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Space systems — Best practices for orbit elements at payload — LV separation

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Space systems — Best practices for orbit elements at payload — LV separation

Systèmes spatiaux — Meilleures pratiques pour les éléments en orbite à charge utile — Séparation LV

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Foreword

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The committee responsible for this document is ISO/TC 20, *Aircraft and space vehicles*, Subcommittee SC 14, *Space systems and operations*.

Introduction

This Technical Report will provide a recommendatory method for post-launch assessment of the orbit elements precision at separation, which is conductive to improving international communication effect and reducing the risks from errors resulting from miscommunication. It can estimate the orbit elements precision at separation, provide the reference for fuel capacity design of launch vehicle and spacecraft, and then help to reduce the manufactory costs of rocket and payload.

Space systems — Best practices for orbit elements at payload — LV separation

1 Scope

This Technical Report provides the best practices for orbit elements at payload-LV separation. It includes orbit elements and calculation conditions, calculation method of orbit elements and their errors at elliptical orbit insertion of various payloads. The fit between the actual and expected values of orbit elements can be used as a criterion of commercial launch.

There are many different sets of orbit elements. Each is best suited for a particular application. The traditionally used set of orbital elements is called the set of Keplerian elements. This Technical Report gives the calculation method of Keplerian elements and the transformation method of all the other orbit elements, in order to satisfy different user's need.

Affected by terrestrial gravitational perturbation, lunisolar gravitation perturbation and other factors, orbit elements change slowly after orbit injection. Orbit elements calculation methods after separation are not included in this Technical Report.

The technical communication and specific progress for orbit elements is relatively easy to be agreed on by applying this Technical Report, which can contribute to avoiding possible disputes.

2 Symbols and abbreviated terms

2.1 Abbreviated terms

BIPM Bureau International des Poids et Mesures

CTP Conventional Terrestrial Pole

GAST Greenwich Apparent Sidereal Time

GMST Greenwich Mean Sidereal Time

GCRF Geocentric Celestial Reference Frame

GPS Global Positioning System

IERS International Earth Rotation and Reference System Service

IRM International Reference Meridian

ITRF International Terrestrial Reference Frame

ITRS International Terrestrial Reference System

LGEIF Launch Geocentric Equatorial Inertial Frame

LV Launch Vehicle

PZ90 Acronym of Russian Parametry Zemli 1990

SI International System of Units

TAI International Atomic Time

TCG Geocentric Coordinate Time

TDT Terrestrial Dynamical Time

THF Topocentric Horizon Frame

UT1 Universal Time

UTC Coordinated Universal Time

WGS84 World Geodetic System, 1984

2.2 Symbols

a semimajor axis

a_e earth semimajor axis of terrestrial ellipsoid IERS used in ITRS

b semiminor axis

E eccentric anomaly

e eccentricity

GM_e earth gravitational parameter used in ITRS

h_a apogee altitude

h_p perigee altitude

i inclination

M mean anomaly

n mean motion of satellite

p semilatus rectum

r_a apogee radius

r_p perigee radius

S_0 GAST at the time of payload – LV separation

T period

t_p time interval between the launch moment and the perigee passing

t_{SEP} time interval between the launch moment and the payload – LV separation

u argument of latitude

V velocity

V_x, V_y, V_z projection of velocity in LGEIF

x, y, z projection of position in LGEIF

α_e flattening of the earth

θ true anomaly

φ_{e0} geocentric latitude at launch point

λ_0 longitude at launch point

λ_N longitude of the ascending node in LGEIF

ω argument of perigee

ω_e angular velocity of the earth

Ω right ascension of the ascending node

3 Orbit elements and calculation conditions

3.1 Orbit elements

Six independent orbit elements describe the orbit of a satellite. Two elements describe orbit size and shape, three elements describe orbit orientation, and one element describes orbit location.

Orbit size and shape parameters include the following:

a) semimajor axis;

b) eccentricity;

c) semiminor axis;

d) semilatus rectum;

e) perigee radius;

f) apogee radius;

g) perigee altitude;

h) apogee altitude;

i) period;

j) mean motion.

Orbit orientation parameters include the following:

a) inclination;

b) right ascension of the ascending node;

c) argument of perigee;

d) longitude of the ascending node.

