrument frames, may be defined in relation to a body frame.

4.3.7 LOCAL ORBITAL FRAME

4.3.7.1 General

Important reference frames defined using the orbital position and velocity at a given time are used for attitude estimation, attitude control, and orbital relative motion. All of the local orbital frames described in this subsection are rotating coordinate systems, unless specified otherwise in the context of some specific data exchange between participants. These systems can be used to study the relative motion between spacecraft using, for example, Hill's equations.

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4.3.7.2 Local Orbital Frame (LVLH)

The Local Vertical Local Horizontal (LVLH) frame is portrayed in Figure 4-2.

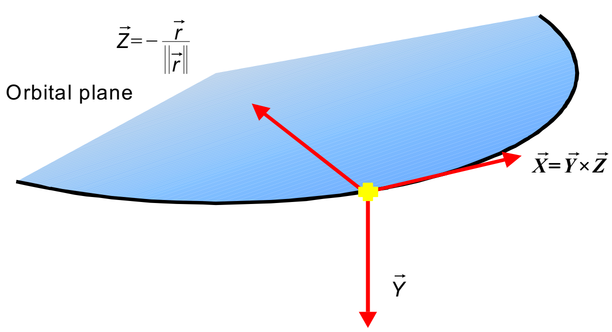

Figure 4-2: Local Orbital LVLH Frame

Frame origin: spacecraft gravity center

$\vec{Z}$: Unit vector collinear and opposite sign of gravimetric satellite position (planet center, spacecraft gravity center)

$\vec{Y}$: Unit vector collinear and opposite sign of the orbital kinetic momentum (normal to orbit plane)

$\vec{X}$: Unit vector equal to $\vec{Y} \times \vec{Z}$

4.3.7.3 Local Orbital Frame (TNW)

In 'TNW', the T stands for tangential, N for normal, and W for the Greek letter omega (ω) denoting the axis of angular momentum. The frame is portrayed in Figure 4-3.

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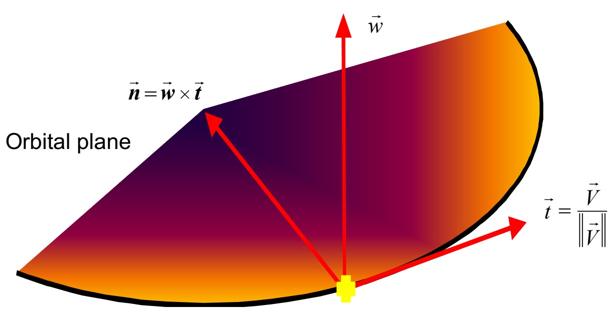

Figure 4-3: Local Orbital TNW Frame

Frame origin: spacecraft gravity center

$\vec{t}$: Unit vector collinear to absolute orbital velocity

$\vec{w}$: Unit vector collinear to orbital kinetic momentum (normal to orbit plane)

$\vec{n}$: Unit vector equal to $\vec{w} \times \vec{t}$

4.3.7.4 Local Orbital Frame (QSW), (RTN), (RIC), or (RSW)

In 'QSW, Q stands for normalized radius, S stands for second, and W for the Greek omega (ω). In 'RTN', R stands for Radial, T for Transverse, and N for Normal. The acronyms RSW and RIC (for radial/intrack/crosstrack) are also used. Figure 4-4 portrays the QSW/RTN/RIC/RSW frame.

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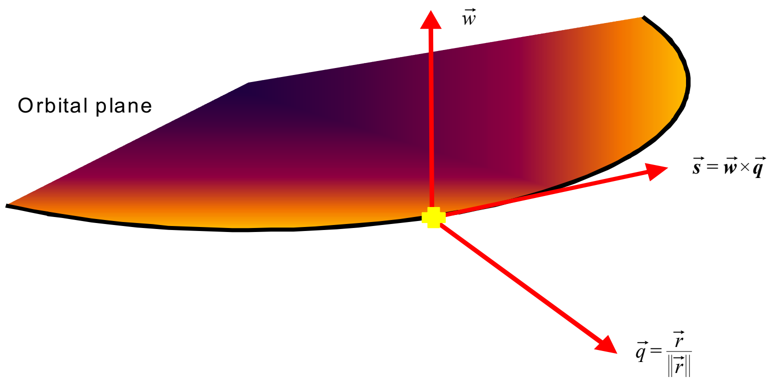

Figure 4-4: Local Orbital QSW Frame

Frame origin: spacecraft gravity center

Q or R: $\vec{q}$, $\vec{r}$ unit vector collinear to geocentric satellite position (planet center, spacecraft gravity center)

W or N or C: $\vec{w}$, unit vector collinear to the orbital kinetic momentum (normal to orbit plane)

S or T or I: $\vec{s}$, unit vector equal to $\vec{w} \times \vec{q}$

4.4 TIME

4.4.1 RATIONALE

The exact definition and understanding of time systems is essential for:

– the modeling of satellite orbits and attitude;

– processing of navigation data; and

– satellite ground operations.

This subsection provides a subset of definitions of time scales relevant to navigation messages (reference [1]). The relative differences between time scales appear as (1) step functions (for example, when leap seconds are added); (2) monotonically increasing differences (when relativistic effects are added); (3) periodic differences (due to solar system dynamics), or (4) constant.

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4.4.2 TIME SCALES

4.4.2.1 Differences Between Time Scales

Figure 4-5 provides an overview of the differences between the most relevant time scales described in references [1] and [5].

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NOTE – Periodic terms in Barycentric Coordinate Time (TCB) and Barycentric Dynamical Time (TDB) have been exaggerated by a factor of 100 to make them discernible.

The SANA registry, https://sanaregistry.org/r/time_systems, contains a list of time scales.

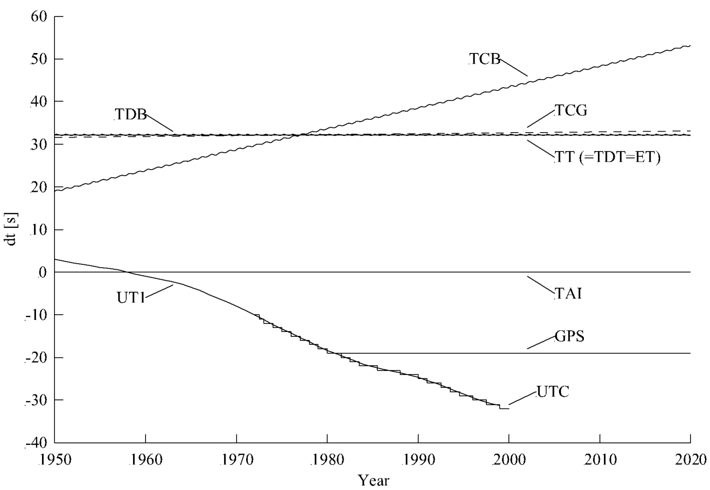

Figure 4-5: Differences between Relevant Time Scales between 1950 and 2020

4.4.2.2 Time Scales for Earth Orbiting Satellites

4.4.2.2.1 Terrestrial Time

Terrestrial Time (TT), previously known as Terrestrial Dynamical Time (TDT), is a conceptually uniform time scale that would be measured by an ideal clock on the surface of the geoid. TT is measured in days of 86400 SI seconds. Between TT and TCG the following relation holds (references [7] and [29]):

$$TT = (1 - L_G)\ TCG$$

where $L_G = 6.9692901 \times 10^{-10}$.

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4.4.2.2.2 Geocentric Coordinate Time

Geocentric Coordinate Time (TCG) is the Time scale of the GCRS. It is defined in the context of general relativity. It differs from Terrestrial Time by removing the effects of the Earth's motion. (references [1], [7], [26], and [29])

4.4.2.2.3 International Atomic Time

International Atomic Time (TAI) provides the practical realization of a uniform time scale based on atomic clocks and agrees with TT, except for a constant offset of 32.184s and the imperfections of existing clocks. (references [7] and [29])

Between TAI and TT, the following relation holds:

TAI = TT - 32.184s

4.4.2.2.4 Global Positioning System Time

Global Positioning System (GPS) time is an atomic time scale like TAI, but differs in the chosen offset and the choice of atomic clocks used in its realization. (reference [35])

The origin of GPS was chosen as:

GPS = UTC on 1980 January 6.0; i.e., GPS time differs from TAI by a constant offset of:

GPS = TAI - 19s

Other Global Navigation Satellite Systems have similar atomic time scales; however, each has its own relation to TAI.

4.4.2.2.5 Greenwich Mean Sidereal Time

Greenwich Mean Sidereal Time (GMST) is defined as the Greenwich hour angle of the mean vernal equinox of date (references [1] and [7]).

4.4.2.2.6 Universal Time

Universal Time (UT1) is today's realization of a mean solar time, which is determined by VLBI of selected radio point sources and interpolated by tracking of GPS satellites (references [1], [7], [25], [26], and [29]).

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4.4.2.2.7 Coordinated Universal Time

Coordinated Universal Time (UTC) is an atomic time scale based on the performance of atomic clocks. It is tied to TAI by an offset of integer seconds (called 'leap seconds'), which is regularly updated to keep UTC in close agreement with UT1 (within 0.9s). Since atomic clocks are more stable than the rate at which the Earth rotates, leap seconds are needed to keep the two time scales in agreement. Although it is possible to have a negative leap second (one second removed from UTC), so far, all leap seconds have been positive (one second added to UTC). Based on what is known about the Earth's rotation, it is unlikely that a second will ever be subtracted. The International Earth Rotation Service (IERS) notifies the world when a leap second is to be added or subtracted, which is done only at the end of June or December (references [1], [7], [24], [25], [26], and [29]).

