Earth Explorer CFI Software

CONVENTIONS DOCUMENT

Code:	EE-MA-DMS-GS-0001
Issue:	1.4
Date:	02/08/07

	Name	Function	Signature
Prepared by:	Mariano Sánchez-Nogales Fabrizio Pirondini José Antonio González Abeytua Juan José Borrego Bote	Project Engineer Project Engineer Project Manager Project Engineer	
Checked by:	Javier Babe	Quality A. Manager	
Approved by:	José Antonio González Abeytua	Project Manager	

DEIMOS Space S.L.
Ronda de Poniente, 19,
Edificio Fiteni VI, Portal 2, 2ª Planta,
Tres Cantos 28670 Madrid, SPAIN
Tel.: +34 91 806 34 50
Fax: +34 91 806 34 51
E-mail: deimos@deimos-space.com

Document Information

Contract Data		Classification	
Contract Number:	15583/01/NL/GS	Internal	
		Public	
Contract Issuer:	ESA / ESTEC	Industry	X
		Confidential	

External Distribution		
Name	Organisation	Copies

Electronic handling	
Word Processor:	Adobe Framemaker 6.0
Archive Code:	P/MCD/DMS/01/026-003
Electronic file name:	ee-ma-dms-gs-0001-12

Document Status Log

Issue	Change Description	Date	Approval
1.0	New document based on Envisat Mission CFI Software Mission Convention Document	08/11/01	
1.1	Minor changes on ASCII time formats (Table 2)	04/02/02	
1.2	Minor changes on ASCII time formats (Table 2) Added CryoSat SIRAL extra counter description (section 4.4.3) Corrected an error in Mean Local Solar Time drift formula (pag.33)	15/04/02	
1.3	Added GOCE values in ANNEX A. See change bars for other minor changes	15/07/03	
1.4	New Reference frame definitions New Attitude reference frame definitions Updates for new missions.	02/08/07	

Table of Contents

List of Tables

List of Figures

1 SCOPE

This document describes in detail the time references and formats, reference frames, parameters, models, and units that will be used by the Earth Explorer Mission CFI Software. The description sometimes goes beyond the CFI-needed information, when deemed necessary for the sake of a correct explanation.

All topics treated along the document are applicable to the following CFI libraries:

- EXPLORER_DATA_HANDLING
- EXPLORER_LIB
- EXPLORER_ORBIT
- EXPLORER_POINTING
- EXPLORER_VISIBILITY
- EXPLORER_GEO_CORRECTIONS
- EXPLORER_RETRACKER

The present document covers the different satellite missions considered in the frame of the Earth Explorer Mission CFI Software, including ERS and Envisat missions (formally not part of Earth Explorer Missions).

The main body of the document covers all features present in the different missions, while the annex shows those topics that are applicable to each mission.

2 ACRONYMS

ADM/Aeolus	Atmospheric Dynamics Mission
ANX	Ascending Node Crossing
AOCS	Attitude and Orbit Control Sub-system
CDMU	Command and Data management unit
CFI	Customer Furnished Item
DRS	Data Relay Satellite
DORIS	Doppler Orbitography and Radio positioning Integrated by Satellite
ERS	European Remote Sensing Satellite
ESA	European Space Agency
ESO	European Southern Observatory
ESTEC	European Space Technology and Research Centre
ET	Ephemeris Time
FK4	Fourth Fundamental Catalogue
FK5	Fifth Fundamental Catalogue
FOS	Flight Operations Segment
GOCE	Gravity Field and Steady-state Ocean Circulation Mission
GPS	Global Positioning System
IAG	International Association of Geodesy
IAU	International Astronomical Union
IERS	International Earth Rotation Service
IRM	IERS Reference Meridian
IRP	IERS Reference Pole
ITRF	IERS Terrestrial Reference Frame
JD	Julian Day
LOS	Line of Sight
LNP	Local Normal Pointing
METOP	Meteorological Operational Polar Satellite
MLST	Mean Local Solar Time
MJD2000	Modified Julian Day of 2000
NEOS	National Earth Observation Service
OBT	On-Board Time
PDS	Payload Data Segment
SBT	Satellite Binary Time
SIRAL	Synthetic-Aperture Interferometric Radar Altimeter

SMOS	Soil Moisture and Ocean Salinity Mission
SR	Satellite Reference
SRR	Satellite Relative Reference
SRAR	Satellite Relative Actual Reference
S/C	Spacecraft
SI	International System of Units
SSP	Sub-Satellite Point
TAI	International Atomic Time
TLST	True Local Solar Time
UT1	Universal Time UT1
UTC	Coordinated Universal Time
YSM	Yaw Steering Mode

3 APPLICABLE AND REFERENCE DOCUMENTS

3.1 Applicable Documents

3.2 Reference Documents

MCD	Envisat-1 Mission CFI Software. Mission Conventions Document. PO-IS-ESA-GS-0561. Issue 2.0. 07/01/97.
CRYO_SRD	CryoSat System Requirements Document. CS-RS-ESA-SY-0006. Issue 6.
BOWRING	Method of Bowring. NGT Geodesia 93-7. P 333-335. 1993.
FLANDERN	Low-precision formulae for planetary positions. Astrophysical Journal Supplement Series: 41. P 391-411. T.C.Van Flandern, K.F. Pulkkinen. November 1979.
LIU_ALFORD	Semianalytic Theory for a Close-Earth Artificial Satellite. Journals of Guidance and Control Vol. 3, No 4. J.J.F. Liu and R.L. Alford. July-August 1980.
KLINKRAD	Semi-Analytical Theory for Precise Single Orbit Predictions of ERS-1. ER-RP-ESA-SY-004. H.K. Klinkrad (ESA/ESTEC/WMM). Issue 1.0. 28/06/87.
DRSENV_ICD	ICD between the DRS and the Envisat-1 System. CD/1945/mad. D/TEL/R. K. Falbe-Hansen. Issue 5. April 1996.
WGS84	World Geodetic System 1984. DMA-TR-8350.2 The Defence Mapping Agency. Second Edition. 01/09/91.
HEISKANEN	Physical Geodesy. Weikko A. Heiskanen, Helmut Moritz. Graz 1987.
STD76	U.S. Standard Atmosphere 1976. National Oceanic and Atmosphere Administration.
OAD_TIME	OAD Standards: Time and Coordinate Systems for ESOC Flight Dynamics Operations. Orbit Attitude Division, ESOC. Issue 1. May 1994.
ALMAN95	The Astronomical Almanac for the year 1995.
IERS_SUPL	Explanatory Supplement to IERS Bulletins A and B. International Earth Rotation Service (IERS). March 1995.
GREEN	Spherical Astronomy. Green, R.M. 1985
ALMAN05	The Astronomical Almanac for the year 2005.

4 TIME REFERENCES AND MODELS

4.1 Time References

The time references which may be used in the context of the Earth Explorer missions are listed in table 1:

Table 1: Earth Explorer time reference definitions

Time reference	Usage
Universal Time (UT1)	Used as time reference for all orbit state vectors.
Universal Time Coordinated (UTC)	Used as time reference for all products datation.
International Atomic Time (TAI)	Found in DORIS products.
GPS Time	Used by GOCE and Aeolus/ADM missions.

The relationships between UT1, UTC and TAI are illustrated in the following figure:

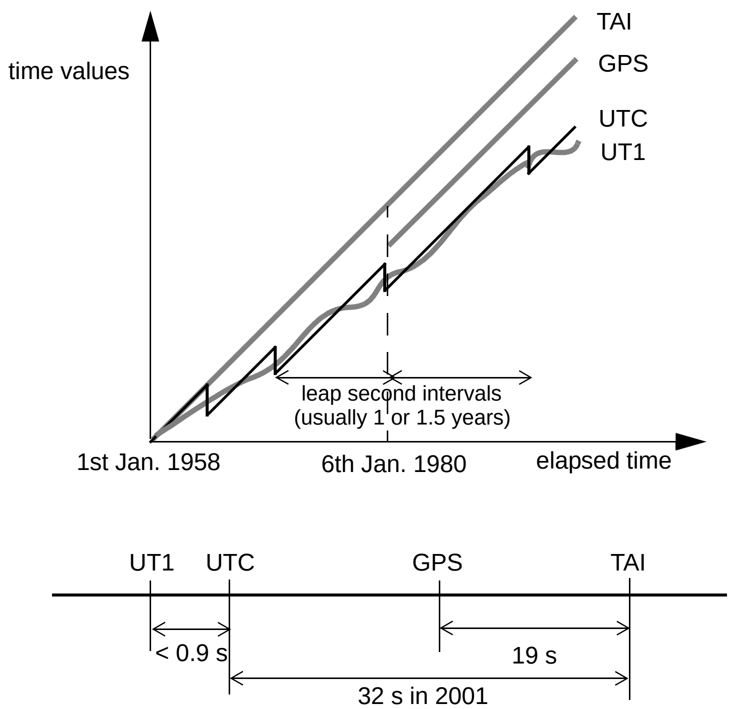


Figure 1: Relationships between UT1, UTC and TAI

Universal Time **(UT1)** is a time reference that conforms, within a close approximation, to the mean diurnal motion of the Earth. It is determined from observations of the diurnal motions of the stars, and then corrected for the shift in the longitude of the observing stations caused by the polar motion.

The time system generally used is the ***Coordinated Universal Time*** **(UTC)**, previously called *Greenwich Mean Time*. The UTC is piece wise uniform and continuous, i.e. the time difference between UTC and TAI is equal to an integer number of seconds and is constant except for occasional jumps from inserted integer *leap seconds*. The leap seconds are inserted to cause UTC to follow the rotation of the Earth, which is expressed by means of the non uniform time reference *Universal Time* UT1.

If UT1 is predicted to lag behind UTC by more than 0.9 seconds, a leap second is inserted. The message is distributed in a *Special Bulletin C* by the International Earth Rotation Service (IERS).

The insertion of leap seconds is scheduled to occur with first preference at July 1st and January 1st at 00:00:00 UTC, and with second preference at April 1st and October 1st at 00:00:00 UTC.

ΔUT1 = UT1 - UTC is the increment to be applied to UTC to give UT1, expressed with a precision of 0.1 seconds, and which is broadcasted, and any change announced in a *Bulletin D*, by the IERS[1].

DUT1 is the predicted value of ΔUT1. Predictions of UT1 - UTC daily up to ninety days, and at monthly intervals up to a year in advance, are included in a *Bulletin A* which is published weekly by the IERS.

International Atomic Time **(TAI)** represents the mean of readings of several atomic clocks, and its fundamental unit is exactly one SI second at mean sea level and is, therefore, constant and continuous.

ΔTAI = TAI - UTC is the increment to be applied to UTC to give TAI.

GPS Time is an atomic clock time similar to but not the same as UTC time. It is synchronised to UTC but the main difference relies in the fact that GPS time does not introduce any leap second. Thus, the introduction of UTC leap second causes the GPS time and UTC time to differ by a known integer number of cumulative leap seconds; i.e. the leap seconds that have been accumulated since GPS epoch in midnight January 5, 1980.

ΔGPS = TAI - GPS is the increment to be applied to GPS to give TAI, being a constant value of 19 seconds.

4.2 Time formats

The ***Julian Day*** **(JD)** is the interval of time in days and fraction of a day since 4713 BC January 1 at Greenwich noon (12:00:00).

The ***Modified Julian Day 2000*** **(MJD2000)** is the interval of time in days and fraction of day since 2000 January 1 at 00:00:00.

JD = MJD2000 + 2451544.5 [decimal days]

The time format year, month, day of month, hour, minute and second follows the Gregorian calendar.

4.2.1 Earth Explorer time formats

The time formats used with the time references proposed in section 4.1 can be one of the following:

- Processing
- Transport
- ASCII

1. ΔUT1 usually changes 1-2 ms per day

Table 2: Earth Explorers time formats

Time format		Description	Usage
Processing		64-bits floating point number, for decimal days	Internal processing, such as product processing sequences. Only for continuous times, i.e. TAI
Transport	Standard	Three 32-bits integer numbers for days, seconds and microseconds	Time values exchange between computers
	Envisat Ground Segment	Same as standard	
	CryoSat Ground segment	Same as standard	
	CryoSat General telemetry	Three 32-bits integer numbers for days, milliseconds and microseconds	
	CryoSat SIRAL telemetry	Four 32-bits integer numbers for days, milliseconds, microseconds and an extra counter of 80 MHz ticks	
	SMOS Transport format	Three 32-bits integer numbers for week number, seconds of the week and fraction of seconds	

Table 2: Earth Explorers time formats

Time format		Description	Usage
ASCII	Standard	Text string: “yyyy-mm-dd_hh:mm:ss“	Readable output, such as file headers, log messages, ...
	Standard with reference	Text string: “RRR=yyyy-mm-dd_hh:mm:ss“	
	Standard with microseconds	Text string: “yyyy-mm-dd_hh:mm:ss.uuuuuu“	
	Standard with reference and microseconds	Text string: “RRR=yyyy-mm-dd_hh:mm:ss.uuuuuu“	
	Compact	Text string: “yyyymmdd_hhmmss“	
	Compact with reference	Text string: “RRR=yyyymmdd_hhmmss“	
	Compact with microseconds	Text string: “yyyymmdd_hhmmssuuuuuu“	
	Compact with reference and microseconds	Text string: “RRR=yyyymmdd_ hhmmssuuuuuu“	
	Envisat	Text string: “dd-mmm-yyyy hh:mm:ss“	
	Envisat with reference	Text string: “RRR=dd-mmm-yyyy hh:mm:ss“	
	Envisat with microseconds	Text string: “dd-mmm-yyyy hh:mm:ss.uuuuuu“	
	Envisat with reference and microseconds	Text string: “RRR=dd-mmm-yyyy hh:mm:ss.uuuuuu“	
	CCSDS-A	Text string: “yyyy-mm-ddThh:mm:ss“	
	CCSDS-A with reference	Text string: “RRR=yyyy-mm-ddThh:mm:ss“	
	CCSDS-A with microseconds	Text string: “yyyy-mm-ddThh:mm:ss.uuuuuu“	
	CCSDS-A with reference and microseconds	Text string: “RRR=yyyy-mm-ddThh:mm:ss.uuuuuu“	

4.3 Time resolution

The time resolution is one microsecond.