Satellite location parameters include the following:

a) true anomaly;

b) eccentric anomaly;

c) mean anomaly;

d) time past perigee;

e) time past ascending node;

f) argument of latitude.

The orbit elements are shown in Figure 1.

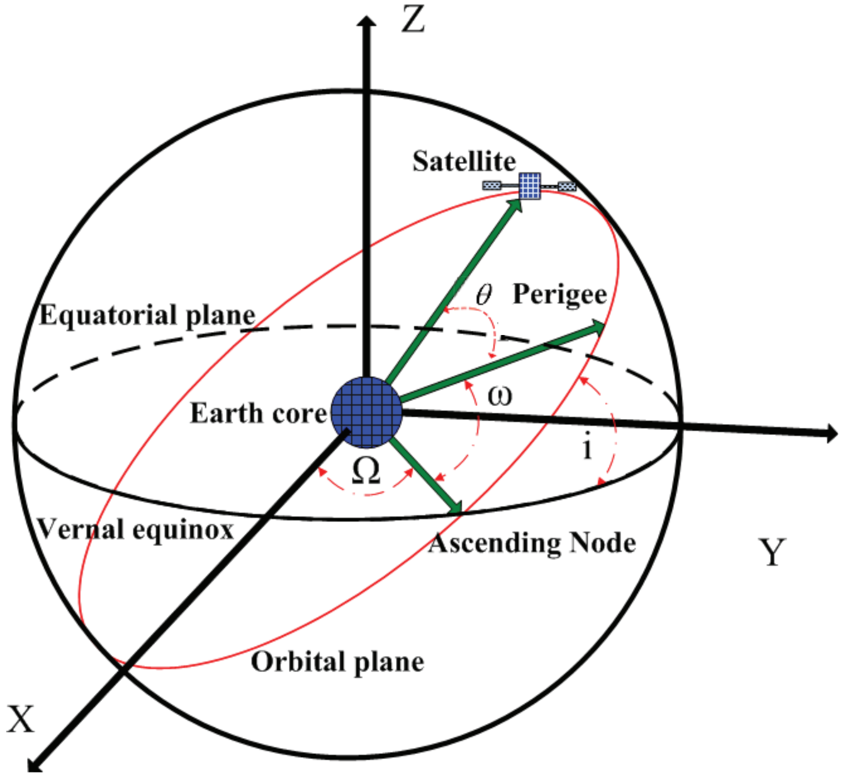

Figure 1 — Orbit elements

The orbit ellipse geometry is shown in Figure 2.

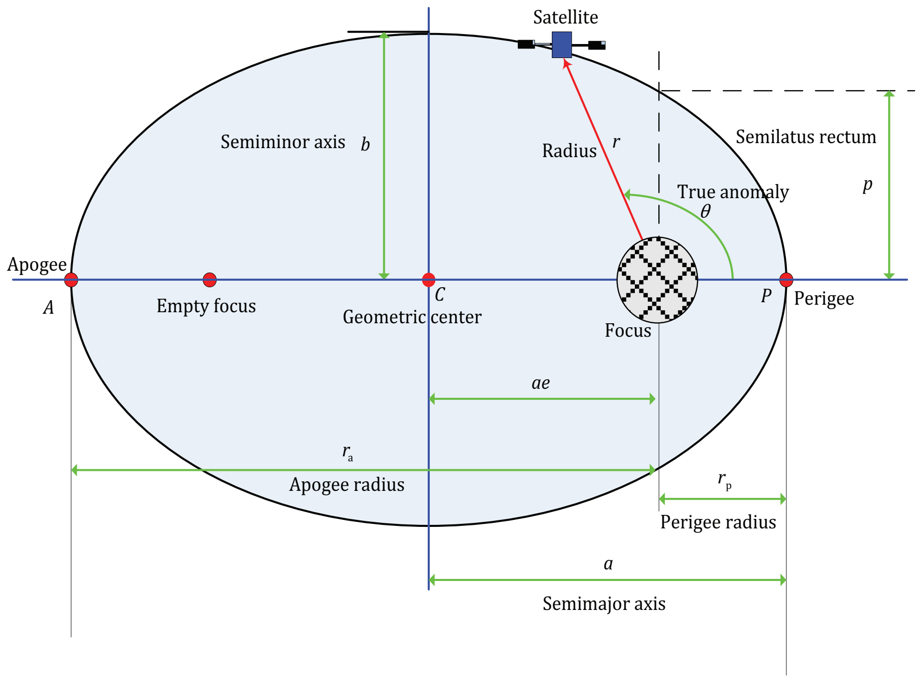

Figure 2 — Ellipse geometry

3.2 Data source

3.2.1 General

Calculation of orbit elements at separation uses velocity vector and position vector. Ground-based or space-based external measurements should be used.