When no leap second is to occur, the clocks count time over the transition from one day to the next as follows:

23:59:58, 23:59:59, 00:00:00, 00:00:01 UTC; no leap second.

To add a leap second, the clocks are made to count time at the transition from June 30 to July 1 or the transition from December 31 to January 1 as follows:

23:59:58, 23:59:59, 23:59:60, 00:00:00 UTC; positive leap second.

If a leap second were to be subtracted, the clocks would count time as follows:

23:59:58, 00:00:00, 00:00:01 UTC; negative leap second (23:59:59 missing).

The current difference between UTC and TAI, as well as a history of this difference due to leap second maintenance, can be obtained at the IERS website (reference

[2325]).

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[24]

4.4.2.3 Time Scales for Interplanetary Missions

4.4.2.3.1 Barycentric Dynamical Time (TDB)

Barycentric Dynamical Time (TDB) is the independent variable of current barycentric solar system ephemerides. This time was introduced by the IAU in 1976, revised in 2006, and defined to deviate from the TDT (which is now identical with TT) by periodic terms (≈2 ms) only (references [2], [7], and [10]).

4.4.2.3.2 Barycentric Coordinate Time

Barycentric Coordinate Time (TCB) is the relativistic time coordinate of the 4-dimensional barycentric frame (BCRS). TCB and TDB exhibit a scale difference of

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$$L_B = 1.550519768 \cdot 10^{-8} \text{ [s(TCB)/s(TDB)]}$$

This results in a secularly increasing difference of

$$\text{TCB - TDB} \approx 0.489\text{seconds/year} * (\text{year-1977.0})$$ (references [1] and [7])

4.4.2.4 Relationships between Time Scales

Figure 4-6 depicts the relationships between the time scales as given in reference [5].

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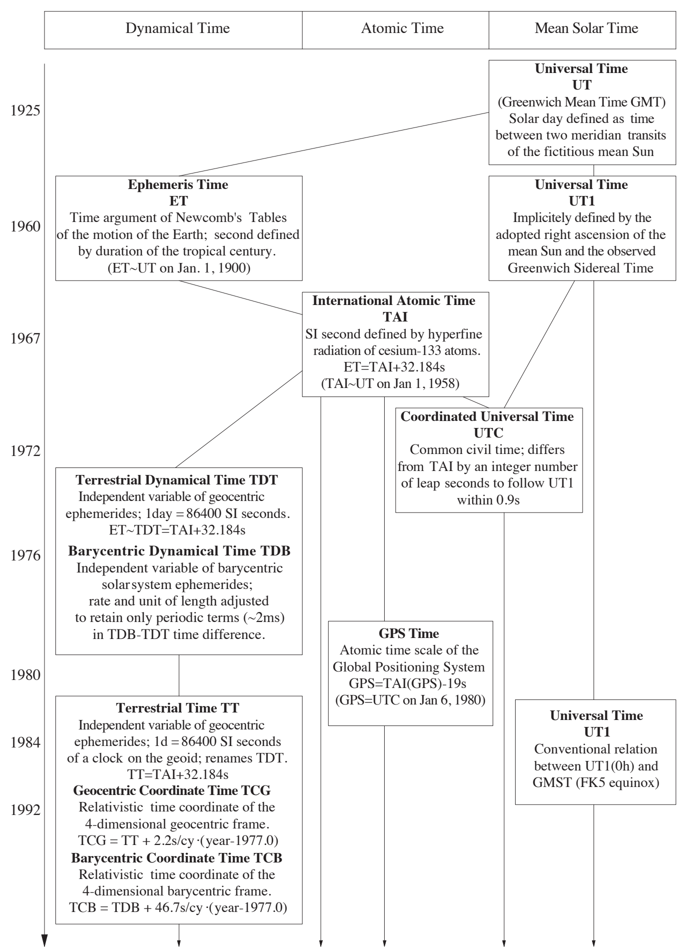

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Figure 4-6: Relationships among Time Scales

4.5 ASTRODYNAMIC CONSTANTS

Examples of astrodynamic constants exchanged for navigation purposes include the vacuum speed of light, gravitational parameter of a celestial body (GM or µ), and reference dimensions for Earth and other solar system bodies. Unless otherwise specified, the international community uses values recommended by the IERS (www.iers.org) or the IAU (www.iau.org) as appropriate (reference [31]).

4.6 ENVIRONMENTAL MODELS

Along with astrodynamic constants, mathematical models have been derived to compute effects on measurements, as well as effects on the equations of motion. Unless otherwise specified, the international community uses models recommended by the IERS (www.iers.org) for the Earth environment. For other bodies there is no general source.

4.7 ANTENNA TYPES

4.7.1 GENERAL

Several different antenna types are used in the process of collecting the tracking data that is used in the navigation process. For understanding the differences in these antenna types, some background is helpful. (references [21], [22], and [23]).

The locations of objects in the sky are described in terms of the 'celestial sphere', a virtual sphere of infinite radius that surrounds the Earth. The center, pole and equatorial plane (reference plane) of the celestial sphere are the same as those of the Earth.

Declination (DEC) represents the angle formed between the equatorial plane and a vector pointing to the object from the center of the Earth, usually expressed in degrees from -90 to 90. Positive declination angles represent objects north of the reference plane of the celestial sphere, and negative declination angles represent objects south of the reference plane.

Right ascension (RA) is conceptually equivalent to longitude. It measures how far the object is away from the zero point of the celestial reference direction (i.e., the vernal equinox). The right ascension may be expressed either in degrees from 0 to 360; or in hours, minutes, and seconds, where an hour of RA is 15 degrees of sky rotation. Together, the RA and DEC uniquely specify the inertial position of an object on the celestial sphere.

An object's hour angle (HA) is the time dependent distance in hours, minutes, and seconds westward along the celestial equator from the observer's meridian to the object's RA. The HA is zero when the object is on the observer's meridian.

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A keyhole is an area in the sky where an antenna cannot track a spacecraft because the required angular rates would be too high or portions of the sky are unviewable as a consequence of antenna architecture. Mechanical limitations may also contribute to keyhole size.

4.7.2 EQUATORIAL OR RADEC MOUNT

An equatorial mount or Right Ascension and Declination (RADEC) antenna mount is designed with one mechanical axis parallel to the Earth's polar axis. To track an object, the antenna is pointed toward the object's known RA and DEC and then rotated about the antenna's polar axis as the Earth rotates during the rest of the tracking pass.

4.7.3 THE X-Y MOUNT

In an X-Y antenna, X-angular movement is about a ground-fixed horizontal axle and Y-angular movement is about a perpendicular axle that rotates with X motion. The Y axle varies from a vertical to horizontal to inverted vertical orientation as X rotates through its range of motion. This configuration cannot directly swivel in azimuth as can an antenna having an AZEL mount. The X-Y mount can rotate freely in any direction from its upward-looking zenith central position.

An X-Y-mounted antenna is mechanically similar to a RADEC antenna. X-Y mounted antennas can have one of two types of configurations. These are XSYE and XEYN, the characteristics for which are shown in table 4-1:

Table 4-1: Characteristics for X-Y Antenna Mounts

X-Y Configuration	Keyholes	X-Axis Point	Y-Axis Point	Best For
XSYE	east/west horizon	south	east	objects in polar orbits
XEYN	north/south horizon	east	north	objects in lower inclination orbits

4.7.4 AZIMUTH-ELEVATION (AZEL) CONFIGURATION

An azimuth-elevation (AZEL) design antenna locates a point in the sky by azimuth (AZ) eastward (clockwise) from true north, and elevation (EL) above the horizon.

The AZEL mount has two perpendicular axes. The azimuth movement is about a ground-fixed vertical axle and elevation movement is about a perpendicular horizontal axle that rotates with azimuth motion. The keyhole in an AZEL system is near the zenith position.

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4.8 RELAY SATELLITES

4.8.1 RELAY SATELLITE USES

A relay satellite is an orbiting spacecraft used to relay communications and navigation signals. An example is the Tracking and Data Relay Satellite System (TDRSS) (reference [34]). Relay satellites are used to provide communication and navigation coverage to surface and orbiting missions, or to bridge a link gap where a direct-to-earth link is unavailable, to assist low power systems to telemeter data across large distances, or to ensure low latency communications through broad coverage.

4.8.2 CONSIDERATIONS

When the signal from which the navigational measurements are derived flows through a relay satellite, there are additional considerations for the orbit estimation of the spacecraft of interest, the User. The relay ephemeris knowledge contributes uncertainty to both the distance measurement (time of flight, range) and the frequency change measurement (Doppler, carrier phase). The relay subsystems contribute time delays as the signal passes through the relay on the forward and return paths affecting the range measurement. These delays are measurable pre-flight and can be applied to the range measurement. Components onboard the relay contribute phase noise that reduces the coherency or alignment to the frequency reference system for the Doppler or carrier phase measurement. Various techniques exist to solve for the relay orbital state either independently of or simultaneously with the User navigation. Regardless, the iterative procedures used for estimating the state of a single satellite still apply to a multi-satellite solution case.

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5 PROPERTIES

5.1 GENERAL

There are many possible ways to group the physical attributes of spacecraft, rovers, equipment, and tracking stations that are needed for navigation. This section first discusses the simplest physical attributes, and then introduces progressively more sophisticated attributes. Examples are provided in Table 5-1 for spacecraft properties, but similar properties apply to rovers, equipment, and tracking stations.