4.4 Earth Explorer On-board times

The **On Board Time** is the time maintained by the CDMU and is the time reference for all spacecraft on-board activities. Depending upon the purpose and requirements of the mission, the time format used on-board the satellite will be drastically different. The following sections describe the format for the on-board time for different satellites.

4.4.1 Envisat On-board clock ticks

Table 3: On-board clock ticks

Time reference and format	Description	Usage
Satellite Binary Time (SBT)	32-bits integer number: · Count of 256 Hz clock ticks	Processing of satellite binary
On Board Time (OBT)	32-bits integer numbers: · obtm = most significant bits · obtl = least significant bits	Processing of instrument on-board time

The Satellite Binary Time (SBT) is a 32 bits counter, incremented by 1 at a frequency of about 256Hz (defined as the step-length PER_0). It varies from **00000000** (Hexadecimal) to **FFFFFFFF** (Hexadecimal), the next value being again **00000000** (Hexadecimal) and so on. This reset of the counter after **FFFFFFFF** (Hexadecimal) is called the **wrap-around**.

The On Board Time (OBT) is a generic term to represent any of the instrument counters, used to date their source packets. Most instruments use a 32 bits counter synchronized with the SBT. Some instruments use a 40 or 43 bits counter, where the 32 most significant bits are synchronized with the SBT (i.e. they use a more precise clock).

figure 2 shows the relationship between SBT and OBT.

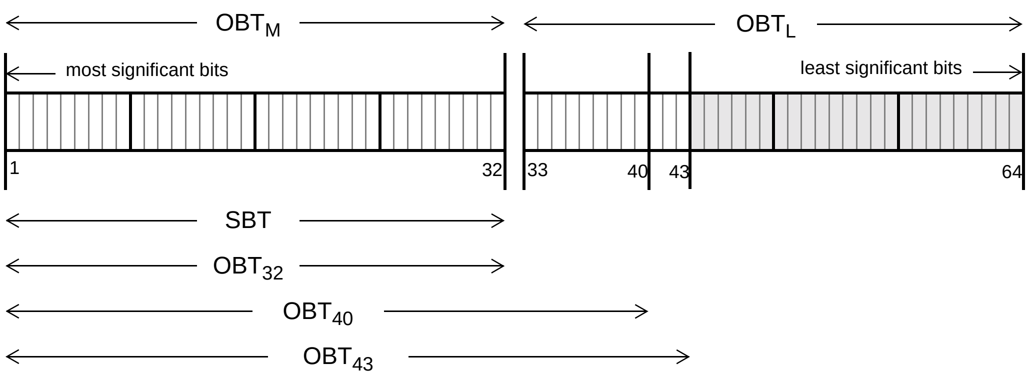


Figure 2: SBT and OBT relationship

4.4.2 TAI time

If DORIS is used to perform the orbit determination, the satellite will work with TAI time reference using dedicated transport formats (Telemetry formats).[2]

4.4.3 CryoSat SIRAL extra counter

The main payload of CryoSat is the Synthetic-Aperture Interferometric Radar Altimeter (SIRAL). The way the SIRAL instrument performs the on-board datation of each TM packet is the following:

Every time SIRAL receives the 1 Hz PPS signal (Pulse-Per-Second) from the central computer, it reads and sets in its memory the first 3 time parameters (days / milliseconds / microseconds). These won't change until the next PPS tick.

At the same time, it resets the fourth time parameter (extra counter) to 0, and starts counting ticks of the internal 80MHz clock in it. Each tick of the 80 MHz clock is 12.5 nanoseconds. The extra counter actually has a lower resolution, it actually counts a multiple (165) of the 80 MHz. This results in a counter resolution of 165 * 12.5 nanoseconds = 2.0625 microseconds.

From then on, at each TM packet production (which is about every 46 ms), SIRAL dates using the "frozen" first 3 parameters, plus the counter of 2 microsec ticks in the fourth parameter.

The actual date of the packet can be calculated by adding up all four parameters (with the appropriate scaling for each, of course), as for any other format.

At the next PPS, the same sequence starts.

It has to be remarked that these TM transport formats use vectors of long integers in the CFI (according to CFI standard). This, however, does not match the TM packet time contents, in which byte efficiency is important. For example, days are on 16-bits, milli-seconds on 32-bits, micro-seconds on 16-bits, and the extra counter on 16-bits.

This does not allow users to simply copy the sequence of bytes into memory and point the time vector to it, they will have to read each time component and set it into a long integer (and vice-versa for users producing test data).

4.4.4 SMOS On-board time

SMOS will manage two time sources:

- OBET: This value is derived by a HW counter 48 bits wide which is increased at a frequency of 65536 KHz, starting in 0 after power-on.
- UTC provided by Proteus each second.

The following tables shows the format for the OBET and the UTC-Proteus times:

Table 4: SMOS OBET time

P-Field				T-Field	
Extension Flag	Time Code identification	Details Bits for the information on the code		Coarse Time	Fine Time Note
1 bit	3 bits	4 bits		32 bits	24 bits
0	110	01	10	(Seconds from epoch)	(2^{-24} seconds)
1 byte				4 bytes	3 bytes

2. Not applicable to ENVISAT

.

Table 5: UTC Proteus time format

Week Number	Unused	H3	Second of Week	Fraction of Seconds
12 bits	3 bits	1 bit	32 bits	16 bits
	000		(Seconds from epoch)	(2^{-16} seconds)
2 bytes			4 bytes	2 bytes

Where “Week Number” is weeks elapsed since January 6-12, 1980. This week is numbered (0). LSB=1 Week.

H3 represents the time source from which the payload is synchronised to the platform

4.4.5 Aeolus On-board Time

The OBT format for Aeolus is given in CCSDS Unsegmented time code (CUC), that is defined as: the time from a defined epoch in seconds coded on 4 octets and sub-seconds coded on 2 octets.

According to this the time is:

$$Time = C_0 * 256^3 + C_1 * 256^2 + C_2 * 256 + C_3 + F_0 * 256^{-1} + F_1 * 256^{-2}.$$

OBT is set to GPS Time such that the UTC zero time-point reference of OBT is the same as that of GPS, i.e. midnight on the night of January 5 1980 / morning of January 6 1980. At this UTC zero time-point reference there had been19 leap seconds applied.

Therefore, the conversion from OBT (in CUC) to UTC is:

$$UTC = (CUCseconds + CUCsub\text{-}seconds * 256\text{-}2) - GPST + UTC_0$$

Where:

CUCseconds is the 4 most significant octets of OBT (C_0 to C_3)

CUCsub-seconds is the 2 least significant octets of OBT (F_0 to F_1)

GPST is the number of leap seconds between UTC and GPS Time (see section 3.2);

UTC_0: UTC time at 06-01-1980 00:00:00.000000

4.4.6 GOCE On-board Time

The OBT for GOCE is provided by telemetry as two parameters, the coarse OBT in 32 bits and the fine OBT in 16 bits. The OBT time is therefore OBT = (Coarse OBT) + (Fine OBT)/2^{16}.

The conversion form a given OBT to UTC is given by:

$$UTC0 = (Coarse\ UTC0) + (Fine\ UTC0)/2^{16}$$

$$OBT0 = (Coarse\ OBT0) + (Fine\ OBT0)/2^{16}$$

$$UTC = Gradient*(OBT-OBT0) + Offset + UTC0$$

The result is the number of seconds from 1st of Januay 2000 at 00:00:00.000000, without counting the leap seconds (i.e. to convert into a calendar date and time, one has to assume that all days have 86400 seconds).

5 REFERENCE FRAMES

The following reference frames are used in the context of Earth Explorer missions:

Table 6: Earth Explorer reference frames usage

Reference frame	CryoSat usage
Galactic	Star position and velocities can be given in this reference frame
Barycentric Mean of 1950	Some star catalogues use this reference frame to express the positions of their stars.
Barycentric Mean of 2000	The star catalogues usually use this reference frame to express the positions of their stars.
Heliocentric Mean of 2000	The ephemeris of the planets are usually expressed in this reference frame.
Geocentric Mean of 2000	The FOCC performs the internal calculations related to the predicted and restituted orbits in this reference frame.
Mean of Date	The Mean Local Solar Time is defined in this reference frame.
True of Date	It is the inertial reference frame used for input and output in the CFI software (e.g. star positions).
Earth fixed	It is the reference frame used for input and output of the satellite state vector (i.e. orbit definition), and for the output for geolocation in the CFI software.
Topocentric	It is the local horizontal reference frame used to define a looking direction.
Satellite Orbital	It is a reference frame centred in the satellite and defined by the satellite position and velocity. Its used as a reference for the application of the selected attitude control mode.
Satellite Nominal Attitude	It is used for the attitude determination. It is based on relation with the Satellite Orbital frame.
Satellite Attitude	It is used for the attitude determination as well. It is based on relation the Satellite Nominal Attitude frame or on measurements.
Instrument Attitude	It is the reference frame that constitutes the reference for the definition of a look direction relative to the satellite (e.g. to express the pointing of an instrument).

5.1 General Reference Frames

5.1.1 Galactic

The galactic plane is determined by the statistical study of the galactic dynamics. In this reference frame, position are determined by a galactic latitude and longitude. The galactic latitude are taken as the angle measured from the galactic plane, while the galactic longitude are measured from the direction of the galactic centre.

In order to relate the galactic coordinates of a star to its equatorial coordinates, it is necessary to know the position of the galactic pole and the position of the galactic centre. These points have been adopted as follow, for the epoch 1950.0:

Right ascension of the Galactic pole = 12h 49m.

Declination of the Galactic pole = 27º.4.

Galactic longitude of the north celestial pole = 123º (also known as the position angle of the galactic centre)

5.1.2 Barycentric Mean of 1950

It is based on the star catalogue FK4 for the epoch B1950, since the directions of its axes are defined relatively to a given number of that star catalogue positions and proper motions.

The centre of this reference frame is the barycentre of the Solar System. The x-y plane coincides with the predicted mean Earth equatorial plane at the epoch B1950, and the x-axis points towards the predicted mean vernal equinox. The latter is the intersection of the mean equator plane with the mean ecliptic, and the ecliptic is the orbit of the Earth around the Sun. The z-axis points towards north.

The word *mean* indicates that the relatively short periodic nutations of the Earth are smoothed out in the calculation of the mean equator and equinox.

5.1.3 Barycentric Mean of 2000

It is based, according to the recommendations of the International Astronomical Union (IAU), on the star catalogue FK5 for the epoch J2000.0, since the directions of its axes are defined relatively to a given number of that star catalogue positions and proper motions.

The accuracy of this reference system, realized through the FK5 star catalogue, is approximately 0.1’’.

The centre of this reference frame is the barycentre of the Solar System. The x-y plane coincides with the predicted mean Earth equatorial plane at the epoch J2000.0, and the x-axis points towards the predicted mean vernal equinox. The latter is the intersection of the mean equator plane with the mean ecliptic, and the ecliptic is the orbit of the Earth around the Sun. The z-axis points towards north.

The word *mean* indicates that the relatively short periodic nutations of the Earth are smoothed out in the calculation of the mean equator and equinox.

5.1.4 Heliocentric Mean of 2000

It is obtained by a parallel translation of the Barycentric Mean of 2000.0 reference frame from the barycenter of the Solar System to the centre of the Sun.

5.1.5 Geocentric Mean of 2000

It is obtained by a parallel translation of the Barycentric Mean of 2000.0 reference frame from the barycenter of the Solar System to the centre of the Earth.

5.1.6 Mean of Date

The centre of this reference frame is the centre of the Earth. The x-y plane and the x-axis are defined by the mean Earth equatorial plane and the mean vernal equinox of date. The expression *mean of date* means that the system of coordinate axes are rotated with the Earth´s precession from J2000.0 to the date used as epoch.The z-axis points towards north.

The precession of the Earth is the secular effect of the gravitational attraction from the Sun and the Moon on the equatorial bulge of the Earth.

5.1.7 True of Date

The centre of this reference frame is the centre of the Earth. The x-y plane and the x-axis are defined by the true Earth equatorial plane and the true vernal equinox of date. The expression *true of date* indicates the instantaneous Earth equatorial plane and vernal equinox. The transformation from the Mean of Date to the True of Date is the adopted model of the nutation of the Earth.

The nutation is the short periodic effect of the gravitational attraction of the Moon and, to a lesser extent, the planets on the Earth's equatorial bulge.

5.1.8 Earth Fixed

The Earth fixed reference frame in use is the *IERS Terrestrial Reference Frame* (ITRF).

The zero longitude or IERS Reference Meridian (IRM), as well as the IERS Reference Pole (IRP), are maintained by the International Earth Rotation Service (IERS), based on a large number of observing stations, and define the IERS Terrestrial Reference Frame (ITRF).

5.1.9 Topocentric

Its z-axis coincides with the normal vector to the Earth's Reference Ellipsoid, positive towards zenith. The x-y plane is the plane orthogonal to the z-axis, and the x-axis and y-axis point positive, respectively, towards east and north.

5.2 Satellite Reference Frames

Four levels of reference frames are used for attitude determination:

- The Satellite Orbital frame (SOF)
- Satellite Nominal reference frame (SNRF)
- Satellite reference frame (SRF)
- Instrument reference frame (IRF)

The SOF is used for the computation of the other satellite reference frames (see section 5.2.1 for the definition of this frame)

The SNRF is an ideal attitude model. The axis definition for this frame depends on the attitude model chosen for the satellite. Let's see two examples:

- Local Normal Pointing attitude (LNP), the z-axis is chosen in the direction of the satellite's zenith and the x-axis is defined in the direction of the satellite's inertial velocity vector (in True of Date).
- Yaw Steering Mode attitude (YSM): the z-axis is chosen in the direction of the satellite's zenith and the x-axis is defined in the direction of the satellite's velocity vector in the Earth Fixed CS.