3.2.2 Guideline of correction about the external measurements

a) Correction about lift-off time

Unified timer start point about different instruments is suggested to be specified. The timer start point needs to be corrected by lift-off time.

b) Correction about tracking point

Tracking point of different measurement instruments at flight region is suggested to be provided. The tracking point is usually transformed into LV navigation system coordinate frame. The data of tracking point correction is usually provided by system engineering department.

c) Correction about exception value

The measurement data need to pass a reasonable test and exception value needs to be deleted if necessary.

d) Correction about measurement data

The measurement data correction includes radio measurement and optical measurement. The measurement data based on ship is suggested to include ship drift position correction, ship deformation correction, and ship attitude correction.

e) Data format

Data format is based on decade float point and the bit numbers are determined by measurement accuracy and measurement mission.

3.2.3 External measurement data accuracy

a) Considering the need for flight test.

b) Considering the need for flight test result and injection accuracy.

c) Considering the need for external measurement accuracy.

d) Considering the accuracy, economy, and configuration about external measurement instrument in the flight region.

3.3 Coordinate systems and time systems

3.3.1 Coordinate systems

a) True Greenwich Frame

— The origin is located at the Earth's centre of mass (including oceans and atmosphere).

— The Z axis coincides with the instantaneous earth's axis of rotation and points northward.

— The x-axis is directed vertical to the Z axis and make X-Z plane coinciding with the plane of the true Greenwich meridian.

— The y-axis completes a right-handed system.

— The X-Y plane is the true earth's equatorial plane.

b) International Terrestrial Reference Frame (ITRF)

ITRF is a realization of the ITRS. The ITRS and ITRF solutions are maintained by IERS.

— The origin is located at the Earth's centre of mass (including oceans and atmosphere).

— As a time scale TCG (Geocentric Coordinate Time) is chosen, x-axis inherits the IRM (International Reference Meridian) plane.

— The Z axis coincides with CTP (Conventional Terrestrial Pole).

— The y-axis completes Cartesian system up to right-hand system.

The coordinate frame definition of WGS-84, PZ90, and ITRF is almost the same, but there is some parameter difference between them. The coordinate difference on the surface of earth between the latest WGS-84 and ITRF is only a few centimetres, and the difference between PZ90 and ITRF is meter class.

For the mission of low-level accuracy requirement, it is admissible to neglect the difference between the true Greenwich frame and ITRF. For the mission of high-level accuracy requirement, it is necessary to consider the difference.

c) Launch Geocentric Equatorial Inertial Frame (LGEIF)

LGEIF is a geocentric equatorial inertial reference frame, made up by the true Greenwich frame fixing at the launch moment. Position and velocity at the separation moment, resulted from LGEIF, are used for orbital elements calculation.

d) Geocentric Celestial Reference Frame (GCRF)

GCRF is an inertial reference frame.

— Origin is located at the Earth's centre of mass (including oceans and atmosphere).

— The x-axis is directed toward the mean vernal equinox.

— The Z axis directed toward the mean celestial pole perpendicular to the equatorial plane.

— The y-axis lies 90° ahead (Eastward) in the equatorial plane, thus completing the right-handed coordinate system.

J2000.0 is a kind of GCRF. Transformation between ITRF and GCRF can be found in 5.5.6 of Reference [7].

e) Topocentric Horizon Frame (THF)

THF is an earth-fixed reference frame.

— The origin is located at the observation point.

— The X-Y plane is the local horizon, which is the plane tangent to the ellipsoid at observation point.

— The X plane is directed eastward.

— The Z axis is normal to this plane directed outward towards the zenith.

— The y-axis points north, thus completing a right-handed system.