Table 5-1: Example Spacecraft Property Data Types

Property Data Type Grouping	Definition	Example Data Types (other Types possible)
Point Source	The attributes that can be associated with an object when it is treated as a point source.	Position Velocity Acceleration
Three-Dimensional Object	The attributes that can be associated with an object when it is treated as a three-dimensional object.	Orientation Angles Angular Velocity Quaternion
Physical	The attributes that are physical characteristics of the spacecraft in its entirety.	Spacecraft Mass Moments of Inertia Solar Radiation Pressure Area Solar Radiation Pressure Coefficient Aerodynamic Drag Area Aerodynamic Coefficient Mass Flow Rate
Hardware	The attributes that are physical characteristics of a specific sub-assembly on a spacecraft.	Solar Panel Area, Bus and Payload Area Transmitter Delays Receiver Delays Oscillator Frequency Oscillator Stability Earth sensor Gyro Star sensor Sun sensor Accelerometer Magnetometer Thrusters

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5.2 POINT SOURCE PROPERTIES

5.2.1 SURFACE STATION LOCATIONS

Spacecraft tracking and communications are made possible by a network of fixed ground stations located around the world. Whenever an orbiting spacecraft passes across the field of coverage of a ground station, the ground station can collect tracking data that allows determination of the spacecraft position and velocity. In this category is also considered the location of surface rovers and remote stations.

For surface tracking stations, the station coordinates and uncertainties are commonly defined based on the ITRF (reference [3]), using Cartesian coordinates (reference [4]). The reference point described by the coordinates is usually independent of its pointing direction. Pointing-dependent corrections, if significantly larger than the location uncertainty, are specified separately.

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5.2.2 STATE VECTORS AND ORBITAL ELEMENTS

5.2.2.1 General

The motion of a satellite around a central body may be described by various sets of parameters called 'orbital elements'. In 5.2.2.3 and 5.2.2.4, short definitions are given for the most commonly used representations.

5.2.2.2 Relay Satellites

The position and motion of relay satellites are represented in the same way as other satellites, with residual ephemeris, carrier phase, and delay errors manifested in the tracking data.

5.2.2.3 State Vector

The time-dependent spacecraft position (km) and velocity vectors (km/s)

$$\vec{r}(t) = \begin{pmatrix} x(t) \\ y(t) \\ z(t) \end{pmatrix} \quad \text{and} \quad \dot{\vec{r}}(t) = \begin{pmatrix} \dot{x}(t) \\ \dot{y}(t) \\ \dot{z}(t) \end{pmatrix}$$

are usually given in the 6-dimensional representation of the state vector

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$$\vec{Y}(t) = \begin{pmatrix} x(t) \\ y(t) \\ z(t) \\ \dot{x}(t) \\ \dot{y}(t) \\ \dot{z}(t) \end{pmatrix}$$

in a specified coordinate system at a specific epoch.

5.2.2.4 Classical Keplerian Elements

For some purposes it may be convenient to use the classic osculating Keplerian elements (Table 5-2), which are an equivalent representation of the Cartesian state vector at a specified epoch.

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Table 5-2: Classical Keplerian Elements

Parameter	Symbol	Unit
Semi-major Axis	a	km
Eccentricity	e	–
Inclination	i	deg
Right Ascension of Ascending Node	Ω	deg
Argument of Pericenter	ω	deg
True Anomaly	ν	deg

NOTE – The application of the Keplerian elements does not make sense for special cases (e.g. e = 1).

For some applications (e.g., orbit maintenance of remote sensing satellites), it is common to use mean Keplerian elements. Figure 5-1 portrays the angles associated with Keplerian elements (reference [19]).

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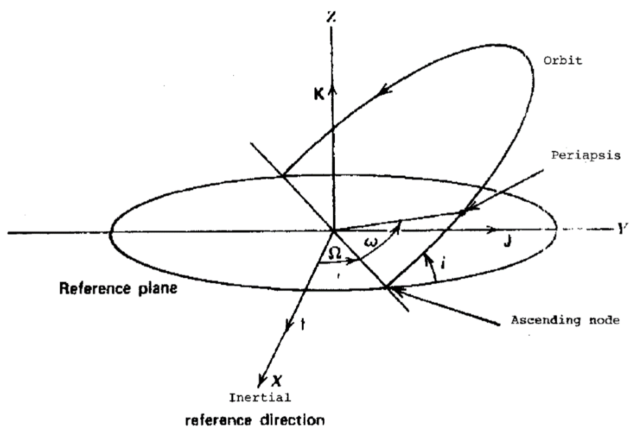

Figure 5-1: Classical Keplerian Orbit Orientation Angles

5.2.2.5 Mean Orbit Elements

The derivation of analytical or semi-analytical models exhibits two different kinds of cumulative effects impacting the motion of the spacecraft:

– the secular (i.e., varying linearly with time) and long-term effects (with periods typically more than one orbit period);

– the short-term effects (with periods less than one orbit period).

The trajectory obtained when only considering the secular and long-term effects is the mean trajectory. This fictitious trajectory corresponds to the real trajectory from which all the short-term (zero-mean) perturbation effects have been removed. The orbit elements that describe this mean trajectory are referred to as the mean orbit elements. The orbit elements are not uniquely defined and make sense for a specific model only. That is why it is important to know which model should be used to propagate a given set of mean orbit elements. The inputs to analytical (or semi-analytical) models are generally mean orbit elements (in addition to other modelling data), as is the case for SGP4 (reference [32]), which uses mean values for the inclination, mean motion, eccentricity, etc.

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5.2.3 ORBIT PREDICTION AND PROPAGATION

5.2.3.1 General

Orbit prediction is the process of forecasting the position and velocity of a spacecraft by using dynamical models to extrapolate from the orbit history to a future time. Propagation is the process of using the dynamic equations of motion and mathematical models describing the effects of the forces on the spacecraft to obtain the position and velocity at a future time and is generally the method of choice when orbit states are needed at an extended time in the future.

5.2.3.2 Orbit Propagation Methods

There exist three main methods to propagate the orbit: numerical integration, use of semi-analytical models, and use of analytical models (references [16] and [22]).

Numerical Integration: The position and velocity at future times are obtained by direct numerical integration of the differential equations of motion, while using appropriately accurate expressions for the forces that affect the motion. This is the most accurate method to propagate the orbit.

Analytical Models: For some applications or studies, computation efficiency is preferred to accuracy, and that is when analytical models may be useful. They yield a direct expression of the position and velocity at some future time as a function of the orbit elements given at some initial instant. Analytical models are derived by using a mathematical formulation of the effects of the forces on the spacecraft. For the analytical computation to be possible, the expression of the forces must be simplified, hence the lesser accuracy of these models. One of the most commonly used models is SGP4, one of a group of simplified propagation models, which is used to propagate two-line element sets. SGP4 considers some main effects of the Earth, Sun, and Moon gravity plus a simplified model for atmospheric drag (reference [32]).

Semi-analytical Models: These models are mainly used for accurate propagation over long periods of time. Analytical expressions of the long-term effects of the forces (described more accurately than in the case of analytical models) are first obtained. These effects are then integrated numerically. As only the long-term effects are considered, large time steps can be used in the integration process, enabling computation time savings. An example of a semi-analytical model is the Draper Semianalytic Satellite Theory (DSST) (reference [33]).

5.2.4 ORBIT MANEUVERS

5.2.4.1 Thrust Forces

The motion of a spacecraft is affected by natural forces. Spacecraft motion may also be affected by the action of an onboard propulsion system. The system’s thrusters are frequently applied for orbit control, attitude control, or a combination of both, and exhibit a variety of performance levels and burn durations. Thrust forces may also be generated by spacecraft

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venting or out-gassing. The mathematical model used for trajectory prediction must factor in the impact of thrust forces, as well as the impact of maneuvers executed for spacecraft attitude.

5.2.4.2 Impulsive Maneuvers

In many cases thrust forces may be modeled as impulsive maneuvers. These are described by a velocity vector $\Delta\vec{v}$, applied to the spacecraft at a specified epoch with a burn duration $\Delta t = 0$.

5.2.4.3 Simplified Modeling of Extended Orbit Maneuvers

For extended maneuvers, a simplified model with the assumptions of constant thrust and mass flow rate is sufficient in most cases.

With

$\vec{F}$: Thrust force vector (assumed as constant during the maneuver)
$|\dot{m}|$: Mass flow rate (assumed as constant during the maneuver)
Δt: Maneuver burn duration
m_0: Spacecraft mass at start of the maneuver

The total velocity increment experienced by the spacecraft is computed in this case as follows:

$$\Delta\vec{v} = -\frac{\vec{F}}{|\dot{m}|} \cdot \ln\left(1 - \frac{|\dot{m}| \cdot \Delta t}{m_0}\right)$$

5.2.4.4 Exact Modeling of Extended Orbit Maneuvers

In cases of high precision orbit computation, a more refined numerical modeling of the time-dependent functions of thrust and mass flow rate is applied.

5.2.5 EPHEMERIS REPRESENTATIONS OF TRAJECTORIES

Orbital trajectories can be described analytically in several ways. Chebyshev polynominals are one such representation. The roots of the degree-n polynomial are used as interpolation nodes and provide a smooth function to represent an orbit (reference [5]). Another manner to represent a trajectory is to use a tabular format, with state vectors at pre-determined time intervals. This format (referred to as an ephemeris representation) requires the use of interpolation techniques to interpret the position and velocity at times different from the tabular epochs. An ephemeris may be definitive up to the observation cut-off time, predictive after that observation cut-off time, or a combination of both definitive and predictive.