A complete list of attitude models can be seen in section 8.1.

The SRF corresponds to the satellite actual (measured) attitude frame. It could be considered as the result of three consecutive rotations of the SNRF over three angles called ***mispointing angles***. The time derivative of those mispointing angles are called ***mispointing rates***.

Finally the IRF is a frame based on an instrument of the satellite. There exists one reference frame per instrument and it is used for location and looking direction from the instrument.

5.2.1 Satellite Orbital

It is a reference frame centred on the satellite and is defined by the Xs, Ys and Zs axes, which are specified relatively to the reference inertial reference frame, namely the True of Date.

The Zs axis points along the radial satellite direction vector, positive from the centre of the TOD reference frame towards the satellite, the Ys axis points along the transversal direction vector within the osculating orbital plane (i.e the plane defined by the position and velocity vectors of the satellite), orthogonal to the Zs axis and opposed to the direction of the orbital motion of the satellite. The Xs axis points towards the out-of-plane direction vector completing the right hand reference frame.

$$\bar{Z} = \frac{\bar{r}}{|\bar{r}|} \qquad \bar{X} = \frac{\bar{r} \wedge \bar{v}}{|\bar{r} \wedge \bar{v}|} \qquad \bar{Y} = \bar{Z} \wedge \bar{X}$$

where $\bar{X}$, $\bar{Y}$ and $\bar{Z}$ are the unitary direction vectors in the (Xs, Ys, Zs) axes, and $\bar{r}$ and $\bar{v}$ are the position and velocity vectors of the satellite expressed in the inertial reference frame.

Next drawing depicts the Satellite Orbital frame:

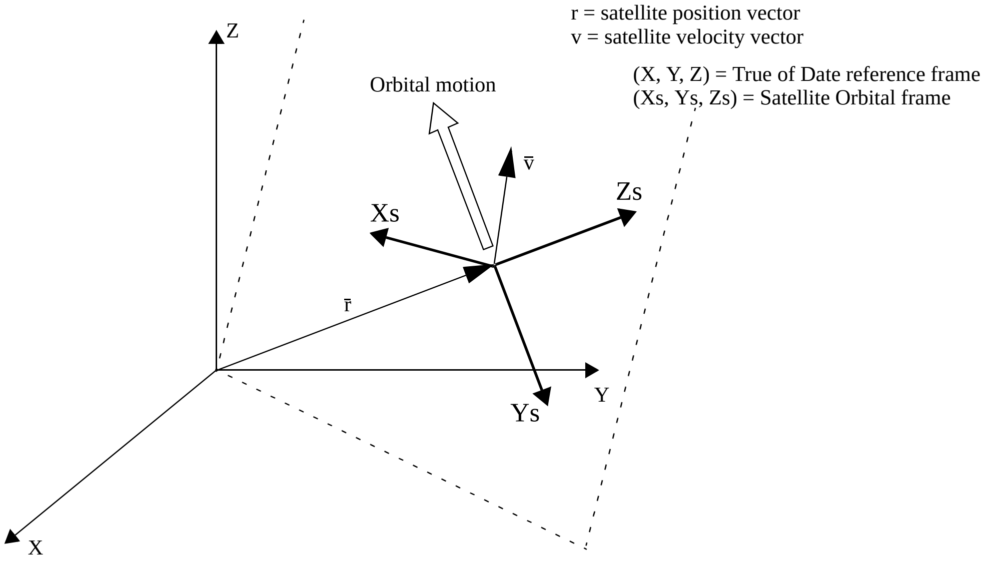
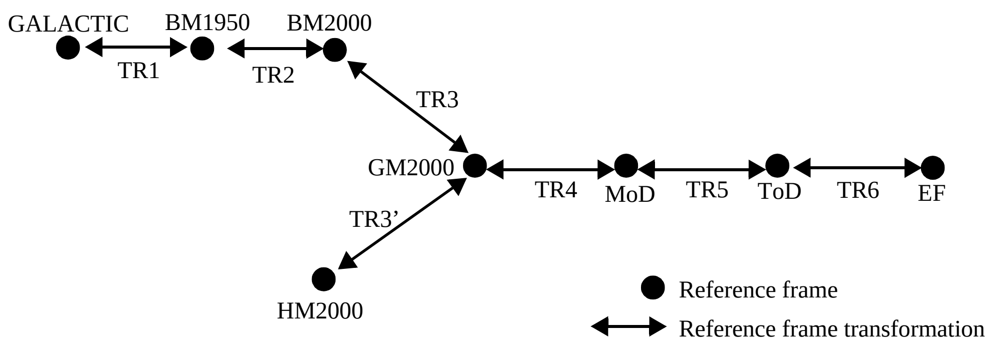


Figure 3: Satellite Orbital Frame

5.3 General Reference Frames Transformations

The following picture identifies the general reference frames transformations that are relevant for the Earth Explorer missions.



Reference frames:

GALACTIC = Galactic CS (see section 5.1.1)
BM1950 = Barycentric Mean of 1950.0 (see section 5.1.2)
BM2000 = Barycentric Mean of 2000.0 (see section 5.1.3)
HM2000 = Heliocentric Mean of 2000.0 (see section 5.1.4)
GM2000 = Geocentric Mean of 2000.0 (see section 5.1.5)
MoD = Mean of Date (see section 5.1.6)
ToD = True of Date (see section 5.1.7)
EF = Earth Fixed (see section 5.1.8)

Transfromations:

TR1 = Galactic to Barycentric Mean of 1950 (see section 5.3.1)
TR2 = Barycentric 1950 to Barycentric 2000 (see section 5.3.2)
TR3 = Solar system barycentre to Earth centre translation (see section 5.3.3)
TR3' = Sun centre to Earth centre translation (see section 5.3.4)
TR4 = Precession (see section 5.3.5)
TR5 = Nutation (see section 5.3.6)
TR6 = Earth rotation + nutation term + polar motion (see section 5.3.7)

Figure 4: General Reference Frames Transformations

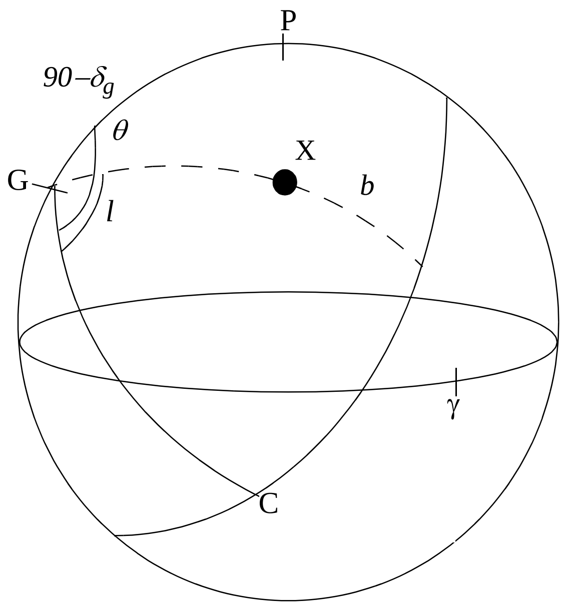
Those transformation are described in the following sections.

Note that whenever a transformation is expressed as a sequence of rotations, the following expressions apply (the angle *w* is regarded positive):

$$R_x(w) = \begin{bmatrix} 1 & 0 & 0 \\ 0 & \cos w & \sin w \\ 0 & -\sin w & \cos w \end{bmatrix} \qquad R_y(w) = \begin{bmatrix} \cos w & 0 & -\sin w \\ 0 & 1 & 0 \\ \sin w & 0 & \cos w \end{bmatrix} \qquad R_z(w) = \begin{bmatrix} \cos w & \sin w & 0 \\ -\sin w & \cos w & 0 \\ 0 & 0 & 1 \end{bmatrix}$$

5.3.1 Galactic to Barycentric Mean of 1950

The following picture represents the galactic and the equatorial coordinate systems. The relationship between both systems are given by the equatorial coordinates of the galactic pole for the epoch 1950 and for by the position of the galactic centre.



P North Pole (for epoch B1950)
G Galactic North Pole
C Galactic centre
δ_g Declination of the Galactic pole = 12^h49^m
α_g Right Ascension of the Galactic pole = 27.4º
θ Position of the galactic centre = 123º
l Galactic longitude of the point X
b Galactic latitude of the point X

Figure 5: Galactic and Equatorial coordinates

In the figure, considering the spheric triangle GPX, the relationship between the galactic and equatorial coordinates can be established (see GREEN for further details)

$$\cos b \sin(\theta\text{-}l) = \cos\delta \sin(\alpha - \alpha_g)$$

$$\cos b \cos(\theta\text{-}l) = \cos\delta_g \sin\delta - \sin\delta_g \cos\delta \cos(\alpha - \alpha_g)$$

Taking into account the relations between spherical and cartesian coordinates, it is easy to derive the rotation matrix from Galactic to Barycentric B1950.0:

$$R_{(\mathrm{galactic} \rightarrow \mathrm{B1950.0})} = \begin{bmatrix} -0.06698874 & 0.49272847 & -0.86760081 \\ -08727557659 & -0.45034696 & -0.1883746 \\ -04835389146 & 0.74458463 & 0.46019978 \end{bmatrix}$$

5.3.2 Barycentric Mean of 1950.0 to Barycentric Mean of 2000

The transformation from barycentric B1950.0 to barycentric J2000 includes the following processes:

1. Removal of the terms of elliptic aberration.
2. Rotation to the dynamical equinox of B1950.0
3. Correcting the proper motions for the equinox motion and the change in the value of precession
4. Changing from tropical to Julian centuries for the time scale of proper motions
5. Updating of positions to the epoch of J2000
6. Precession of positions and proper motions from B1950.0 to J2000.

For further details about this transformation, refer to:
- ALMAN05 (B32)
- Astronomical and Astrophysical Journal 128, 263-267 (1983)

5.3.3 Barycentric Mean of 2000 to Geocentric Mean of 2000

The transformation from the Barycentric Mean of 2000 to the Geocentric Mean of 2000 reference frame is calculated with the following expressions (figure 6):

$$\vec{r}_E = \vec{r}_B - \vec{r}_{B,\,Earth}$$
$$\vec{v}_E = \vec{v}_B - \vec{v}_{B,\,Earth}$$

where $\vec{r}_E$ and $\vec{v}_E$ are the position and velocity vectors in the Geocentric Mean of 2000 reference frame, $\vec{r}_B$ and $\vec{v}_B$ are the position and velocity vectors in the Barycentric Mean of 2000 reference frame, and $\vec{r}_{B,\,Earth}$ and $\vec{v}_{B,\,Earth}$ are the position and velocity vectors of the Earth in the Barycentric Mean of 2000 reference frame.

$\vec{r}_{B,\,Earth}$ and $\vec{v}_{B,\,Earth}$ are calculated according to BOWRING reference.

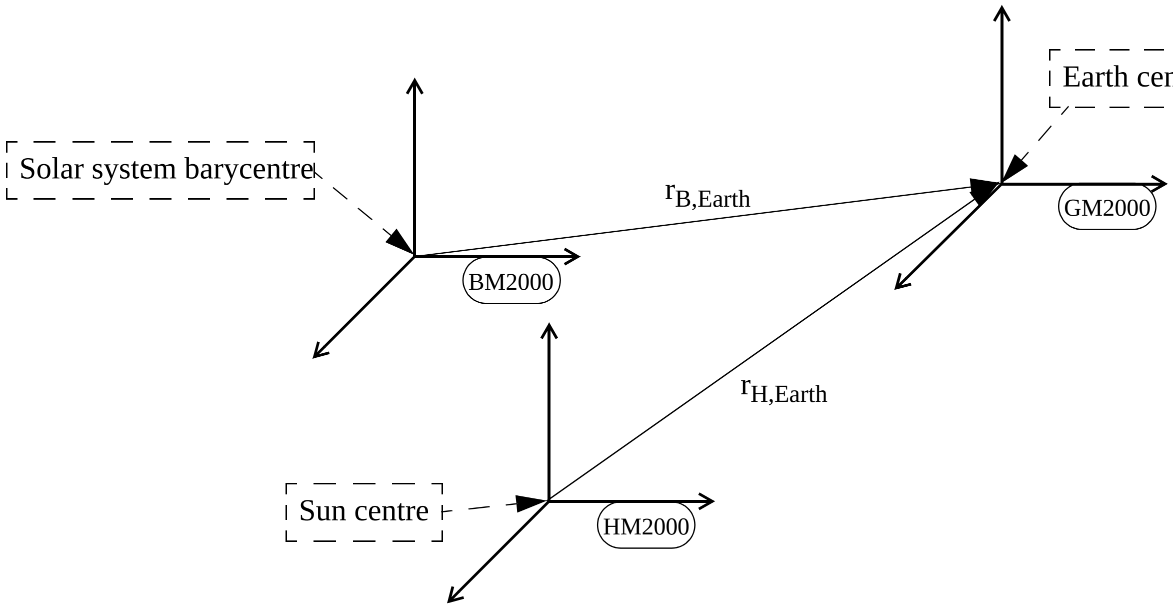


Figure 6: Transformations between BM2000, HM2000 and GM2000 reference frames

5.3.4 Heliocentric Mean of 2000 to Geocentric Mean of 2000

The transformation from the Heliocentric Mean of 2000 to the Geocentric Mean of 2000 reference frame is calculated with the following expressions (figure 6):

$$\bar{r}_E = \bar{r}_H - \bar{r}_{H,\,Earth}$$
$$\bar{v}_E = \bar{v}_H - \bar{v}_{H,\,Earth}$$

where $\bar{r}_E$ and $\bar{v}_E$ are the position and velocity vectors in the Geocentric Mean of 2000 reference frame, $\bar{r}_H$ and $\bar{v}_H$ are the position and velocity vectors in the Heliocentric Mean of 2000 reference frame, and $\bar{r}_{H,\,Earth}$ and $\bar{v}_{H,\,Earth}$ are the position and velocity vectors of the Earth in the Heliocentric Mean of 2000 reference frame.