3.3.2 Time systems

The time systems may be used by coordinate transformation includes the following.

a) Greenwich Apparent Sidereal Time (GAST)

The origin of the GAST is the equinox, which has components of motion along the equator; these are due to the motion of the equator and ecliptic with respect to each other. Thus, the relationship between GAST and UT1 includes terms due to precession and nutation. The earth rotation angle and its relation to UT1 do not depend on combinations of precession and nutation.

b) Universal Time (UT1)

Universal Time (UT1) is the angular measure of Earth rotation inferred from observations of extra-galactic radio sources. The Earth-rotation angle provides a sequentially increasing continuum that is everlasting and widely apparent and serves as the astronomical basis of civil time of day. Earth rotation is only regular to about one part in 10 million per day. Being an observed quantity, it is measured and predicted by the IERS and distributed as a differential quantity relative to atomic UTC. Specifically, Universal Time indicates how the Earth's terrestrial reference frame is oriented relative to the celestial reference frame used by satellites.

c) Coordinated Universal Time (UTC)

Coordinated Universal Time (UTC) is a civil broadcast standard governed by ITU-R Recommendation 460, providing both astronomical time of day and atomic-time interval. UTC is atomic time kept within ±0,9 s of UT1 by the introduction of so-called leap seconds, and for this reason, is a legally recognized proxy for mean solar time in most countries. UTC is always offset from TAI by an integer number of seconds, and is thus a carrier of precision frequency and time interval for broadcast standards based on the SI second. Although atomic UTC is completely sequential and coherent (continuous) within the prescriptions of the UTC standard, the length of UTC day is non-uniform, owing to the possible addition or subtraction of an intercalary leap second at the end of the UTC month, usually June or December. DUT1 is a prediction of the difference between UT1 and UTC in tenths of a second and is available in advance. UTC is recommended for the sources of data in this Technical Report.

d) International Atomic Time (TAI)

International Atomic Time (TAI) is a physical time scale, which is affected by the Earth's gravitational and rotational potential (the geoid), and can be deduced from a weighted average of various international frequency standards. Relative weighting is based on the historical stability of the individual standards. TAI is maintained by the Bureau International des Poids et Mesures (BIPM) and is a reference basis of other time scales. Global Positioning System (GPS) Time is the reference time scale of the GPS navigation system; ideally, it is steered to lag TAI by 19 (19) s. TAI can be computed from UTC.

e) Terrestrial Dynamical Time (TDT)

Terrestrial Dynamical Time (TDT) is a theoretically ideal time at the Earth. A practical realization is TT = TAI + 32,184 s, which is the difference between this ideal time scale and the actual rotation of the Earth. TDT is used to compute the bias-precession-nutation matrix.

4 Calculation method of Keplerian elements

4.1 Calculation method of orbit elements

The semimajor axis, eccentricity, inclination, right ascension of the ascending node, argument of perigee, and true anomaly are usually considered as Keplerian elements. In orbit elements calculation, separation time, velocity vectors, and position vectors in LGEIF are used. They can be obtained from external measurements.

If the measurement data are not in LGEIF, a coordinate frame transformation is needed. Mostly possible coordinate frames are ITRF and THF. The transformation between ITRF and LGEIF considers the angle of Earth Rotation after take-off. The transformation between THF and ITRF considers topocentric geocentric latitude and topocentric longitude.

The Keplerian elements can be calculated as follows.

a) semimajor axis, *a*

$$r = \sqrt{x^2 + y^2 + z^2}$$

$$V = \sqrt{V_x^2 + V_y^2 + V_z^2}$$

$$\upsilon = \frac{rV^2}{GM_\mathrm{e}}$$

$$a = \frac{r}{2-\upsilon}$$

b) eccentricity, *e*

$$\gamma = \arcsin \frac{V_x x + V_y y + V_z z}{Vr}$$

$$e = \sqrt{1-\left(2-\upsilon\right)\upsilon \cos^2\gamma}$$

c) inclination, *i*

$$\begin{bmatrix} h_x \\ h_y \\ h_z \end{bmatrix} = \begin{bmatrix} yV_z - zV_y \\ zV_x - xV_z \\ xV_y - yV_x \end{bmatrix}$$

$$h = \sqrt{h_x^2 + h_y^2 + h_z^2}$$

$$i = \arccos \frac{h_z}{h}$$

d) right ascension of the ascending node, *Ω*

$$\sin /_0 = \frac{h_x}{\sqrt{h_x^2 + h_y^2}}$$

$$\cos /_0 = \frac{h_y}{\sqrt{h_x^2 + h_y^2}}$$

$$\Omega_0 \in \left[0,2\pi\right]$$

The quadrant of Ω_0 should be determined according to sign of sin Ω_0 and cos Ω_0.