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5.3 ATTITUDE

5.3.1 GENERAL

The motion of a rigid body is specified by its position, velocity, attitude, and angular velocity. The attitude of a rigid body is the orientation of a body-connected frame in a 3-dimensional space at a given time, with respect to a defined reference frame. Participant attitude is the orientation, at each time, of a participant with respect to a known reference (e.g., celestial objects, frame, etc.). Attitude motion describes the attitude evolution around its center of mass in a defined reference frame. Understanding the attitude requires knowing: how it is determined/estimated, how it is controlled, and how its future changes are predicted. References [12], [16], and [30] provide additional detail.

5.3.2 ATTITUDE ESTIMATION

Attitude estimation, or determination, is the process of computing a set of parameters that describe spacecraft orientation using measurements (generally onboard measurements). Because attitude determination accuracy depends on data from remote, onboard attitude sensors, determination of the quality, change of quality, and improvement of quality (calibration) are inseparably associated with attitude determination. All available attitude measurements can be processed to compute a best-estimate time history of spacecraft attitude. This history is called 'definitive attitude'. Most spacecraft now carry Onboard Computers (OBC) with the capability of computing the spacecraft's own attitude parameters. Some attitude sensors now contain internal computers that are capable of autonomous attitude determination, such as star trackers.

5.3.3 ATTITUDE REPRESENTATIONS

Several attitude representations are available, and the particular representation is generally chosen to suit the attitude stabilization mode of the spacecraft. Examples of stabilization modes include:

– single axis (spinning);

– three axis;

– gravity gradient;

– uncontrolled.

Because of this wide domain of configurations it is convenient to use a single representation to describe the status of the attitude. See reference [17] for a survey of attitude representations. This mathematical representation of a rigid body's attitude is called a 'quaternion'. As it is non-ambiguous and singularity free, it is the most convenient for attitude kinematics, and can be used for every attitude stabilization mode. Quaternions unambiguously define the attitude; however, because in some cases where attitude cannot be completely determined, or complete attitude determination is not required, attitude may not unambiguously define a quaternion.

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The attitude elements needed are as follows:

– given time;

– quaternion at the given time;

– description of body frame;

– description of reference frame.

NOTE – Often the reference frame is an inertial reference frame (J2000, for example). In the following, the transformations will be written generically as transforming from a frame A to a frame B. Frame A is typically the reference frame and frame B is the body frame.

The attitude of the body frame with respect to the reference frame is represented by a unique rotation around a vector **e**, which is invariant in both frames, with an angular amplitude Φ. The vector **e** is oriented in such a way that makes Φ positive directly around the **e** vector in the movement from frame A to frame B.

At this rotation is associated a unit quaternion Q = {**e** sin(Φ/2), cos(Φ/2)}. The scalar component of this 4-vector, cos(Φ/2) ≡ QC, is written as the last component per CCSDS convention. Care must be taken to ensure that the same convention is used by exchange participants. In the following description the convention placing the scalar last is used, and is consistent with the definition in the SANA registry.

The attitude quaternion is defined by a 4-dimension vector $Q = \begin{bmatrix} Q1 \\ Q2 \\ Q3 \\ QC \end{bmatrix}$ with:

$$Q1 = e_1 * \sin(\Phi/2);$$

$$Q2 = e_2 * \sin(\Phi/2);$$

$$Q3 = e_3 * \sin(\Phi/2);$$

$$QC = \cos(\Phi/2).$$

Where Φ is the Euler rotation angle between frame A and frame B and e_1, e_2, and e_3 are the components of the Euler unit rotation vector **e** with the relation

$$Q1^2 + Q2^2 + Q3^2 + QC^2 = 1$$

Also defined is the conjugate quaternion $Q^* = \begin{bmatrix} -Q1 \\ -Q2 \\ -Q3 \\ QC \end{bmatrix}$ which defines the transformation from frame B to frame A (when Q defines the transformation from frame A to frame B).

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Given

$\mathbf{x}_A$ the components of a vector in frame A, with $\mathbf{x}_A = (\mathbf{x}_{A1}, \mathbf{x}_{A2}, \mathbf{x}_{A3})$

$\mathbf{x}_B$ the components of a vector in frame B, with $\mathbf{x}_B = (\mathbf{x}_{B1}, \mathbf{x}_{B2}, \mathbf{x}_{B3})$

$\mathbf{x}_B$ and $\mathbf{x}_A$ are linked by

$$\begin{bmatrix}\mathbf{x}_B \\ 0\end{bmatrix} = Q \otimes \begin{bmatrix}\mathbf{x}_A \\ 0\end{bmatrix} \otimes Q^* \quad \text{and} \quad \begin{bmatrix}\mathbf{x}_A \\ 0\end{bmatrix} = Q^* \otimes \begin{bmatrix}\mathbf{x}_B \\ 0\end{bmatrix} \otimes Q$$

(These products are defined by the quaternion algebra.)

The above transformation can equivalently be represented using a direction cosine or rotation matrix, M_{BA}, to transform from frame A to frame B, where M_{BA} is a function of the quaternion components.

$$\mathbf{x}_B = M_{BA} * \mathbf{x}_A$$

$$M_{BA} = \begin{bmatrix} Q1^2 - Q2^2 - Q3^2 + QC^2 & 2(Q1Q2 + Q3QC) & 2(Q1Q3 - Q2QC) \\ 2(Q1Q2 - Q3QC) & -Q1^2 + Q2^2 - Q3^2 + QC^2 & 2(Q2Q3 + Q1QC) \\ 2(Q1Q3 + Q2QC) & 2(Q2Q3 - Q1QC) & -Q1^2 - Q2^2 + Q3^2 + QC^2 \end{bmatrix}$$

The following formulae give the relations for the associated quaternion (reference [12]):

$$Q1 = (M_{23} - M_{32})/(4*QC)$$
$$Q2 = (M_{31} - M_{13})/(4*QC)$$
$$Q3 = (M_{12} - M_{21})/(4*QC)$$
$$QC = +/-(M_{11}+M_{22}+M_{33}+1)^{1/2}/2$$

where M_{ij} is the element from row i and column j of M_{BA}. In addition, it is customary to restrict the representation of QC to positive (reference [30]).

The matrix M_{BA} can be used to elaborate a set of attitude angles like Euler angles (Roll, Pitch, Yaw) giving the rotation angles around X=1=roll, Y=2=pitch, Z=3=yaw axes. The rotation order must be defined to have a set of values consistent with the desired rotation.

For example, if the rotation order is Φ around axis 3, followed by Θ around axis 1,followed by Ψ around axis 2, the Euler angles Φ, Θ, and Ψ can be obtained by the following relations:

$$\Phi = \tan^{-1}\left(\frac{-2(Q1Q2 - Q3QC)}{-Q1^2 + Q2^2 - Q3^2 + QC^2}\right)$$

$$\Theta = \sin^{-1}\left(2(Q2Q3 + Q1QC)\right)$$

$$\Psi = \tan^{-1}\left(\frac{-2(Q1Q3 - Q2QC)}{-Q1^2 - Q2^2 + Q3^2 + QC^2}\right)$$

NOTE – The angles Φ and Ψ are undefined if cos(Θ) = 0. Several solutions are possible depending on the quadrants in which the inverse trigonometric function solutions are taken.

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The definition of roll, pitch, and yaw axes (Euler angles) varies from mission to mission and often even changes within a mission (reference [16]). In determining rotations or rotation rates about these axes, care must be taken to define the axes, which are often misunderstood. Using quaternions avoids the need to use an Euler angle attitude representation and its resulting complexity.

Spin-stabilized spacecraft rotate about a known body-frame spin-axis. Their attitude is often represented as the orientation of the spin-axis with respect to an external reference frame, a phase angle of rotation about this axis, and/or a spin rate about this axis. Generally, the phase of rotation about the spin axis is neither determined nor needed. In this case the spacecraft attitude is sufficiently determined by only two parameters, and a quaternion is not unambiguously specified by the conventionally determined attitude. In this case the most common attitude representation is the right ascension and declination of the spin-axis in the reference frame.

5.3.4 EPHEMERIS REPRESENTATIONS OF ATTITUDE

Under the proper conditions, attitude states allow for the use of a propagation technique (analytical or numerical) to interpret the orientation of an object connected frame at times different from the specified epoch. Another manner to represent attitude is to use a tabular format, with attitude states at pre-determined time intervals. This format (referred to as an ephemeris representation) requires the use of interpolation techniques to interpret the attitude at times different from the tabular epochs.