$\bar{r}_{H,\,Earth}$ and $\bar{v}_{H,\,Earth}$ are calculated according to BOWRING reference.

5.3.5 Geocentric Mean of 2000 to Mean of Date

The transformation from the Geocentric Mean of 2000 to the Mean of Date reference frame is performed with the following expression (figure 7):

$$\bar{r}_m = R_z\left(-\frac{\pi}{2} - z\right)R_x(\theta)R_z\left(\frac{\pi}{2} - \zeta\right)\bar{r}_{J2000}$$

where $\bar{r}_m$ and $\bar{r}_{J2000}$ are the position vector in the Mean of Date and the Mean of 2000 reference frame, respectively.

The rotation angles of the precession model are calculated as follows (OAD_TIME reference):

$$\zeta = 0.6406161T + 0.0000839T^2 + 0.0000050T^3 \text{ [deg]}$$
$$z = 0.6406161T + 0.0003041T^2 + 0.0000051T^3 \text{ [deg]}$$
$$\theta = 0.5567530T - 0.0001185T^2 - 0.0000116T^3 \text{ [deg]}$$

where T is the TDB time expressed in the Julian centuries format (1 Julian century = 36525 days).

However, the precession motion is so slow that the UTC time can be used instead of the TDB time, and therefore T can be calculated from t, the UTC time expressed in the MJD2000 format, with the following expression:

$$T = (t - 0.5)/36525 \text{ [Julian centuries]}$$

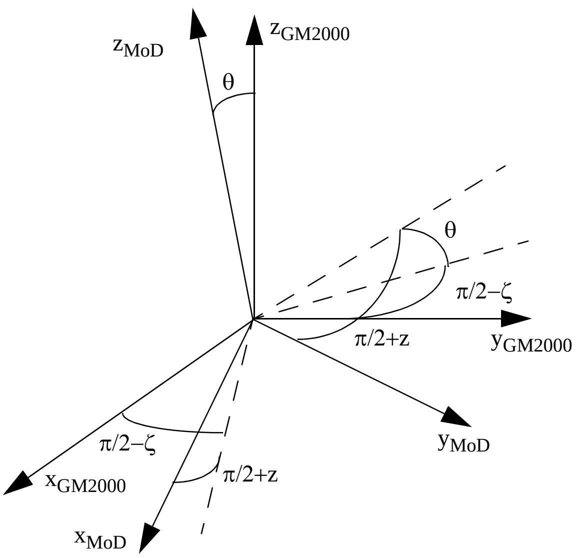


Figure 7: Transformation between GM200 and MoD reference frames

5.3.6 Mean of Date to True of Date

The transformation from the Mean of Date to the True of Date reference frame is performed with the following expression (figure 8):

$$\bar{r}_t = R_z(-\delta\mu)R_x(-\delta\varepsilon)R_y(\delta\upsilon)\bar{r}_m$$

where $\bar{r}_t$ and $\bar{r}_m$ are, respectively, the position vector in the True of Date and the Mean of Date reference frame.

The rotation angles of the simplified nutation model are calculated with (OAD_TIME reference):

$$\delta\mu = \delta\psi\cos\varepsilon$$

$$\delta\upsilon = \delta\psi\sin\varepsilon$$

where ε is the obliquity of the ecliptic at the epoch J2000:

$$\varepsilon = 23.439291 \text{ [deg]}$$

and $\delta\varepsilon$ and $\delta\psi$ is expressed by the *Wahr* model taking only the nine largest terms, and using UT1 instead of TDB as the time reference.

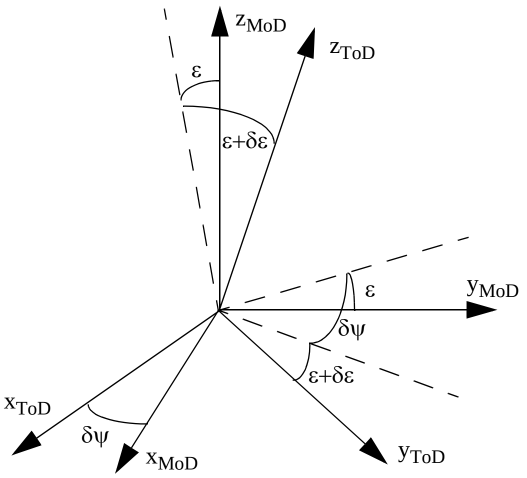


Figure 8: Transformation between MoD and ToD reference frames

5.3.7 True of Date to Earth Fixed

The transformation from the True of Date to the Earth fixed reference frame is performed with the following expression (figure 9):

$$\vec{r}_e = R_z(H)\vec{r}_t$$

where $\vec{r}_e$ and $\vec{r}_t$ are, respectively, the position vector in the Earth fixed and in the True of Date reference frames.

The ***Earth rotation angle H*** is the sum of the Greenwich sidereal angle and a small term from the nutation in the longitude of the equinox.

The Greenwich sidereal angle moves with the daily rotation of the Earth and is calculated with the Newcomb's formula according to international conventions as a third order polynomial, although the third order term will be neglected in our calculations.

The nutation term is calculated with the simplified nutation model (see section 5.1.7).

$$H = G + \delta\mu$$

$$G = 99.96779469 + 360.9856473662860T + 0.29079 \times 10^{-12}T^2 \text{ [deg]}$$

where T is the UT1 time expressed in the MJD2000 format.

Note that the transformation from the Mean of Date to the Earth fixed reference frame can be performed in one step being the $\delta\mu$ rotation term cancelled out:

$$\vec{r}_e = R_z(G)R_x(-\delta\varepsilon)R_y(\delta\upsilon)\vec{r}_{qm}$$

Finally, the polar motion parameters X and Y (measured and predicted by the IERS) are not taken into account in the True of Date to Earth-Fixed transformation.

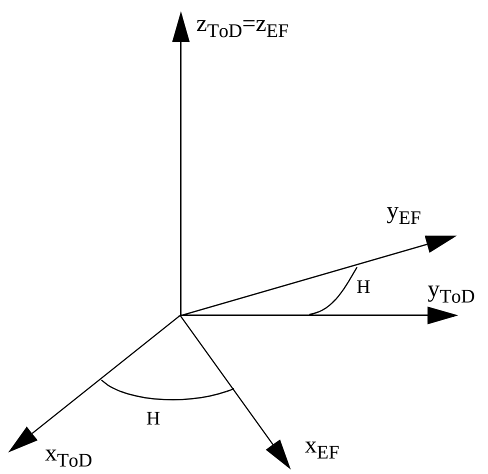


Figure 9: Transformation between ToD and EF reference frames

5.4 Satellite Reference Frames Transformations

There is not a general rule for transforming from one satellite reference frame to another. The attitude computation provides the transformation matrix from the satellite frame to an inertial reference frame. The following picture identifies the CFI-specific reference frames transformations that are relevant for the Earth Explorer missions:

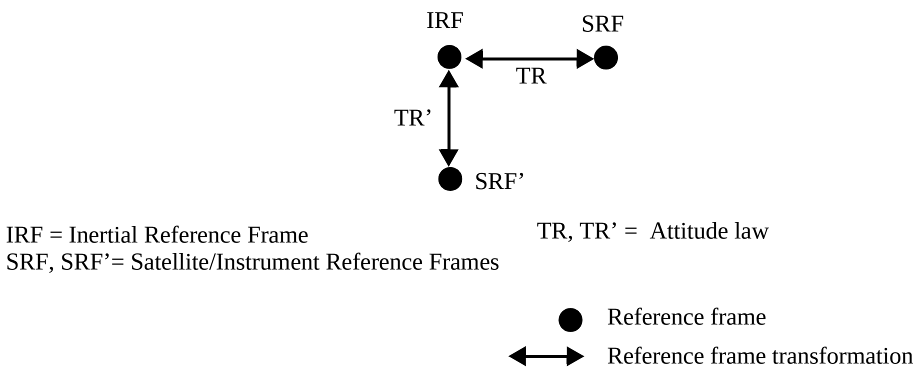


Figure 10: CFI-specific Reference Frames Transformations

6 ORBIT CHARACTERISATION

6.1 Orbit Definition

6.1.1 Sun-synchronous Orbit

The orbit is ***Sun-synchronous*** when the rate of change of the mean right ascension of the ascending node coincides with the motion of the mean Sun:

$$\dot{\Omega} = \dot{L}_{sun}$$

which implies that the MLST of the ascending node is also constant. Its behaviour is graphically presented in figure 11(a).

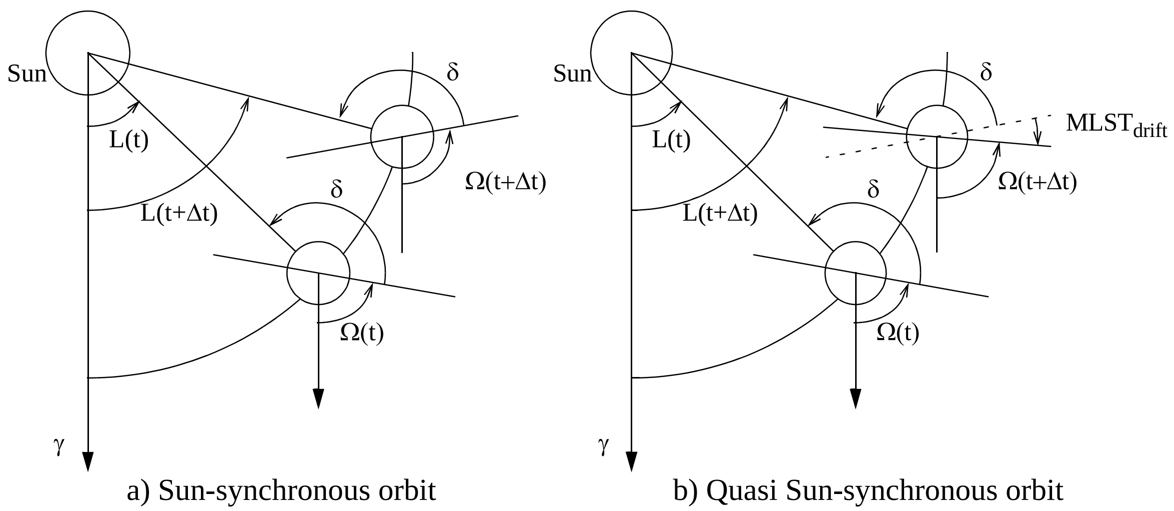


Figure 11: Sun-synchronous and quasi Sun-synchronous orbits descriptions

6.1.2 Quasi Sun-synchronous Orbit

The orbit is ***quasi Sun-synchronous*** when the rate of change of the mean right ascension of the ascending node is shifted from the motion of the mean Sun by a constant drift. It implies that the orbit line of nodes moves backward/forward with respect to the Sun-Earth LOS. The condition can be expressed mathematically in the following way:

$$\dot{\Omega} = \dot{L}_{sun} + MLS\dot{T}_{drift}$$

The behaviour of a quasi Sun-synchronous orbit compared to that of a Sun-synchronous orbit is presented in figure 11(b).

6.1.3 Geo-synchronous Orbit

The orbit is ***Geo-synchronous*** when the ground tracks repeats precisely after a constant number of integer days (*repeat cycle*) and a constant number of integer orbits (*cycle length*).

6.2 Orbit Types

6.2.1 Reference Orbit

The ***reference orbit*** consists of a scenario file, containing orbit information per repeat cycle change, i.e. the position and velocity vectors expressed in the Earth fixed reference frame, corresponding to the ascending node of that orbit and its associated time.

This state vector of the ascending node is calculated using the satellite-specific propagation mode, and imposing the conditions pertained the particular orbit definition.

6.2.2 Predicted Orbit

The ***predicted orbit*** consists of a single satellite cartesian state vector per orbit, i.e. the position and velocity vectors expressed in the Earth fixed reference frame, and the time corresponding to the ascending node crossing of that orbit, or its vicinity.

6.2.3 Restituted Orbit

The ***restituted orbit*** consists of a series of satellite cartesian state vectors computed at regular intervals (for instance each integer minute).

6.3 Orbit Propagation Definition

To calculate the state vector at any point in the orbit, it is sufficient to have a state vector at a given time, and then propagate that initial state vector to the required time using an orbit propagation model.

That initial state vector can come from different sources (see section 6.2) and depending on the type of orbit and the satellite mission, there are different requirements on the accuracy of the position and velocity vectors of that initial state.

6.4 Orbit Propagation Models

The propagation models must incorporate an initialisation mode. It basically starts with an initial cartesian state vector expressed in the Earth fixed reference frame at a given time, supplied externally (see section 6.2), to calculate the time and the state vector of the true ascending node in the Earth fixed reference frame (i.e. $z_{AN} = 0$ and $\dot{z}_{AN} > 0$).

The initialisation mode implements an iterative algorithm which is based upon a propagation mode.

6.4.1 Simulation mode

The simulation mode is one of reduced accuracy. In this case only the zonal (i.e. latitude independent) of the geoid J_2, J_2^2, J_3 and J_4 are used to calculate the secular perturbations of the *mean*[3] Kepler elements, and the zonal harmonic J2 is used to calculate the short periodic perturbations to transform the mean Kepler elements to the osculating Kepler elements.

3. Averaged with respect to the osculating mean anomaly over 2π.

This mode is based on the equations derived in LIU_ALFORD reference.

6.4.2 Operational mode

The operational mode is one the high accuracy. In this case, the effect of the latitude and longitude dependent geoid anomalies up to degree and order 36 (GEM-10B), as well as the effect of a medium air drag (MSIS'77) and luni-solar perturbations, have been modelled in the form of second order correction terms to the satellite position and velocity components (radial, along track, and across track).