$$\Omega = \Omega_0 - \omega_e t_\mathrm{SEP} + S_0, \Omega \in \left[0,2\pi\right]$$

S_0 is Greenwich Apparent Sidereal Time at the time of payload – LV separation. It can be replaced by GMST when the orbit elements are computed. The calculation method of GMST can be found in 4.4.2.4 of Reference [3].

e) true anomaly, θ

$$\theta = \begin{cases} \arccos \dfrac{a\left(1-e^2\right)-r}{er} & \left(\gamma \geq 0\right) \\ 2\pi - \arccos \dfrac{a\left(1-e^2\right)-r}{er} & \left(\gamma < 0\right) \end{cases}$$

f) argument of perigee, ω

$$u = \begin{cases} \arccos \dfrac{x\cos\Omega_0 + y\sin\Omega_0}{r} & \left(z \geq 0\right) \\ 2\pi - \arccos \dfrac{x\cos\Omega_0 + y\sin\Omega_0}{r} & \left(z < 0\right) \end{cases}$$

u is argument of latitude

$$\omega = u - \theta$$

4.2 Transformation of other orbit elements

4.2.1 Parameters of orbit size and shape

a) semiminor axis, b

$$b = a\sqrt{1-e^2}$$

b) semilatus rectum, p

$$p = a\left(1-e^2\right)$$

c) perigee radius, r_p

$$r_p = a\left(1-e\right)$$

d) apogee radius, r_a

$$r_a = a\left(1+e\right)$$

e) perigee altitude, h_p

$$h_p = r_p - a_e$$

f) apogee altitude, h_a

$$h_a = r_a - a_e$$

g) period, T

$$T = 2\neq\sqrt{\frac{a^3}{GM_e}}$$

h) mean motion, n

$$n = \sqrt{\frac{GM_e}{a^3}}$$

4.2.2 Parameters of orbit orientation

Longitude of the ascending node, λ_N

$$\lambda_N = \Omega - S_0$$

4.2.3 Parameters of satellite location

a) eccentric anomaly, E

$$E = 2\,\mathrm{arc\,tan}\left(\sqrt{\frac{1-e}{1+e}}\tan\frac{\theta}{2}\right)$$

$\frac{E}{2}$ and $\frac{\theta}{2}$ should be in the same quadrant.

b) mean anomaly, M

$$M = E - e\sin E$$

c) time interval between the launch moment and the perigee passing, t_p

$$t_p = t_{SEP} - \frac{E - e\sin E}{n}$$

d) argument of latitude, u

$$u = \omega + \theta$$

5 Calculation method of orbit elements error

Orbit elements error is an important criterion to assess whether commercial launch is successful. Requirements of orbit elements error vary with different payloads and orbits.

Orbit elements error can be calculated as follows:

$$\Delta a = a - \overline{a}$$

$$\Delta e = e - \overline{e}$$

$$\Delta i = i - \overline{i}$$

$$\Delta\Omega = \Omega - \overline{\Omega}$$

$$\Delta\omega = \omega - \overline{\omega}$$

$$\Delta\theta = \theta - \bar{\theta}$$

$\bar{a}, \bar{e}, \bar{i}, \bar{\Omega}, \bar{\omega}, \bar{\theta}$ are the expected values of orbit elements according to the requirements of launch task, or the values which are calculated using normal parameters $\bar{t}_{\mathrm{SEP}}, \bar{V}_{\mathrm{x}}, \bar{V}_{\mathrm{y}}, \bar{V}_{\mathrm{z}}, \bar{x}, \bar{y}, \bar{z}$ at payload-LV separation.

$a, e, i, \Omega, \omega, \theta$ are the values which will be calculated using measured parameters $t_{\mathrm{SEP}}, V_{\mathrm{x}}, V_{\mathrm{y}}, V_{\mathrm{z}}, x, y, z$ at payload-LV separation.

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