5.3.5 ATTITUDE DYNAMICS

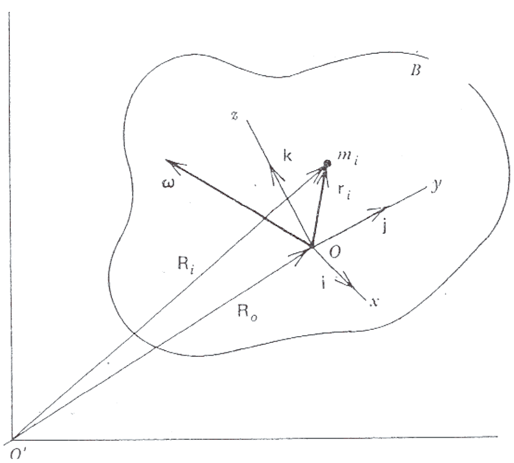

Figure 5-2: Rigid Body Angular Momentum

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Consider the situation of figure 5-2 in which the rigid body B contains the body-fixed x, y, z coordinate system attached at the center of mass, O. The angular momentum is

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$$H_{SC} = \int_B \mathbf{r} \times (\omega_{SC} \times \mathbf{r})\, dm$$

where ω_{SC} denotes the instantaneous angular velocity vector of the rotating x, y, z coordinate system, resolved in the spacecraft coordinate system:

$$\omega_{SC} = \begin{bmatrix} \omega_x \\ \omega_y \\ \omega_z \end{bmatrix}$$

The angular momentum may be written in matrix form as

$$H_{SC} = I_{SC}\omega_{SC}$$

where, I is known as the inertia matrix or the inertia tensor

$$I_{SC} = \begin{bmatrix} I_x & -I_{xy} & -I_{xz} \\ -I_{xy} & I_y & -I_{yz} \\ -I_{xz} & -I_{yz} & I_z \end{bmatrix}$$

The individual terms of the inertia matrix are as follows:

$$I_x = \int_B (y^2 + z^2)dm, \quad I_y = \int_B (x^2 + z^2)dm, \quad I_z = \int_B (x^2 + y^2)dm$$

$$I_{xy} = \int_B xydm, \quad I_{xz} = \int_B xzdm, \quad I_{yz} = \int_B yzdm$$

I_x, I_y, I_z are the moments of inertia of the body about the x, y, and z axes, respectively, I_{xy}, I_{xz}, I_{yz} are the products of inertia of the body B. Thus, any rigid body is characterized by a set of constants I_x, I_y, I_z, I_{xy}, I_{xz}, and I_{xz} for the purpose of analyzing the angular momentum and, ultimately, attitude control. The torque vector satisfies Euler's Equation,

$$\tau = \frac{d\mathbf{H}_{SC}}{dt} = I_{SC}\dot{\omega}_{SC} + \omega_{SC} \times I_{SC}\omega_{SC}$$

which equals zero in a torque-free environment. As a parallel to the conservation of linear momentum in orbital mechanics, the principle of conservation of angular momentum states that angular momentum remains constant in magnitude and direction in inertial space, if the body is not acted on by any external torques. For spacecraft with devices that generate angular

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momentum internally, such as reaction wheels, additional angular momentum terms must be included in Euler's equation.

The kinetic energy

$$T = \frac{1}{2}\omega_{SC}^{T} I_{SC} \omega_{SC}$$

is reminiscent of the familiar

of particle dynamics with the spacecraft angular velocity playing the role of the linear velocity and the spacecraft inertia matrix being the analog of the mass for rotational systems.

A principal axis is any axis $\hat{\mathbf{P}}$ such that the resulting angular momentum is parallel to $\hat{\mathbf{P}}$ when the spacecraft rotates about $\hat{\mathbf{P}}$. If the principal axes are used as the coordinate axes of the spacecraft x, y, z reference frame (see figure 5-2), the inertia matrix is the diagonal matrix of eigenvalues (characteristic values), called the principal moments of inertia. References [12], [18], [19], and [20] provide additional detail.

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5.3.6 ATTITUDE PREDICTION

Attitude prediction is the process of forecasting the orientation of the spacecraft by using dynamical models to extrapolate the attitude history. Propagation is the process of using the dynamic equations of motion (EOM) and mathematical models of environmental torques to model the attitude for an extended period of time. Environmental torques that are typically included are gravity gradient, aerodynamic torque (if applicable), solar pressure, differential gravity and self-gravity. Which torques are important will depend on the spacecraft location.

Differential and self-gravity arise from gravity forces acting on a particular point in space, and these will eventually be dominant terms if the spacecraft is far enough away from a gravity source, such as a planet, sun, or moon.

To propagate the attitude of a spacecraft without a gyro, a simple algorithm is typically followed:

a) Estimate the environmental torques on the body.

b) Integrate the momentum equation to determine the new momentum state.

c) Recover the rate from the new momentum state.

d) Propagate the attitude quaternion.

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5.4 COVARIANCE MATRIX

Covariance matrices are often used in the processes of orbit determination, orbit propagation, attitude determination, and attitude propagation to quantify uncertainties and provide an estimate of accuracy. Covariances are the expected values of correlations and cross-correlations among the elements relative to their mean values. They constitute a measure of the interdependence of the variables (measurements or predictions). The covariance of an element with itself is the variance, the square root of which is the standard deviation from the mean. The correlation coefficient for two variables is their covariance divided by the product of their individual standard deviations. The number of 'solve fors' in the orbit/attitude determination process determines the dimension of the covariance matrix. If there are 'n' state variables, the covariance is an $n \times n$ matrix. The matrix will be diagonal if all state variables are completely independent. The matrix will be symmetric if correlations and cross-correlations are governed by Gaussian (normal distributions). Reference [12] provides additional detail.

5.5 PHYSICAL PROPERTIES

5.5.1 PARAMETERS

The parameters for the simplified models discussed in 5.5.2 and 5.5.3 are as follows: C_R, A_R, C_D, A_D. The meanings of these parameters are defined below.

5.5.2 SOLAR RADIATION PRESSURE (SIMPLIFIED MODEL)

The absorption or reflection of photons associated with solar radiation causes a spacecraft to accelerate. For most applications of navigation data exchange, a simplified model that assumes the surface normal of the spacecraft is pointing to the Sun is sufficient to account for the effect of solar radiation. The following model for the acceleration of a satellite due to solar radiation pressure may be used:

$$\ddot{\vec{r}}_R = -C_R \frac{A_R}{M} \frac{\Phi}{c} \frac{\vec{r}_S}{r_S^3} AU^2$$

where:

- C_R: Solar radiation pressure coefficient
- A_R: Effective satellite cross section area for solar radiation pressure (m^2)
- M: Spacecraft mass (kg)
- Φ: Solar flux at 1 AU (≈ 1367 Wm^{-2})
- c: Speed of light (ms^{-1})
- $\vec{r}_S$: Vector spacecraft-Sun (m)
- AU: Astronomical unit (m)

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5.5.3 ATMOSPHERIC DRAG (SIMPLIFIED MODEL)

For low altitude satellites, the interaction of particles from the central body's atmosphere with the spacecraft surface causes acceleration. For most applications of navigation data exchange, a simplified model that assumes a spherical shape of the satellite with a unique surface is sufficient to account for the effect of atmospheric drag. The following model for the acceleration of a satellite due to atmospheric drag may be used:

$$\ddot{\vec{r}}_D = -\frac{1}{2} C_D \frac{A_D}{m} \rho \cdot v_r^2 \frac{\vec{v}_r}{v_r}$$

where:

C_D: Drag coefficient
A_D: Effective satellite cross section area for drag (m2)
m: Spacecraft mass (kg)
ρ: Atmospheric density at spacecraft location (kg m^{-3})
v_r: Velocity of spacecraft relative to atmosphere (m s^{-1})

5.6 HARDWARE PROPERTIES

Hardware properties are associated with a particular subsystem of the entire spacecraft, and not the vehicle as a whole. For example, in addition to the effective areas defined in 5.5 for the entire spacecraft for solar radiation and aerodynamic drag calculations, area information may be exchanged for individual components of significance on the vehicle (such as a solar panel or parabolic antenna). Other examples include the mass flow rate (which can depend on the engines or thrusters being used for a particular maneuver), transmitter and receiver delays (which can be a function of the transponder or transceiver being used), and oscillator frequency and stability (multiple frequency standards can exist on a single spacecraft). Another example is antenna offsets from the center of gravity (CG) and the spin axis of the satellite. Use of these properties enables more complex models of solar radiation pressure and atmospheric drag to be formulated.

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6 MEASUREMENTS

6.1 GENERAL

Spacecraft navigation is based on measurements including velocity, distance, and angular direction. Data for orbit determination is obtained from tracking radio frequency or optical laser signals, telemetry or radar tracking signals, and a variety of other sources including onboard sensors (references , [7], [14], [15], and [22]). Spacecraft attitude estimation and control are based on onboard measurements (reference [12]).

6.2 ORBIT DETERMINATION MEASUREMENT DATA TYPES

6.2.1 ANGLES

6.2.1.1 General

Many tracking stations are able to measure the angles from a ground station to a spacecraft. These angles are a fundamental data type for orbit determination in many missions, particularly in launch support (initial acquisition, Launch and Early Orbit Phase [LEOP]) and at other times when the spacecraft is close to the Earth. The angles measured in this fashion are only useable in the proximity of Earth. The angles help to measure plane-of-sky position.

Pointing angles of antennas on relay satellites are available by telemetry. These angles similarly assist in orbit estimation of the user spacecraft, especially when derived from an auto tracking feedback system.

6.2.1.2 Aberration Corrections Per Reference Frame

The reference frame used for angular measurements in equatorial coordinates (right ascension and declination measurements) should be chosen to reflect whether or not annual aberration corrections were applied to the measurements.

Star catalogues are always given in heliocentric/barycentric frames, but the Earth's velocity in that frame (i.e. there is a relative, non-linear motion between the observer and the center of the reference system) causes a small difference between the apparent and actual direction of the stars:

a) the Earth's orbit around the Sun causes annual aberration (up to 20 arcsec);
b) The Earth's rotation causes daily aberration (up to 0.3 arcsec, which is generally smaller than typical telescope astrometric precision and can be ignored).