These correction terms are function of the longitude of the true ascending node in the Earth fixed reference frame, and of the true latitude of the propagated state vector using the longitude independent mode, expressed in the True of Date reference frame.

This mode is based on the equations derived in KLINKRAD reference.

6.4.3 Non-Sun-synchronous Simulation mode

The simulation mode for non-Sun-synchronous orbits is identical to that for Sun-synchronous orbits.

6.4.4 Non-Sun-synchronous Operational mode

The operational mode for non-Sun-synchronous orbits is identical to that for Sun-synchronous orbits but not taking into account the luni-solar perturbations.

7 PARAMETERS

7.1 Orbit Parameters

7.1.1 Cartesian State Vector

It comprises the cartesian components of the position $\bar{r}_{SC}$, velocity $\bar{v}_{SC}$ and acceleration $\bar{a}_{SC}$ vectors of the satellite expressed in the Earth fixed reference frame at a given epoch.

7.1.2 Orbit Radius, Velocity Magnitude and Components

The satellite ***orbit radius*** is the module of the satellite position vector $\bar{r}_{SC}$:

$$R = |\bar{r}_{SC}|$$

The ***velocity magnitude*** is the module of the satellite velocity vector $\bar{v}_{SC}$:

$$V = |\bar{v}_{SC}|$$

The satellite ***velocity*** vector when is expressed in the True of Date reference frame can be split into two ***components***:

Radial component: $\bar{v}_r = \bar{v}_{SC} \bullet \bar{Z}$

Transversal component: $\bar{v}_t = \overline{-v_{SC}} \bullet \bar{Y}$

where $\bar{Y}$ and $\bar{Z}$ are the direction vectors of the Satellite Reference frame (see section 5.2.1).

7.1.3 Osculating Kepler State Vector

The ***osculating Kepler elements*** are related to the cartesian state vector, at the corresponding epoch, expressed in the True of Date reference frame.

The six Kepler elements are:

- Semi-major axis (a)
- Eccentricity (e)
- Inclination (i)
- Argument of perigee (ω)
- Mean anomaly (M)
- Right ascension of the ascending node (Ω)

Other auxiliary elements are:

- Eccentric anomaly (E)
- True anomaly (υ)
- True latitude (α)
- Mean latitude (β)

The relationships between these auxiliary elements and the six Kepler elements are:

$$\tan\frac{E}{2} = \sqrt{\frac{1-e}{1+e}}\tan\frac{\upsilon}{2}$$

$$M = E - e\sin E \text{ (Kepler's equation)}$$

$$\alpha = \omega + \nu$$

$$\beta = \omega + M$$

7.1.4 Mean Kepler State Vector

The osculating six Kepler elements in the True of Date reference frame can be averaged with respect to the mean anomaly over 2π, to obtain the mean Kepler elements:

$$\bar{a}, \bar{e}, \bar{i}, \bar{\omega}, \bar{\Omega}, \bar{M}$$

7.1.5 Equinoctial State Vector

The osculating Kepler elements are usually replaced by the equivalent osculating ***equinoctial elements*** for quasi-equatorial and quasi-circular orbits:

- $x_1 = a$
- $x_2 = e_x = e\cos(\Omega+\omega)$
- $x_3 = e_y = e\sin(\Omega+\omega)$
- $x_4 = i_x = +2\sin(i/2)\sin(\Omega)$
- $x_5 = i_y = -2\sin(i/2)\cos(\Omega)$
- $x_6 = \Omega + \omega + M$

7.1.6 Ascending Node, Ascending Node Time, Nodal Period, Absolute Orbit Number

The ***ascending node*** of an orbit is the intersection of that orbit, when the satellite goes from the southern to the northern hemisphere, with the x-y plane of the Earth fixed reference frame.

The ***ANX time*** is the UTC time of that ascending node.

The ***relative time*** with respect to the ANX time is the time elapsed since that ascending node till the current position within the orbit.

The ***nodal period*** of an orbit is the interval of time between two consecutive ascending nodes.

The Launch orbit from Kourou is regarded as ***absolute orbit number*** zero. From then on, each time a new ascending node is crossed the absolute orbit number is incremented by one.

7.1.7 Mean Local Solar Time Drift

The ***Mean Local Solar Time drift*** expressed the difference in angular velocity between the rate of change of the mean right ascension node and the motion of the mean Sun. This constant drift produces an increasing gap between the MLST of the ascending node and the angle measured from the line of nodes and the vernal equinox direction (see section 6.1.2). For a Sun-synchronous orbit, the MLST drift is zero.

The relationship between MLST of subsequent days is the following:

$$MLST_{dayN} = MLST_{day(N-1)} + MLST_{drift}$$

7.1.8 Repeat Cycle and Cycle Length

In the geo/helio-synchronous orbits, the ground track repeats precisely after a constant integer number of orbits and days. The number of days of that period is called the ***repeat cycle***, whereas the corresponding number of orbits is called the ***cycle length***.

The repeat cycle of a Sun-synchronous orbit is an integer number of days, while it is not an integer number when considering a non Sun-synchronous orbit. Thus, the orbit information contained within a scenario file comprises an integer repeat cycle plus a drift on it, to cope with non Sun-synchronous orbits. The true repeat cycle shall result from the following:

$$TrueRepeatCycle = RepeatCycle(1 + MLSTdrift)$$

7.1.9 Sub-satellite Point, Satellite Nadir and Ground Track

The ***subsatellite point (SSP)*** is the normal projection of the position of the satellite in the orbit on to the surface of the Earth´s Reference Ellipsoid. It is also referred as ***nadir***.

The trace made by the subsatellite point on the surface of the Earth's Reference Ellipsoid due to the motion of the satellite along its orbit is called the ***ground track***.

7.1.10 Mean Local Solar Time and True Local Solar Time

7.1.10.1 Mean Local Solar Time

The ***Mean Local Solar Time (MLST)*** is the difference between the right ascension of the selected point in the orbit RA and the *mean longitude of the Sun* L, expressed in hours.

$$MLST = (RA - L + \pi)\frac{24}{2\pi} \text{ [hours]}$$

The mean longitude $\bar{L}$ of the Sun represents the motion of the ***mean Sun*** and is given, in the Mean of Date reference frame, by (FLANDERN reference):

$$\bar{L} = 280.46592 + 0.9856473516(t - 0.5) \text{ [deg]}$$

where *t* is the UT1 time expressed in the MJD2000 format.

The motion of the mean Sun has a constant mean longitude rate, namely $\dot{\bar{L}} = 0.9856473516$ [deg/s].

7.1.10.2 True Solar Local Time

The ***True Local Solar Time (TLST)*** is the difference between the right ascension of the selected point in the orbit RA and the right ascension of the Sun RA_{Sun}, expressed in hours.

$$TLST = (RA - RA_{Sun} + \pi)\frac{24}{2\pi} \text{ [hours]}$$

The RA_{Sun} is calculated, in the Mean of Date reference frame, according to FLANDERN reference.

Mean and True Local Solar Time are normally expressed in hours considering the equivalence existing between hours and degrees; i.e. the Earth completes a complete revolution with respect to the Sun (360 degrees) in one day (24 hours).

7.1.11 Phase and Cycle

The phase is considered to be a portion of the mission characterised by a ground track pattern different from the previous and following. Each time a change of repeat cycle period is applied, a new phase starts. The decision of starting a new phase is performed by the mission management.

A cycle is defined as a full completion of the repeat period. A cycle starts by definition on an ascending node crossing closest to the Greenwich Meridian.

7.1.12 Absolute and Relative Orbit Number

The absolute orbit number considers the orbits elapsed since the first ascending node crossing after launch.

The relative orbit number is a count of orbits from 1 to the number of orbits contained in a repeat cycle. The relative orbit number 1 corresponds to the orbit whose ascending node crossing is closest to the Greenwich Meridian (eastwards). The relative orbit number is incremented in parallel to the absolute orbit number up to the cycle length, when it is reset and the cycle number is incremented by one.

When an orbit change is introduced, the relative orbit number of the new orbit is calculated such that the definition of the relative orbit number 1 is kept in the new repeat cycle.

7.1.13 Track Number

The track number is a count of orbits from 1 to the number of orbits contained in a repeat cycle. The track number 1 corresponds to the orbit whose ascending node crossing is closest to the Greenwich Meridian (eastwards). Two subsequent track numbers are those which have the nearest longitude of its ascending node crossing. Track number counter is incremented eastwards.

Track number 1 and relative orbit number 1 correspond to the same orbit. Furthermore, it exists a one-to-one relationship between track and relative orbit numbers within a repeat cycle.

7.2 Attitude Coordinate Systems Parameters

7.2.1 Attitude determination parameters

There are different ways for providing the attitude parameters in order to establish the transformations between the satellite reference frames:

1. **Attitude Mispointing Angles:**

   The transformation from one satellite reference frame to another is accomplished by three consecutive rotations over the angles pitch η, roll ξ and yaw ζ.

   The time derivative of those angles are the ***pitch, roll and yaw rates***.

   Both those angles and their rates are a function of the selected attitude control mode (see attitude control section particular to each satellite).

   Usually these angles are used for transforming from the satellite orbital frame to the satellite nominal attitude frame. Frequently there are superimposed on them a set of ***mispointing angles*** that make the Satellite Nominal Attitude Reference frame transform to the Satellite Attitude Reference frame.

The mispointing angles are expressed as three components, namely pitch $\Delta\eta$, roll $\Delta\xi$, and yaw $\Delta\zeta$.

The time derivative of those mispointing angles are the ***mispointing rates***.

The following picture shows the order in which a reference frame is transformed when aplying the three rotation angles:

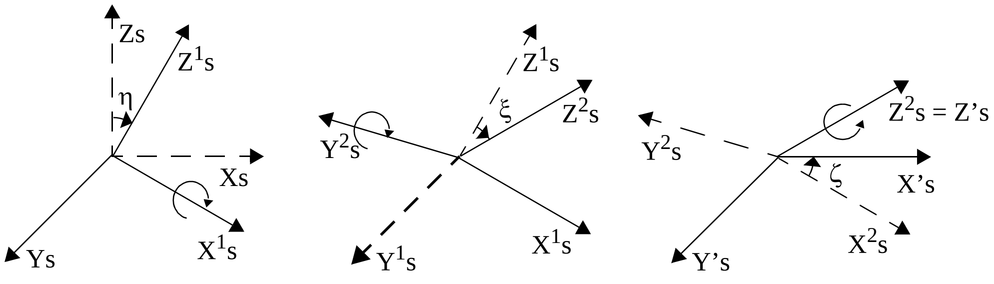


Figure 12: Rotation over the angles pitch η***, roll*** ξ ***and yaw*** ζ***.***

2. Attitude Quaternions:

The previous transformations could be given via quaternions (also called Euler symmetric parameters) instead of angles.

Quaternions are base on Euler's theorem that given two coordinate systems, there is one invariant axis (***e***) along which measurements are the same in both coordinate systems and that is possible to move from one system to the other through a rotation (β) about the axis ***e***. Acording to this theorem, the quaternions is defined as:

$$\mathbf{q} = \begin{bmatrix} q_1 \\ q_2 \\ q_3 \\ q_4 \end{bmatrix} = \begin{bmatrix} \mathbf{e}\sin\frac{\beta}{2} \\ \cos\frac{\beta}{2} \end{bmatrix} = \begin{bmatrix} e_x\sin\frac{\beta}{2} \\ e_y\sin\frac{\beta}{2} \\ e_z\sin\frac{\beta}{2} \\ \cos\frac{\beta}{2} \end{bmatrix}$$

A rotation matrix (direction cosine matrix) can be expressed in term of the quaternion parameters as follows:

$$R = \begin{bmatrix} (q_1^2 - q_2^2 - q_3^2 + q_4^2) & 2(q_1q_2 + q_3q_4) & 2(q_1q_3 - q_2q_4) \\ 2(q_1q_2 - q_3q_4) & (-q_1^2 + q_2^2 - q_3^2 + q_4^2) & 2(q_2q_3 + q_1q_4) \\ 2(q_1q_3 + q_2q_4) & 2(q_2q_3 - q_1q_4) & (-q_1^2 - q_2^2 + q_3^2 + q_4^2) \end{bmatrix}$$

There are for possible solutions for getting the quaternion from the rotation matrix:

$$\mathbf{Q}_1 = \begin{bmatrix} \frac{\sqrt{1+R_{11}-R_{22}-R_{33}}}{2} \\ \frac{1}{4Q_1}(R_{12}+R_{21}) \\ \frac{1}{4Q_1}(R_{13}+R_{31}) \\ \frac{1}{4Q_1}(R_{23}-R_{32}) \end{bmatrix} \quad \mathbf{Q}_2 = \begin{bmatrix} \frac{1}{4Q_2}(R_{12}+R_{21}) \\ \frac{\sqrt{1-R_{11}+R_{22}-R_{33}}}{2} \\ \frac{1}{4Q_2}(R_{23}+R_{32}) \\ \frac{1}{4Q_2}(R_{31}-R_{13}) \end{bmatrix} \quad \mathbf{Q}_3 = \begin{bmatrix} \frac{1}{4Q_3}(R_{13}+R_{31}) \\ \frac{1}{4Q_3}(R_{23}+R_{32}) \\ \frac{\sqrt{1-R_{11}+R_{22}-R_{33}}}{2} \\ \frac{1}{4Q_3}(R_{12}-R_{21}) \end{bmatrix} \quad \mathbf{Q}_4 = \begin{bmatrix} \frac{1}{4Q_4}(R_{23}-R_{32}) \\ \frac{1}{4Q_4}(R_{31}-R_{13}) \\ \frac{1}{4Q_4}(R_{12}-R_{21}) \\ \frac{\sqrt{1-R_{11}+R_{22}-R_{33}}}{2} \end{bmatrix}$$

The EXPCFI returns the weighted mean of the four possible solutions (with q_4 as real part of the quaternion):

$$\mathbf{q} = \begin{bmatrix} \mathbf{Q}_1^2 \\ \mathbf{Q}_2^2 \\ \mathbf{Q}_3^2 \\ \mathbf{Q}_4^2 \end{bmatrix} = \begin{bmatrix} \frac{1+R_{11}-R_{22}-R_{33}+R_{12}+R_{21}+R_{13}+R_{31}+R_{23}-R_{32}]}{4} \\ \frac{R_{12}+R_{21}+1-R_{11}+R_{22}-R_{33}+R_{23}+R_{32}+R_{31}-R_{13}}{4} \\ \frac{R_{13}+R_{31}+R_{23}+R_{32}+1-R_{11}+R_{22}-R_{33}+R_{12}-R_{21}}{4} \\ \frac{R_{23}-R_{32}+R_{31}-R_{13}+R_{12}-R_{21}+1-R_{11}+R_{22}-R_{33}}{4} \end{bmatrix}$$

3. AOCS Rotation Amplitudes

The AOCS rotation amplitudes are the three constants Cx, Cy and Cz that define the transformation from the Satellite Nominal Attitude to the Satellite Attitude Reference frame according to the selected attitude control mode (see attitude control section particular to each satellite).