These two effects do not apply to observations of satellites orbiting the Earth as viewed from an Earth ground telescope, as the center of the topocentric reference system is the telescope and the satellite orbits the Sun at the same rate as the Earth, so there is no relative motion due to the Earth's orbit and the apparent and actual direction are the same (ignoring daily aberration effects).

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Telescope astrometric data reduction software determines the actual right ascension and declination of a satellite in a detector coordinate frame, i.e. based on the values for known stars, by assuming the satellite is also a star, several parsecs away. The values need to be converted back to the apparent ones, by applying the annual aberration correction. If the optical telescope is also in orbit around Earth, an additional correction of the aberration per orbit needs to be applied. The corrections can be computed based solely on the right ascension and declination values of the observations, and the related observation epoch.

For telescopes in heliocentric orbit or in orbit around celestial bodies other than Earth, the appropriate aberration corrections to the angle data for that orbital reference frame should be applied.

Typical orbit determination software expects that right ascension and declination are the actual, values for an Earth-orbiting satellite, ie annual aberration correction should be applied before orbit determination. The following convention should be used if necessary to provide a reference frame for the tracking asset used:

a) If the annual aberration correction was not applied, a barycentric reference frame (e.g. ICRF) should be used.
b) For ground-based sensors, if the annual aberration correction was applied, a geocentric reference frame (e.g. GCRF, EME2000) should be used.
c) For space-based sensors, no corrections should be applied and a barycentric frame (e.g. ICRF) should be used. If corrections are applied, the type of corrections applied and the associated reference frame should be specified in an ICD.

More details on annual aberration and its computation can be found in reference [2].

6.2.2 RADIOMETRIC TRACKING DATA

6.2.2.1 General

Spacecraft tracking is the process that provides the measurements (observables) needed to determine where the spacecraft is located (its state) in its trajectory at a particular time. Tracking data is obtained from the spacecraft in flight as it passes within the field of signal acquisition from one participant to another. The ground- or space-based measurements used to navigate spacecraft can be derived from the radio frequency (radio) link between the spacecraft and the tracking stations on the Earth and referred to as radiometric tracking (references [6], [8], [13], [14], [22], [23], and [27]). In the case of a relay satellite system, the ground-based measurements have additional signal paths to model between the spacecraft and the relay and a reference link (reference [34]). Each link may also be established in the optical realm (via a laser system) with the navigation measurements referred to as optimetrics. For ease of explanation, the ensuing text that refers to radio frequency (radio) links, also applies to optical links unless noted otherwise.

The primary navigation measurements that are obtained by the radio system are range along the line of sight (summed among all radio link paths), Doppler along the line of sight, and Delta-Differential One-Way Range (ΔDOR). These three data types are complementary since

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range and Doppler provide line-of-sight information while ΔDOR adds the orthogonal (plane-of-sky) directions. Other data types include differenced range, differenced Doppler, and Differenced Range Versus Integrated Doppler (DRVID)

6.2.2.2 Link Paths

The radio signal transmitted from a ground station located on the surface of the Earth to a spacecraft is known as an uplink. The transmission from spacecraft to the Earth is a downlink. The transmission from a relay satellite to a user spacecraft is known as the forward link. The transmission from a user spacecraft to a relay satellite is known as the return link. A ground-to-flight link may be either an uplink or an uplink combined with forward link(s), and a flight-to-ground link consists of a downlink following any return links. Figure 6-1 provides a diagram of these links. These links may consist of a pure RF tone or optical stream, called a carrier. Such a pure carrier is useful in many ways, including radio science experiments and Doppler frequency shift measurement. On the other hand, carriers may be modulated to carry information in each direction. Commands may be transmitted to a spacecraft by modulating the uplink carrier (and forward link carrier if via a relay satellite). Telemetry containing science and engineering data may be transmitted to the Earth by modulating the downlink carrier (and return link carrier if via a relay satellite) (references [14], [21], [23], and [34]). Ranging is accomplished through a series of specified tones or via Pseudo Random Noise coded signals. To maintain coherency, an additional reference link may be included between the ground station and the relay (reference [34]).

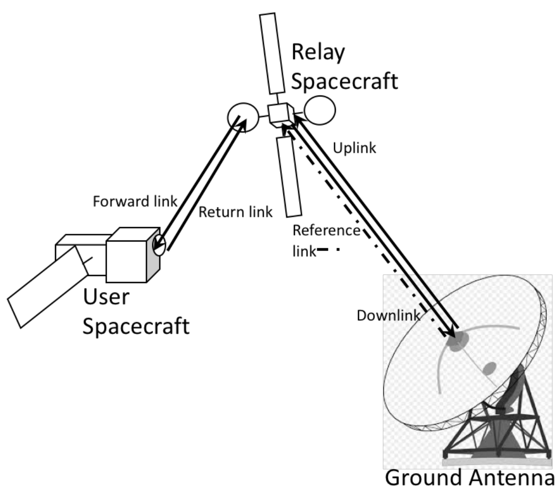

Figure 6-1: Link Terminology Including a Relay

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6.2.2.3 One-Way, Two-Way, Three-Way Data

Figure 6-2 shows the configurations for link paths associated with one-way forward, two-way, and three-way data types. (Note: one-way return is not shown in Fig. 6-2.) When an Earth-based station is only receiving a flight-to-ground link from a spacecraft, the communication is called 'one-way Return' or downlink. When the station sends a ground-to-flight link that the spacecraft receives, the communication is called 'one-way Forward'or uplink (see Fig. 6-2, left diagram). When the spacecraft receives that uplink from the station and at the same time the spacecraft transmits a flight-to-ground link that is received at the Earth, the communications mode is called 'two-way' (see Fig. 6-2, center diagram). If the one-way light time between the ground and the spacecraft is significantly long, then the uplink and downlink portions of the two-way link may be received and transmitted at the same time onboard the spacecraft, but the ground station transmission and reception may not occur at the same time.

The communications mode is still called one-way Return even when an uplink is being received by the spacecraft, but the full round-trip light time has not elapsed. Consider the following situation: The station is receiving the downlink and watching telemetry that shows the state of the spacecraft's own receiver. As long as the spacecraft's receiver is not receiving the uplink, communications are called 'one-way Return'.

After the spacecraft's receiver has locked onto the uplink/forward link, it is considered 'two-way'. There are two modes for two-way data: coherent, where the return/downlink signal is directly and proportionally related to the forward/uplink signal; and non-coherent, where the return/downlink signal is generated independently from the forward/uplink signal. Two-way coherent data relies on a single reference source typically at the ground station, whereas two-way non-coherent data relies on two independent reference sources, one at the ground station and one on the satellite. A signal may be considered one-way Forward if the spacecraft receives an uplink from the ground station (and a forward link from a relay), with two different reference sources: one at the ground station and one onboard the spacecraft of interest.

If the receiving station on the Earth is different from the station that uplinked the signal to the spacecraft, the tracking mode is called 'three-way' (see Fig. 6-2, right diagram). The three-way mode is employed when: (1) the round-trip light time (RTLT) is large enough that the transmitting station rotates out of sight of the spacecraft before the signal returns to the Earth, and, therefore, a second station must be employed to receive the signal; or (2) continuous tracking of a spacecraft is desired across the transition from the view period of one tracking complex to the view period of another; or (3) simultaneous geometric diversity must be obtained for tracking data.

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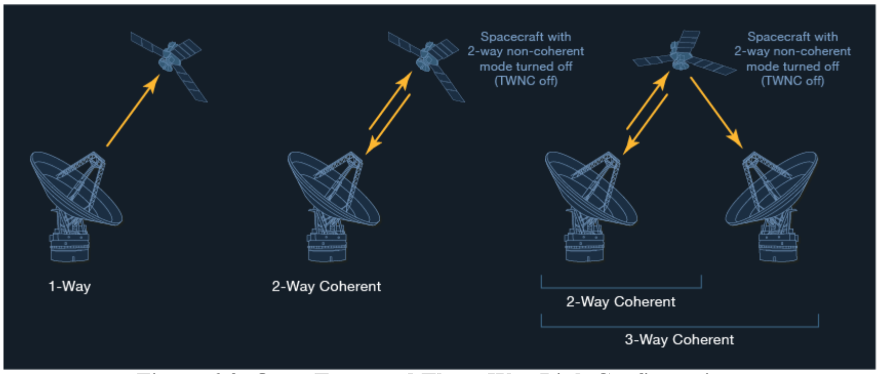

Figure 6-2: One-, Two-, and Three-Way Link Configurations

6.2.2.4 Four-Way Data

Data is referred to as 'four-way' if there are four or five directed legs and three nodes. In one example scenario, a signal might be sent from a ground-based station to a relay satellite, on to another satellite, back to the relay satellite, and then back to the same ground-based station, as portrayed in Fig. 6-1. Because relay systems often incorporate a reference signal to maintain system-wide coherency between the ground station, relay, and user spacecraft, the coherent range and Doppler data paths are referred to as five-way, incorporating the reference link as one path link.

In another scenario, the signal might be sent from the station on the ground to a relay satellite, on to another satellite, and to a different ground-based station or relay satellite and ground station. Another term for this scenario is hybrid support tracking and is portrayed in Fig. 6-3.

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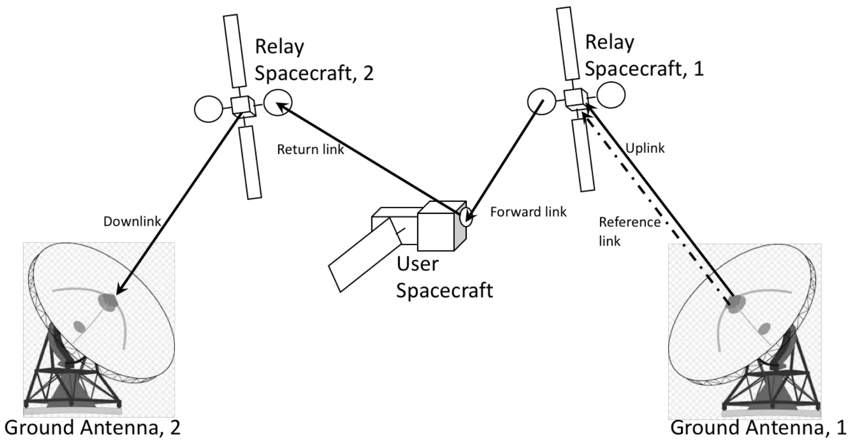

Figure 6-3: Hybrid Tracking Link Configuration With Two Relays

6.2.2.5 Coherence

Aside from carrying the information modulated on the downlink as telemetry, the carrier itself is used for tracking and navigating the spacecraft, as well as for carrying out some types of science experiments such as radio science or gravity field mapping. For each of these uses, an extremely stable downlink frequency is required onboard the spacecraft and at the ground station, so that Doppler shifts on the order of fractions of a hertz may be detected out of many GHz over periods of many hours with resolutions below millihertz. Currently only Global Navigation Satellite System (GNSS) satellites carry the equipment required to maintain such frequency stability, although miniature atomic clocks for satellite use are being developed. Spacecraft transmitters are subject to wide temperature changes, which cause their output frequency to drift. The solution is to have the spacecraft generate a downlink that is coherent in translated frequency, phase, and code (if applicable) to the uplink it receives.

This coherent signal is generated on the Earth with the aid of an atomic-based frequency standard phase-locked to a low-phase noise crystal oscillator, in an environmentally controlled room, sustained by an uninterruptible power supply. This standard is used as a reference for generating an extremely stable (over short and long time intervals) uplink frequency for the spacecraft to use, and in turn, to generate its coherent downlink, and for a relay satellite to also produce a coherent forward link and downlink. (It also supplies a signal to the master clock that counts cycles and distributes UTC time.) For example, a hydrogen-maser time and frequency reference system stability is equivalent to the gain or loss of 1 second in 30 million years.

Once the spacecraft receives the stable uplink/forward link frequency, it multiplies that frequency by a predetermined constant (the ‘turnaround ratio’), and uses that value to generate

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its return link/downlink frequency. In this way, the coherent downlink assumes all the extraordinarily high stability in frequency that belongs to the sensitive equipment that generated the uplink/forward link. It can thus be used for precisely tracking the spacecraft and for carrying out precision science experiments. The reason a spacecraft's transponder multiplies the received frequency by a constant is to assure the return/downlink it generates will not cause interference with the forward/uplink being received.

One-way data is classified as non-coherent. The spacecraft carries a low-mass oscillator to use as a reference in generating its return/downlink for periods when a forward/uplink signal is either not available or is not provided to the transmitter. But this oscillator is not necessarily highly stable, since its output frequency is affected by temperature variations. Some spacecraft carry an Ultra-Stable Oscillator (USO), but even the USO does not match the ideal stability of a well-developed coherent link.

6.2.2.6 Range

Range is the distance along the line of sight between the flying spacecraft and another participant or participants. A range measurement is obtained by determining the RTLT of the radio signal after accounting for a number of systematic delays (ionospheric, tropospheric, and solar plasma media; onboard instrument systems; ground-based antenna systems; and clock drift on the spacecraft and at the stations). Several units are commonly used for range data, specifically:

– kilometers (km) because range is a radial distance;

– seconds (s) because range measurements are instrumentally measured by a time delay; and

– 'range units' (RU), a function of the uplink frequency to the spacecraft, typically used when the uplink frequency is not fixed.

Range is generally considered a two-way measurement with the same time reference source for receiver and transmitter that provides RTLT. However, a one-way link can provide a pseudo-range measurement that provides a one-way light time (OWLT) measurement. Systems tailored for navigation, such as GNSS, provide information in data on the link that allow for clock resolution and time transfer between the transmitter and receiver to enable a pseudo-range measurement.

6.2.2.7 Doppler

The spacecraft topocentric range-rate is obtained from the Doppler shift in the frequency of the radio signal produced by the relative motion along the line of sight from the station to the spacecraft. Because the spacecraft is moving with respect to the Earth, the onboard radio system receives an electromagnetic wave whose frequency differs from the one transmitted from the ground. The frequency of the received signal increases if the spacecraft is moving toward the Earth and decreases if the spacecraft is receding. For relay systems, the Doppler is

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measured along the line of sight between the relay and the user craft. Doppler on the ground-to-spaceflight link to the relay may also be accounted for in the total measurement, but some systems remove that contribution. Doppler accuracies are often given in terms of radial velocity or the hertz unit of frequency, but the fundamental measurement is range change.

The least accurate Doppler measurement is the one-way (non-coherent) mode, in which the frequency of the signal received by the tracking station is compared with the best estimate of the frequency of the signal sent by the onboard oscillator. The most accurate measurements are obtained in the two-way coherent mode, for which the transmitting and receiving stations, and hence the frequency standards, are the same.

6.2.2.8 Delta-Differential One-Way Range

The Delta-Differential One-Way Range (ΔDOR) data type provides angular information in the plane-of-sky orthogonal to the line-of-sight direction. Two tracking stations separated by a long baseline (on the order of several thousand kilometers) track a single spacecraft simultaneously. Each station makes high-rate recordings of the downlink's wave fronts while maintaining precise timing data. Stations also record the pointing angles of their antennas, which slew directly to the position of an extragalactic object (usually a quasar) the position of which is accurately known. The antennas then slew back to the spacecraft. This data type measures geometric delay of the signal by cross-correlating the signal from the two geometrically separated stations, and performs a double-differencing to cancel a large portion of common error sources such as instrumental effects, clock errors, media effects, and baseline uncertainties. The result is a very precise triangulation from which angular position in the plane-of-sky may be determined. The ΔDOR observing geometry is shown in figure 6-4 (reference [28]).

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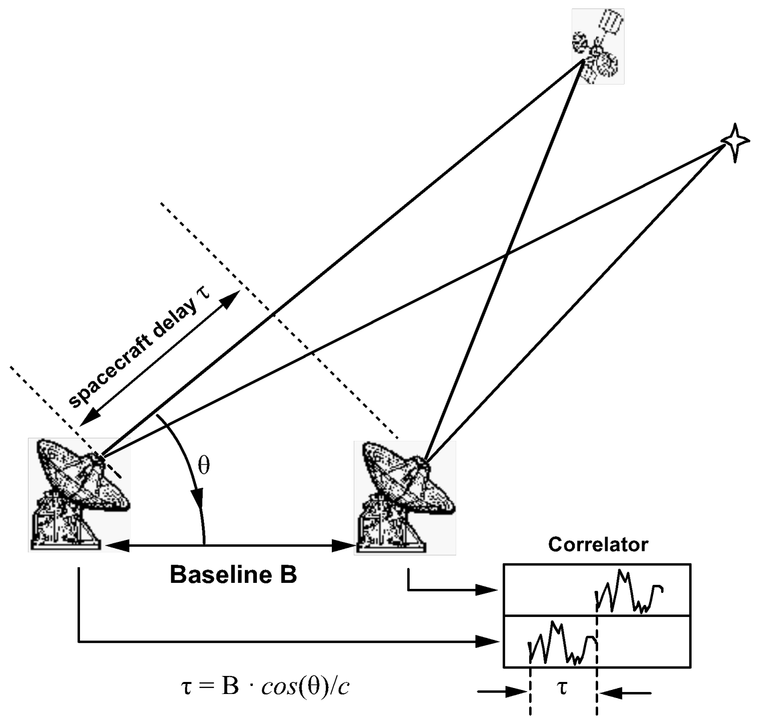

Credit: C.D. Edwards, J.S. Border, J.E. Patterson

Figure 6-4: ΔDOR Observation Geometry

6.2.2.9 Differenced Range

Simultaneous reception of ranging signals by two complexes during view period overlaps can provide a measure of angular position. Such a measurement type might be considered when the spacecraft is at a low declination (δ), because the uncertainty in the declination determined from other data types is proportional to 1/sinδ. This data type is referred to as differenced two-way and three-way range and is illustrated in figure 6-5. The limiting errors for these observables are uncalibrated biases due to clock offsets and instrumental delays at the two stations. This data type can be operationally difficult because of the round-trip light time and the uplink handover from one station to the other. Furthermore, as the time between two-way measurements increases, the differenced observables are increasingly contaminated by uncalibrated space plasma and other line-of-sight delay variations.