7.2.2 Satellite Centered Direction

The parameters that define a direction in a Satellite Reference frame are the satellite related azimuth (Az) and the satellite related elevation (El):

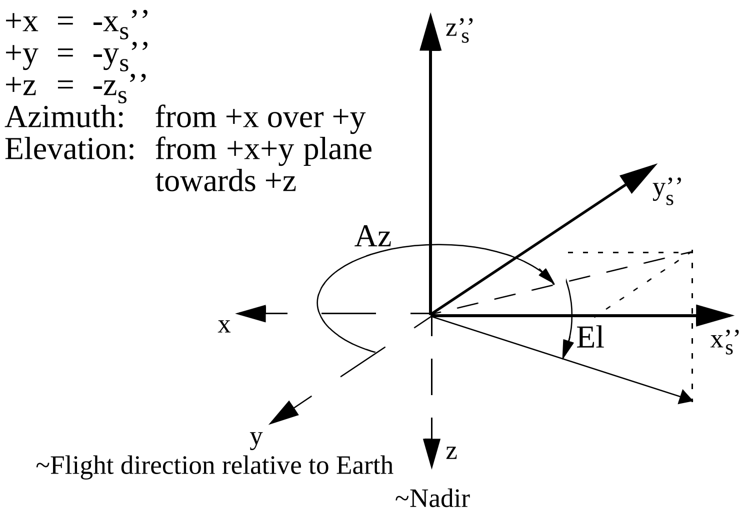


Figure 13: Satellite centred direction

7.3 Earth-related Parameters

Note that altitude refers always to geodetic altitude except when the contrary is explicitly said.

7.3.1 Geodetic Position

The geodetic coordinates of a point, related to the Earth´s Reference Ellipsoid, are the ***geocentric longitude*** λ, ***geodetic latitude*** φ, and ***geodetic altitude*** h, represented in figure 14.

The ***geocentric latitude*** φ’, ***geocentric radius*** ρ and the ***geocentric distance*** d are also represented in figure 14.

The parameters **a**, **e** and **f**, i.e. the semi-major axis, the first eccentricity and the flattening of the Earth’s Reference Ellipsoid (see section 8.3.2), define the equations that express these other parameters.

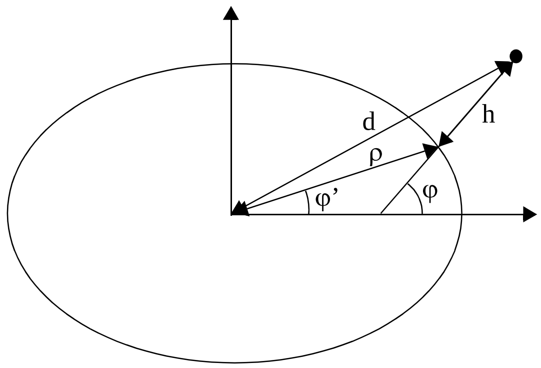


Figure 14: Geodetic position

The geocentric latitude φ’ and the geodetic latitude φ are related by the expression:

$$\tan\varphi = \frac{1}{(1-f)^2}\tan\varphi'$$

The geocentric radius ρ is calculated with:

$$\rho = \frac{a\sqrt{1-e^2}}{\sqrt{1-e^2\cos^2\varphi'}}$$

The relationship between the cartesian coordinates of a point and its geodetic coordinates is:

$$x = (N+h)\cos\varphi\cos\lambda$$
$$y = (N+h)\cos\varphi\sin\lambda$$
$$z = [(1-e^2)N+h]\sin\varphi$$

where N is the ***East-West radius of curvature***:

$$N = \frac{a}{\sqrt{1 - e^2 \sin^2 \varphi}}$$

The inverse transformation, from the cartesian to the geodetic coordinates, cannot be performed analytically. The iterative method that will be used will be initialized according to (BOWRING reference).

The normal projection of a point on the surface of the Earth's Reference Ellipsoid is called ***Nadir***, and when that point corresponds to the position of the satellite, the projection is called subsatellite point.

Another important radius of curvature is M, the ***North-South radius of curvature***:

$$M = \frac{a(1 - e^2)}{\sqrt{(1 - e^2 \sin^2 \varphi)^3}}$$

The ***radius of curvature*** in any selected direction R_{Az} can be calculated with the expression:

$$\frac{1}{R_{Az}} = \frac{\cos^2 Az}{M} + \frac{\sin^2 Az}{N}$$

where Az is the angle of the selected direction expressed in the Topocentric reference frame.

The ***satellite centred aspect angle*** $\alpha_{s/c}$ is the angle measured at the satellite between the *geometric direction*[4] from the satellite to the subsatellite point and the geometric direction from the satellite to the centre of the Earth.

The ***geocentric aspect angle*** α_g is the angle measured at the centre of the Earth between the geometric direction from the Earth centre to the subsatellite point and the geometric direction from the Earth centre to the satellite.

The ***subsatellite point centred aspect angle*** α_{ssp} is the angle measured at the subsatellite point between the geometric direction from the subsatellite point to the satellite and the geometric direction from the subsatellite point to the centre of the Earth.

The geodesic ***distance*** or ***ground range*** between two points that lay on an ellipsoid is by definition the minimum distance between those two points measured over that ellipsoid.

The ***velocity*** $\bar{v}_E$ and $\bar{a}_E$ ***acceleration relative to the Earth***, i.e the Earth's Reference Ellipsoid, of a point that lays on its surface can be split into different components.

- Northward component = $\bar{v}_E \bullet \bar{N}$ or $\bar{a}_E \bullet \bar{N}$
- Eastward component = $\bar{v}_E \bullet \bar{E}$ or $\bar{a}_E \bullet \bar{E}$
- Ground track tangential component = $\bar{v}_E \bullet \bar{t} = v_E$ or $\bar{a}_E \bullet \bar{t}$
- Magnitude = $v_E = |\bar{v}_E|$ or $a_E = |\bar{a}_E|$
- Azimuth = the azimuth of the $\bar{v}_E$ or $\bar{a}_E$ vectors measured in the Topocentric reference frame

where $\bar{N}$ and $\bar{E}$ are the north and east direction axes of the Topocentric reference frame centred on that point, and $\bar{t}$ is the unitary vector tangent to the ground track at that point.

4. The geometric direction is defined by the straight line that connects the initial and the final point.

7.3.2 Earth Centered Direction

The parameters that define a direction from the centre of the Earth to a point in the Mean of Date reference frame are the right ascension (α) and the declination (δ), shown in next figure:

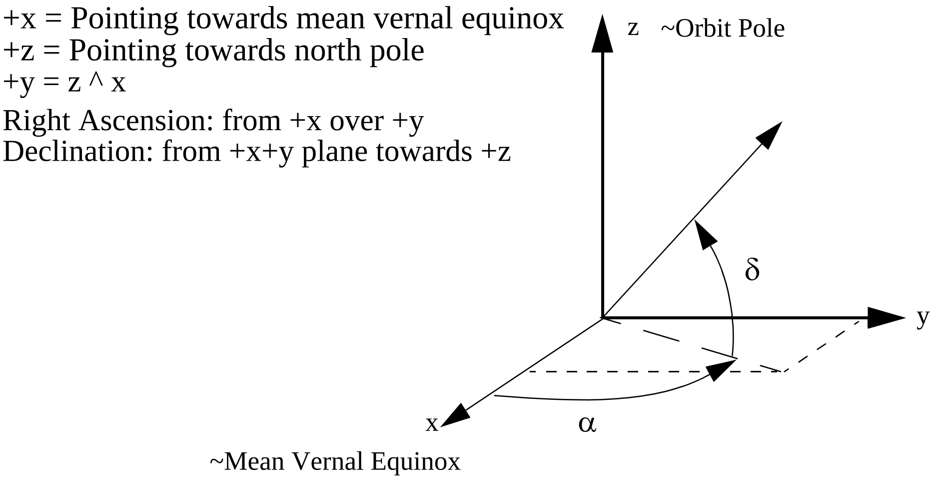


Figure 15: Earth centred direction

7.3.3 Topocentric Direction

The parameters that define a direction in the Topocentric reference frame are the topocentric azimuth (Az) and the topocentric elevation (El), represented in the next drawing:

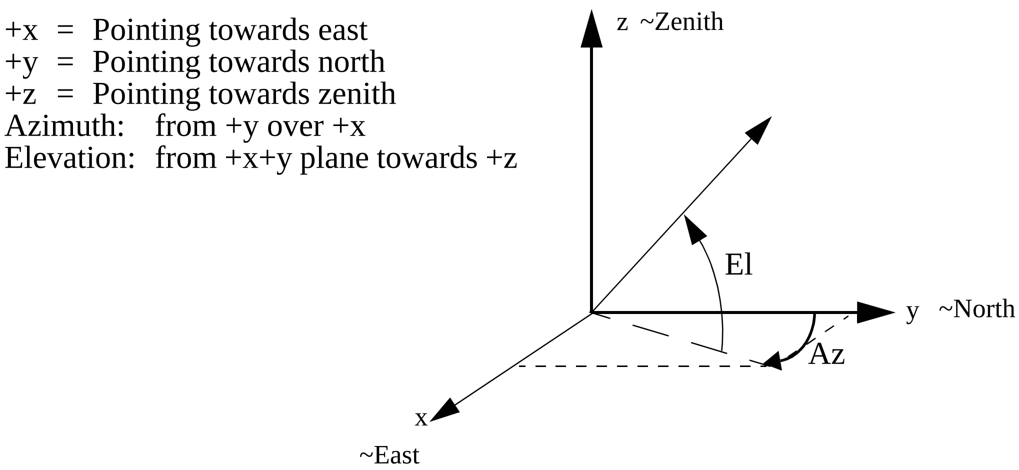


Figure 16: Topocentric direction

7.4 Ground Station Parameters

7.4.1 Ground Station Location

The location of a ***Ground Station*** is defined by its geodetic parameters: i.e. geocentric longitude λ, geodetic latitude φ, and geodetic altitude h with respect to the Earth's Reference Ellipsoid.

7.4.2 Ground Station Visibility

The visibility of a point from a Ground Station is limited by the ***minimum link elevation*** at which that point must be in order for the link between that Ground Station and that point to be established.

That minimum topocentric elevation is expressed in the Topocentric reference frame centred at that Ground Station (see section 7.3.3), and although it is ideally a constant, in fact a real Ground Station usually has a physical mask that makes the minimum topocentric elevation be a function of the topocentric azimuth.

7.5 Target Parameters

7.5.1 Moving and Earth-fixed Targets

A ***target*** $\bar{r}_t$ is a point that is observed from the satellite and that satisfies certain conditions.

The ***look direction***, or ***line of sight*** (LOS), $\bar{u}_0$ is the light direction, at the satellite, of the path followed by the light in its travel from the target to the satellite.

If the target moves with respect to the Earth, as a result of a change in the satellite position or a change in the look direction, it is called the ***moving target***.

If the target is fixed with respect to the Earth, which implies that if the satellite position changes then the look direction has to change in the precise way to keep looking to that particular point fixed to the Earth, it is called the ***Earth fixed target***.

In other words, the velocity of the moving target is the result of the motion of the satellite and the change in the look direction, or in the conditions that define it, with time. On the other hand, the velocity of the Earth fixed target is only a function of the position of that point with respect to the Earth's Reference Ellipsoid and the rotation of the Earth fixed reference frame.

7.5.2 Location Parameters

The location of a target is defined by its geodetic parameters: i.e. geocentric longitude λ, geodetic latitude φ, and geodetic altitude h with respect to the Earth's Reference Ellipsoid, although it also can be defined by its cartesian position vector (x, y, z) expressed in the Earth fixed reference frame.

7.6 Sun and Moon Parameters

The ***Sun semi-diameter*** D_{Sun} is the apparent semi-diameter of the Sun, expressed in radians, as seen from the satellite, and is calculated with the equation:

$$D_{Sun} = \frac{d_{Sun}}{R_{Sun-SC}}$$

where $d_{Sun} = 6.96 \times 10^8$ [m] is the semi-diameter of the Sun, and $R_{Sun\text{-}S/C}$ is the geometric distance between the satellite and the Sun centre.

The ***Moon semi-diameter*** D_{Moon} is the apparent semi-diameter of the Moon, expressed in degrees, as seen from the satellite, and is calculated with the equation:

$$D_{Moon} = \frac{d_{Moon}}{R_{Moon-SC}}$$

where d_{Moon} = 1738000 [m] is the semi-diameter of the Moon, and $R_{Moon-S/C}$ is the geometric distance between the satellite and the Moon centre.

The ***area of the Moon lit by the Sun*** $A_{Moon-Sun}$ is calculated with the expression:

$$A_{Moon-Sun} = \frac{1 + \cos\theta_{Sun-Moon-SC}}{2}$$

where $\theta_{Sun-Moon-S/C}$ is the angle measured at the centre of the Moon between the geometric direction from the centre of the Moon to the centre of the Sun and the geometric direction from the centre of the Moon to the satellite.

If $A_{Moon-Sun} = 0$ it is a new Moon, and if $A_{Moon-Sun} = 1$ it is a full Moon

The ***satellite eclipse flag*** indicates whether or not the path followed by the light from the centre of the Sun to the satellite intersects the Earth´s Reference Ellipsoid. It is equivalent to the ***satellite to Sun visibility flag***.

The ***satellite to Moon visibility flag*** indicates whether or not the path followed by the light from the centre of the Moon to the satellite intersects the Earth´s Reference Ellipsoid.

The ***target to Sun visibility flag*** indicates whether or not the path followed by the light from the centre of the Sun to the target intersects the Earth´s Reference Ellipsoid.

7.7 Euler angles

The Earth Explorer CFI applies the following convention when using Euler angles to rotate one reference frame to another.

The rotated reference frame (X’s, Y’s, Z’s) is obtained by applying three consecutive rotations to the original reference frame:

1. Rotation around -Ys over a roll angle η
2. Rotation around $-X^1s$ (i.e the rotated Xs) over a pitch angle ξ
3. Rotation around $+Z^2s$ (i.e the rotated Z^1s) over a yaw angle ζ.

Next drawing depicts the three rotations:

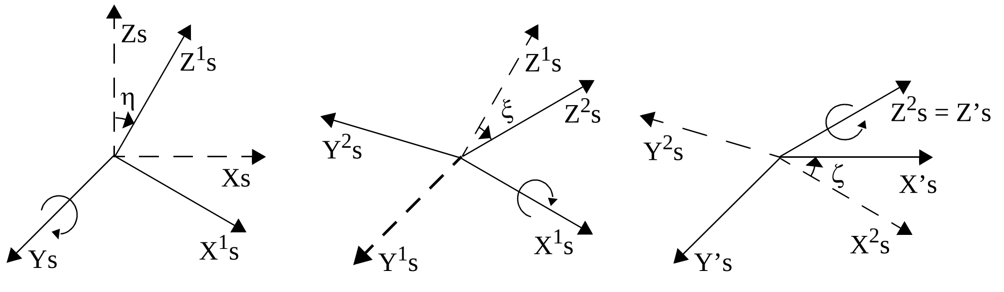


Figure 17: Euler Angles

8 MODELS

8.1 Attitude Control

The attitude control is based upon a set of attitude control modes, each one aiming a dedicated purpose and therefore following different attitude laws. table 7 presents different attitude control modes applicable to Earth observation missions.

Table 7: Attitude control modes

Mode	Purpose	Attitude
Rate Reduction Mode (RRM)	Used during initial acquisition, in re-acquisition case after failures, and also returning to nominal operations from safe mode. The RRM is intended to reduce the angular rates down to predefined values on the three axes	Arbitrary attitude.
Coarse Acquisition Mode (CAM)	Allows to acquire a geocentric pointing of the pitch and roll axis with a predefined accuracy, while maintaining a small angular rate on the yaw axis	Arbitrary to Geocentric, rotating
Coarse Pointing Mode (CPM)	The shall provide earth oriented 3-axis stabilisation	- The negative satellite z-axis (Yaw) parallel to the earth vector within 15 deg (1 sigma) - The satellite y-axis (Pitch) within 35 deg (1 sigma) to the orbital plane.
Fine Acquisition Mode 1 (FAM1)	Allows to acquire a pointing on the yaw axis lower than a certain level of accuracy, while maintaining the geocentric pointing on the pitch and roll axis	Geocentric, rotating to FAM2 attitude.
Fine Acquisition Mode 2 (FAM2)	Stable waiting mode, ending the acquisition phase; it maintains satisfactory pointing performances while minimizing the hydrazine consumption	The Satellite Relative Reference frame is fixed to the Local Orbital Reference Frame, rotated 180 deg around Z_S
Fine Acquisition Mode 3 (FAM3)	Transient mode between FAM2 and FPM.	Identical to FAM2 and FPM
Fine Pointing Mode (FPM)	Steady-state transition mode between YSM (not SYSM) and OCM. Triggered from the FAM2 through FAM3 to OCM or YSM	The Satellite Relative Reference frame is fixed to the Local Orbital Reference Frame. In some cases, the reference frame can be rotated 180 deg around Z_S

Table 7: Attitude control modes

Mode	Purpose	Attitude
Orbit Control Mode (OCM)	Third operational mode, used to perform out-of-plane orbit corrections (inclination updating) and large in-plane orbit correction (eccentricity and semi-major axis updating)	During thrust phase, depending on the type of manoeuvre: · *In-plane correction*: Identical to FPM attitude. · *Out-of-plane correction*: Derived from FPM attitude after a rotation of +/- 90 deg around Z_S axis
Stellar Yaw Steering Mode (SYSM)	This mode is activated either from YSM, or automatically after the SFCM. In this mode the satellite must ensure yaw steering pointing and local normal pointing with a pointing performance better than a certain level on each axis and a rate stability below an allowed threshold	The Satellite Relative Reference frame is fixed to the Local Relative Yaw Steering Reference Frame, rotated 180 deg around Z_S It moves wrt the Local Orbital Reference Frame according to the Local Normal Pointing and Yaw Steering laws
Yaw Steering Mode (YSM)	Stable transition mode before entering in Stellar Yaw Steering Mode (SYSM)	Identical to SYSM
Stellar Fine Control Mode (SFCM)	Operational mode dedicated to fine and short orbit corrections in the orbit plane in order to ensure a fine update of the semi-major axis and the eccentricity of the orbit	Identical to SYSM
Fine Control Mode (FCM)	Back-up mode of the Stellar Fine Control Mode (SFCM). Same purpose as SFCM	Identical to SYSM
Satellite Safe Mode (SSM)	Ensures a survival state after a major service module anomaly	Heliocentric pointing of the $+Z_S$ face for the satellite, plus a spin motion around about the Z_S axis
Local Normal Pointing (LNP)	In this mode the satellite must ensure local normal pointing with a pointing performance better than a certain level on each axis and a rate stability below an allowed threshold.	Yaw angle is set to zero. It moves wrt the Local Orbital Frame according to the Local Normal Pointing law

The three rotation angles (roll, pitch, yaw) that transform the Satellite Reference to the Satellite Relative Reference frame (see section section 5.2), are calculated as follows:

Table 8: Envisat-1 attitude control mode rotation angles

Mode	Roll	Pitch	Yaw
Rate Reduction Mode	Arbitrary attitude, not required to be simulated.		
Coarse Acquisition Mode	Arbitrary to Geocentric, rotating attitude, not required to be simulated.		
Fine Acquisition Mode 1	Geocentric, rotating to Orbital attitude, not required to be simulated.		
Fine Acquisition Modes 2 & 3 Fine Pointing Mode Orbit Control Mode[a]	0	0	0
[Stellar] Yaw Steering Mode [Stellar] Fine Control Mode	$C_Y \sin(U_{LAT})$	$C_X \sin(2U_{LAT})$	$C_Z \cos(U_{LAT})\left(1 - \frac{[C_Z \cos(U_{LAT})]^2}{3}\right)$
Local Normal Pointing	$C_Y \sin(U_{LAT})$	$C_X \sin(2U_{LAT})$	0
Satellite Save Mode	Heliocentric pointing, not required to be simulated.		

a. During the thrust phase of an out-of-plane correction the attitude mispointing should be increased / decreased by 90 degrees with respect to the nominal attitude mispointing.

where C_X, C_Y and C_Z are called the AOCS rotation amplitudes, in radians, and U_{LAT} is the satellite osculating true latitude in the True of Date reference frame.

8.2 DRS-Artemis Orbit

8.2.1 DRS-Artemis Orbit Definition

The initial DRS space segment comprises the Artemis Satellite located in the GEO orbit over Europe (16.4^o E). Artemis was launched the 12th of July 2001 reaching its operational orbit in xxxxx and is planned to be moved to 59^o E when the first DRSS is launched (DRSENV_ICD reference).

The orbit of the DRS is known on ground to an accuracy corresponding to the following errors ± 20.0 Km along track, ± 15.0 Km across track and ± 15.0 Km radial. These accuracies are achieved for a 24 hour prediction and are achieved when UT is the time reference (DRSENV_ICD reference)

The CFI software will check the compliance of the DRS orbit supplied on input with a set of requirements on the main osculating Kepler elements:

Table 9: DRS orbit tolerance requirements

Osculating Kepler element	Tight tolerance	Loose tolerance
Semi-major axis	42000 / 43000 Km	30000 / 50000 Km
Eccentricity	0.0 / 0.1	0.0 / 0.9
Inclination	- 0.1 / + 0.1 deg	- 1.0 / + 1.0 deg

If the tight tolerance requirements are not satisfied, but the loose ones are, then a warning will be returned by the CFI software.

If even the loose requirements are not satisfied, then an error will be returned.

8.2.2 DRS-Artemis Orbit Propagation Model

The 24-hour prediction of DRS will be available in equinoctial elements at a given epoch valid for certain validity period, and assuming that the user will propagate this state vector, within the validity period using the following algorithm:

$$a = a_{initial}$$
$$e_x = (e_x)_{initial}$$
$$e_y = (e_y)_{initial}$$
$$i_x = (i_x)_{initial}$$
$$i_y = (i_y)_{initial}$$
$$\lambda = \lambda_{initial} + (t - t_{initial}) \cdot d\lambda_{initial}dt$$
$$d\lambda_{initial}dt = (\mu / a^3)^{1/2}$$
$$\mu = 3,9860044 \ 10^5 \ km^3 / s^2$$

where $a_{initial}$, $(e_x)_{initial}$, $(e_y)_{initial}$, $(i_x)_{initial}$, $(i_y)_{initial}$ and $\lambda_{initial}$ are the equinoctial elements at $t_{initial}$.

8.3 Earth

8.3.1 Earth Position

The position and velocity of the Earth in the Barycentric and Heliocentric Mean of 2000 reference frames will be calculated according to FLANDERN reference.

8.3.2 Earth Geometry

The geometry of the Earth is modelled by a Reference Ellipsoid. Different definitions of reference ellipsoids can be found hereafter in table 10.

Table 10: WGS84 parameters

Parameter	Notation	WGS 84
Semi major axis (m)	a	6378137
Flattening = (a-b)/a	f	1/298.257223563
First Eccentricity = $(a^2 - b^2)/a^2$	e	0.0818191908426
Semi minor axis (m)	b	6356752.3142

The minimum distance between two points located on an ellipsoid is the length of the geodesic that crosses those two points. This geodesic distance will be calculated according to HEISKANEN reference.

The surface at a certain ***geodetic altitude h*** over the Earth's Reference Ellipsoid is defined by:

$$x = (N+h)\cos\varphi\cos\lambda$$
$$y = (N+h)\cos\varphi\sin\lambda$$
$$z = [(1-e^2)N+h]\sin\varphi$$

where N is the radius of curvature parallel to the meridian:

$$N = \frac{a}{\sqrt{1-e^2\sin^2\varphi}}$$

and φ and λ are the geodetic latitude and geocentric longitude of a point on that ellipsoid.

Nevertheless, the surface at a certain geodetic altitude h over the Earth´s Reference Ellipsoid will be modelled as another ellipsoid, *concentric* with it, and with (a+h) and (b+h) as semi-major and semi-minor axis.

This simplification is quite accurate and have the advantage that allows the analytical calculation of the intersection or tangent points with such a surface.

8.3.3 Earth Atmosphere

The Earth atmosphere can be represented by different models. The selection of a certain atmosphere model depends upon the requirements imposed by the mission definition. It could include certain simplifications to the generic definition.

It is also assumed that the atmosphere rotates with the same angular velocity as the Earth.

The definition of the Earth atmosphere is important for instrument pointing task and refraction.

8.3.3.1 US Standard Atmosphere 1976

The U.S Standard Atmosphere 1976 Atmosphere model is modified as follows:

- it ranges from Z = 0 Km to Z = 86 Km.
- the ratio M/M_0 decreases linearly from Z = 80 to Z = 86 Km.
- the linear relationship between T_M and H is replaced by either an arc of a circle or by a polynomial function in the vicinity of the points where the molecular-scale temperature gradient changes, in order to have a continuous and differentiable function $T_M = f(H)$

The U.S Standard Atmosphere 1976 is defined as follows (STD76 reference):

- The air is assumed to be dry, and at altitudes sufficiently below 86 Km, the atmosphere is assumed to be homogeneously mixed with a relative-volume composition leading to a constant mean molecular weight M.
- The air is treated as if it were a perfect gas, and the total pressure P, temperature T, and total density ρ at any point in the atmosphere are related by the equation of state, i.e. the perfect gas law, one form of which is:

$$P = \frac{\rho R T}{M}$$

where $R = 8.31432 \times 10^3$ [Nm/(KmolK)] is the universal gas constant.

- Besides the atmosphere is assumed to be in hydrostatic equilibrium, and to be horizontally stratified so that dP, the differential of pressure, is related to dZ, the differential of geometric altitude, by the relationship:

$$dP = -g\rho dZ$$

where g is the altitude-dependent acceleration of gravity, which can be calculated with the expression:

$$g = g_0\left(\frac{r_0}{r_0 + Z}\right)^2$$

where r_0 = 6356766 [m] and g_0 = 9.80665 [m/s^2], and that yields:

$$H = \frac{r_0 Z}{r_0 + Z}$$

where H is the geopotential altitude.

- The molecular-scale temperature T_M at a point is defined as:

$$T_M = T\frac{M_0}{M}$$

where M_0 = 28.9644 [Kg/Kmol] is the sea-level value of M.