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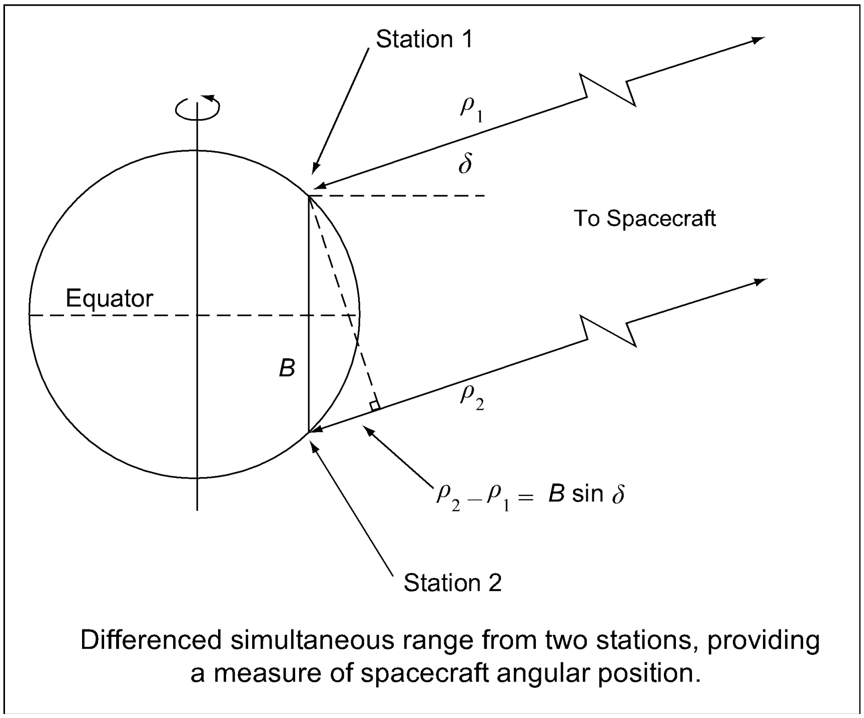

Figure 6-5: Differenced Range

6.2.2.10 Differenced Doppler

Doppler signals received simultaneously by two complexes or relays during view-period overlaps can be differenced to produce the data type known as differenced Doppler. For a planetary orbiter, the orientation of the orbit plane about the line of sight from the Earth to the planet is not determined from Doppler or range measurements as accurately as the other components of state. The orientation component may be directly observed by Doppler data acquired simultaneously at two stations and then differenced. For two spacecraft in orbit about the same planet, which may be observed simultaneously in the same beamwidth of Earth-based tracking antennas, differential measurements may dramatically improve orbit accuracy for both spacecraft. Similarly, for relay-based systems, differencing of one-way Doppler from an orbiting spacecraft that is received by two relay assets can remove phase noise errors on the spacecraft, for example, from an onboard oscillator.

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6.2.2.11 Combined Range, Doppler and Interferometric Tracking Data

Combining range and interferometric observables is an alternative to using long, continuous Doppler arcs for cruise navigation. In this method, the three components of spacecraft position are directly measured in just a few minutes. Doppler data may then be applied to infer better force models of the spacecraft's dynamics without the problem of aliasing model parameters into weakly observed spacecraft state components. Simultaneously fitting all data types leads to improved navigation reliability and robustness. Range measurements with sub-meter accuracy would have application to the relative tracking of planetary orbiters, rovers, and landers (reference [15]).

6.2.3 NON-GRAVITATIONAL ACCELERATION

Non-gravitational accelerations such as radiation pressure, atmospheric drag, and thrust can be measured by accelerometers and other devices.

6.2.4 SPACE BASED POSITIONING NETWORKS

The Global Positioning System (GPS), and other similar systems that form the Global Navigation Satellite System (GNSS), provide several observable quantities to the user's receivers, including the PN code and the carrier. From the phase of the code after ambiguity resolution, the receiver can extract the one-way range, known as pseudorange, from the system space vehicle(s). From the frequency of the carrier, the receiver can extract the Integrated Doppler Count. From the phase of the carrier, the receiver can extract carrier phase or accumulated Doppler range. Apart from these observables, the receiver can get the GNSS system time. GNSS systems provide information about the constellations ephemeris on the data channel, which can be used directly by the receiver or used in concert with augmented corrections (reference [4]). The relevant GNSS measurement is referred to as the PVT (Position, Velocity and Time) measurement. Receivers are now available that can use multiple GNSS systems and signals.

6.3 ATTITUDE DETERMINATION MEASUREMENT DATA TYPES

6.3.1 GENERAL

Many kinds of attitude detectors or sensors are available, and the spacecraft equipment depends mainly on the mission. Although at this time there is no CCSDS standard for exchange of attitude measurements and the current Recommended Standards apply only to transfer of attitude state and/or attitude ephemeris, examples are given below of the types of data that are needed for attitude determination. Agency-specific processes use attitude data measurements to produce attitude states and attitude ephemerides, which are the subject of CCSDS Recommended Standards.

All attitude measurements are determined in the frame of the sensor that makes the measurement. In order to combine measurements from separate sensors in attitude

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determination, they must first be transformed into a common frame—usually the spacecraft body frame.

Alignment of the sensor frame with the body frame is a rotation that transforms measurements in the sensor frame to the body frame. In order to compute attitudes (rotations between the body frame and a reference frame); measurements must be converted to the body frame using the sensor alignments. Effects of shifts in alignment may be minimized through sensor alignment calibration.

6.3.2 ANGLES

A primary source of attitude data is the angle from a sensor frame to an observed reference body. For some sensors, a pair of angles is used to completely specify the direction from the spacecraft to the reference body in the sensor frame. In spin-stabilized spacecraft, a single angle often describes the angle between a sensor axis (usually the spacecraft nominal spin axis) and the reference body. In either case, the alignment of the sensor frame with respect to the body frame is required in order to use the angles for attitude determination.

In another context, relative rotation angles are obtained at regular intervals from some attitude rate sensors. The time derivatives of these angles provide angular rates that are important in more accurate attitude determination algorithms.

Typical angle-based sensors are sun sensors, earth sensors, and star sensors.

6.3.3 ANGULAR RATES

Onboard measurements of angular rates (or derivatives of relative rotation angles) use two sensor measurements at different times in the same attitude determination. These rate measurements are the change of the orientation of the rate sensor with respect to the reference frame. The alignments of rate sensor frames with respect to the body frame must also be known to use them for attitude determination.

6.3.4 STAR MAGNITUDES

For attitude determination using stars as references, each observed star must be matched with the reference frame direction of a particular star. Matching observations from observed objects with reference stars is star identification, and often uses the star magnitude (in a defined spectral passband). When used in this manner, star magnitudes are attitude measurements.

Star magnitudes are defined for a particular passband (range of wavelengths for which the sensor is tuned). Spectral passband terminology and definitions are generally inherited from astronomical science and need not be rigorously defined as long as the instrument sensitivity approximately matches the passband of the star catalog from which the positions of the reference stars are to be taken. Magnitudes are represented in a negative logarithmic scale

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(negative values are brighter) with a difference of 5 magnitudes representing a factor of 100 in brightness.

Modern star sensors are typically autonomous and produce an attitude quaternion. Internally they use the location of stars in the Field of View (FOV) (and the magnitudes) to determine an attitude quaternion. Then the attitude quaternion can become an attitude data type used in an attitude estimation algorithm.

6.3.5 MAGNETIC FIELDS

In the vicinity of the Earth a spacecraft may measure the direction of the Earth's magnetic field in a detector frame. Such measurements can produce in the sensor frame a magnetic field vector that can be converted into a corresponding body frame vector and used together with an Earth magnetic field model in attitude determination.

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ANNEX A

ABBREVIATIONS AND ACRONYMS

AZ	Azimuth
AZEL	Azimuth and Elevation antenna configuration
BIH	Bureau International de l'Heure
CCSDS	Consultative Committee for Space Data Systems
CIO	Conventional International Origin
DEC	Declination
EL	Elevation
EME2000	Earth Mean Equator and Equinox of J2000
GLONASS	Global Navigation Satellite System, Russian
GM	Gravitational Parameter (gravitational constant times mass)
GMST	Greenwich Mean Sidereal Time
GNSS	Global Navigation Satellite System, generic
GPS	Global Positioning System
HA	Hour angle
IAU	International Astronomical Union
ICRF	International Celestial Reference Frame
ICRS	International Celestial Reference System
IERS	International Earth Rotation and Reference Systems Service
IRM	IERS Meridian
IRP	IERS Reference Pole
ITRF	International Terrestrial Reference Frame (-xx before 2000, xxxx after to indicate year issued)

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ITRS	International Terrestrial Reference System
LEOP	Launch and Early Orbit Phase
LVLH	Local Vertical Local Horizontal
MEME	Mean Equator Mean Equinox
MET	Mission Elapsed Time (time since beginning of mission, typically launch)
OBC	Onboard Computer
OWLT	One Way Light Time
PVT	Position, Velocity and time
QSW	Normalized Radius, Second, Omega (ω)
QZSS	Quasi-Zenith Satellite System
RA	Right Ascension
RADEC	Right Ascension (or Hour Angle) and Declination antenna
RIC	Radial, In-track, Cross-track
RSW	Alternative name for RTN
RTLT	Round Trip Light Time
RTN	Radial, Transverse, Normal
SC	Spacecraft Body Frame
SI	International System of Units
TAI	International Atomic Time
TCB	Barycentric Coordinate Time
TCG	Geocentric Coordinate Time
TDB	Barycentric Dynamical Time
TDR	True of Date, Rotating
TDRSS	Tracking and Data Relay Satellite System

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TDT	Terrestrial Dynamical Time
TEME	True Equator, Mean Equinox
TNW	Tangential, Normal, ω (omega)
TOD	True of Date
TT	Terrestrial Time
UT1	Universal Time
UTC	Coordinated Universal Time
XEYN	X-east and Y-north antenna configuration
XSYE	X-south and Y-east antenna configuration
X-Y	X-Y antenna mount such as XEYN or XSYE
ΔDOR	Delta-Differential One-Way Range data type
δ	Declination

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