In the region from Z = 0 Km to Z = 80 Km M is constant and M = M_0, whereas between Z = 80 Km and Z = 86 Km, the ratio M/M_0 is assumed to decrease from 1.000000 to 0.999578

Up to altitudes up to 86 Km the function T_M versus H is expressed as a series of seven successive linear equations. The general form of these linear equations is:

$$T_M = T_{M,b} + L_{M,b}(H - H_b)$$

The value of $T_{M,b}$ for the first layer (b = 0) is 288.15 [K], identical to T_0 the sea-level value of T.

The six values of H_b and $L_{M,b}$ are:

Table 11: Molecular-scale temperature coefficients

Subscript	Geopotential altitude H_b [Km]	Molecular-scale temperature gradient LM,b [K/Km]
0	0	-6.5
1	11	0.0
2	20	1.0
3	32	2.8
4	47	0.0

Table 11: Molecular-scale temperature coefficients

Subscript	Geopotential altitude H_b [Km]	Molecular-scale temperature gradient LM,b [K/Km]
5	51	-2.8
6	71	-2.0
7	84.8520 (Z = 86)	

Finally, the pressure can be calculated with the following expressions:

$$P = P_b\left(\frac{T_{M,b}}{T_{M,b} + L_{M,b}(H - H_b)}\right)^{\frac{g_0 M_0}{R L_{M,b}}} \quad (L_{M,b} \neq 0)$$

$$P = P_b \cdot \exp\left[\frac{-g_0 M_0 (H - H_b)}{R T_{M,b}}\right] \quad (L_{M,b} = 0)$$

The reference-level value for P_b for b = 0 is the defined sea-level value P_0 = 101325.0 N/m^2. Values of P_b for b = 1 through b ≥ 6 are obtained from the application of the appropriate equation above for the case when $H = H_{b+1}$.

8.4 Sun and Moon

Sun and Moon position and velocity in the True of Date reference frame will be calculated according to FLANDERN reference.

8.5 Stars

To calculate the look direction from the satellite to a star, two consecutive steps must be performed:

- To calculate the stars coordinates in the Mean of Date reference frame at the current epoch, taking as input a star catalogue (assumed to be expressed in the Barycentric Mean of 2000.0 reference frame for the epoch J2000.0).
- To calculate the star coordinates in the Satellite Relative Actual Reference frame at the same epoch.

The first step must apply the following corrections:

Table 12: First step correction of star looking direction

Correction	Description	Effect
Proper motion	Intrinsic motion of the star across the background with respect to a reference epoch (e.g J2000.0) leading to a change in the apparent star position at the current epoch	Lower than 0.3 mdeg/year

Table 12: First step correction of star looking direction

Correction	Description	Effect
Annual parallax	Apparent displacement of the position of the star caused by the difference in the position of the barycenter and the position of the Earth with the motion of the Earth around the Sun during the year	Lower than 0.3 mdeg
Light deflection	Gravitational lens effect of the Sun	Lower than 500 μdeg at the limb of the Sun and falling off rapidly with distance, e.g. to 6 μdeg at an elongation of 20 deg (so it will be ignored)
Annual aberration	Apparent displacement of the position of the star caused by the finite speed of light combined with the motion of the Earth around the Sun during the year	Lower than 6 mdeg
Precession	Change of the position of the star caused by the transformation from the Geocentric Mean of 2000.0 to the Mean of Date reference frame	Lower than 6.0 mdeg/year

whereas the second step must apply the following ones:

Table 13: Second step corrections of star looking direction

Correction	Description	Effect
Satellite parallax	Apparent displacement of the position of the star caused by the difference in the position of the satellite and the position of the Earth with the motion of the satellite around the Earth during an orbit	Lower than 0.015 μdeg even for the closest stars (so it will be ignored)
Satellite aberration	Apparent displacement of the position of the star caused by the finite speed of light combined with the motion of the satellite around the Earth during an orbit	Lower than 1 mdeg for LEO spacecraft

8.5.1 Stars Positions

To apply some of the necessary corrections to calculate the coordinates of a star in the Satellite Relative Actual Reference frame, the following expressions shall be used (ALMAN95 reference):

- Get the following variables from a star catalogue:
  - Right ascension at J2000.0 expressed in the Barycentric Mean of 2000.0: α_0 [rad]
  - Declination at J2000.0 expressed in the Barycentric Mean of 2000.0: δ_0 [rad]
  - Proper motion in the right ascension: μ_α [rad/century]
  - Proper motion in the declination: μ_δ [rad/century]
  - Radial velocity: ν [au/century]
  - Parallax: π [rad]
- Correct the star position obtained from the star catalogue (α_0, δ_0) for the proper motion and annual parallax effects using the expressions:

$$\bar{q} = (\cos\alpha_0\cos\delta_0, \sin\alpha_0\cos\delta_0, \sin\delta_0)$$

$$\bar{m} = (m_x, m_y, m_z)$$

$$m_x = -\mu_\alpha\cos\delta_0\sin\alpha_0 - \mu_\delta\sin\delta_0\cos\alpha_0 + \upsilon\pi\cos\delta_0\cos\alpha_0$$

$$m_y = \mu_\alpha\cos\delta_0\cos\alpha_0 - \mu_\delta\sin\delta_0\sin\alpha_0 + \upsilon\pi\cos\delta_0\sin\alpha_0$$

$$m_z = \mu_\delta\cos\delta_0 + \upsilon\pi\sin\delta_0$$

$$\bar{P} = \bar{q} + T\bar{m} - \pi\bar{r}_{B,\,Earth}$$

where T = (t - 0.5)/36525, and t is the current TDT expressed in the MJD2000 format, and $\bar{r}_{B,\,Earth}$ is the position of the Earth in AU at that TDT, expressed in the Barycentric Mean of 2000 reference frame.

- Correct the star position for the annual aberration effect, using the following expressions:

$$\bar{p}_2 = \frac{\dfrac{\bar{p}_1}{\beta} + \left(1 + \dfrac{\bar{p}_1 \bullet \bar{v}}{1 + \frac{1}{\beta}}\right)\bar{v}}{1 + \bar{p}_1 \bullet \bar{v}}$$

$$\bar{v} = \frac{\bar{v}_{B,\,Earth}}{c} = 0.0057755\bar{v}_{B,\,Earth}$$

$$\beta = \frac{1}{\sqrt{1 - |\bar{v}|^2}}$$

where $\bar{v}_{B,\,Earth}$ is the velocity of the Earth in AU/d at the current TDT expressed in the Barycentric Mean of 2000 reference frame.

- The satellite aberration can be calculated with the expression (IERS_SUPL reference):

$$\Delta\theta = \operatorname{asin}\left[\frac{v}{c}\sin\theta - \frac{1}{4}\left(\frac{v}{c}\right)^2 \sin 2\theta\right] \text{ [rad]}$$

where $\Delta\theta$ is the change in the look direction from the satellite to the star, v is the velocity of the satellite expressed in the True of Date reference frame, and c is the velocity of the light in a vacuum.

The following drawing sketches the satellite aberration:

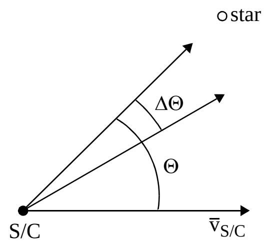


Figure 18: Satellite aberration

9 UNITS

In general, the units that will be used in all the CFI software will be the SI units, except for the angle that will use the ***degree*** instead of the ***radian***.

Table 14: Units in CFI Software

Quantity	Unit	Symbol
Length	meter	m
Mass	kilogram	kg
Time	second	s
Thermodynamic temperature	kelvin	K
Amount of substance	mole	mol
Plane angle	degree	deg
Frequency	hertz	Hz
Pressure	pascal	Pa

ANNEX A. MISSIONS USAGE OF CONVENTIONS

The current annex particularises the concepts presented throughout the document for the different missions.

A.1 Time References and Models

A.1.1 Time References

The different missions utilise a sub-set of those time references presented in the section 4.1. The table below shows the time references applicable to each particular one:

Table 15: Time references usage

Time reference	ERS 1/2	Envisat	CryoSat	GOCE (TBC)	SMOS (TBC)	ADM-Aeolus (TBC)
Universal Time (UT1)	X	X	X	-	X	X
Universal Time Coordinated (UTC)	X	X	X	X	X	X
International Atomic Time (TAI)	X	X	X	X	X	X
GPS Time (GPS)	-	-	-	X	X	X

A.1.2 Time formats

The different missions utilise a sub-set of those time formats presented in the section 4.2. The table below shows the time formats applicable to each particular one:

Table 16: Time formats usage

Time format		ERS 1/2	Envisat	CryoSat	GOCE (TBC)	SMOS (TBC)	ADM-Aeolus (TBC)
Processing		X	X	X	X	X	X
Transport	Standard	X	X	X	X	X	X
	Envisat Ground Segment	X	X	-	-	-	-
	CryoSat Ground segment	-	-	X	-	-	-
	CryoSat General telemetry	-	-	X	-	-	-
	CryoSat SIRAL telemetry	-	-	X	-	-	-
	SMOS Transport format	-	-	-	-	X	-

Table 16: Time formats usage

	Time format	ERS 1/2	Envisat	CryoSat	GOCE (TBC)	SMOS (TBC)	ADM-Aeolus (TBC)
ASCII	Standard	X	X	X	X	X	X
	Standard with reference	-	-	X	-	-	-
	Standard with microseconds	-	-	X	-	-	-
	Standard with ref. and microsecs	-	-	X	-	-	-
	Compact	-	-	X	-	-	-
	Compact with reference	-	-	-	-	-	-
	Compact with microseconds	-	-	-	-	-	-
	Compact with ref. and microsecs	-	-	-	-	-	-
	CCSDS-A	-	-	-	-	-	-
	CCSDS-A with reference	-	-	X	X	X	X
	CCSDS-A with microsecs	-	-	-	-	-	-
	CCSDS-A with ref. and microsecs	-	-	-	-	-	-
	CCSDS-A compact	-	-	X	X	X	X
	CCSDS-A compact with reference	-	-	-	-	-	-
	CCSDS-A compact with microsecs	-	-	-	-	-	-
	CCSDS-A compact with ref. and microsecs	-	-	-	-	-	-
	Envisat	X	X	-	-	-	-
	Envisat with microseconds	X	X	-	-	-	-

A.2 On-board times

The different missions utilise a sub-set of those on-board times presented in section 4.4. The table below shows the on-board time applicable to each particular.

Table 17: On-board times usage

On-board time	ERS 1/2	Envisat	CryoSat	GOCE (TBC)	SMOS (TBC)	ADM-Aeolus (TBC)
Satellite Binary Time	X	X	-	-	-	-
On-Board Time	X	X	-	X	X	X
TAI reference	-	-	X	-	-	-

A.3 Reference Frames

The different missions utilise a sub-set of those reference frames presented in the section 5.1. The table below shows the time formats applicable to each particular one:

Table 18: Reference frames usage

Reference frame	ERS 1/2	Envisat	CryoSat	GOCE (TBC)	SMOS (TBC)	ADM-Aeolus (TBC)
Galactic	X	X	-	-	-	-
Barycentric Mean of B1950	-	X	-	-	-	-
Barycentric Mean of 2000	X	X	X	-	-	-
Heliocentric Mean of 2000	X	X	-	-	-	-
Geocentric Mean of 2000	X	X	-	-	-	-
Mean of Date	X	X	X	X	X	X
True of Date	X	X	X	-	X	X
Earth fixed	X	X	X	X	X	X
Topocentric	X	X	X	-	X	X
Satellite	X	X	X	X	X	X
Satellite relative	X	X	X	X	X	X
Satellite relative actual	X	X	X	X	X	X

A.4 Orbit Characterisation

A.4.1 Orbit Definition

The different missions utilise a sub-set of those orbit definitions presented in the section 6.1. The table below shows the orbit characteristics applicable to each particular one:

Table 19: Orbit characterisation

Orbit Characteristic	Sun-synchronous	Sun-synchronous	Sun-synchronous
ERS 1/2	X	X	-
Envisat	X	X	-
CryoSat	-	X	X
GOCE	X	X	-
SMOS	X	X	-
ADM-Aeolus	X	X	-
Terrasar			
EarthCare			
Swarm A			
Swarm B			
Swarm C			
Sentinel 1A			
Sentinel 1B			

A.4.1.1 Orbit Parameters

The nominal orbit of each mission has the following mean elements:

Table 20: Orbit mean elements

Mean element	Semi-major axis (km) [a]	Eccentricity (initial frozen orbit) [e]	Inclination (deg) [i]	Argument of perigee (deg) [w]	Right ascension at ascending node [RA_{AN}]	Mean Local Solar Time drift (deg/day) [$MLST_{drift}$]	Mean altitude (km) [h]
ERS 1/2	7153.139	0.001165	98.5228	90.000	10:30 hours	0.000	793.445
Envisat	7159.493	0.001165	98.549387	90.000	22:00 hours	0.000	799.790

Table 20: Orbit mean elements

Mean element	Semi-major axis (km) [a]	Eccentricity (initial frozen orbit) [e]	Inclination (deg) [i]	Argument of perigee (deg) [w]	Right ascension at ascending node [RA_{AN}]	Mean Local Solar Time drift (deg/day) [$MLST_{drift}$]	Mean altitude (km) [h]
CryoSat	7096.643	0.0012	92.000000	90.000	130.000 deg	0.784	718.506
GOCE	6613	< 0.0045	96.500000	90.000	06:00 or 18:00 hours	0.000	250
SMOS	7134.552	0.00116	98.445129	90.000	TBC	06:00	755
ADM-Aeolus	6767.958	0.001286	97.016300	90.000	TBC	18:00	400
Terrasar	7006.174	0.001203	97.922730	90.000	TBC	TBC	TBC
EarthCare	6827.559	0.001265	97.235612	90.000	TBC	TBC	TBC
Swarm A	6828.135	0.001258	86.7	90.000	TBC	TBC	TBC
Swarm B	6828.135	0.001258	86.7	90.000	TBC	TBC	TBC
Swarm C	6908.138	0.001230	87.3	90.000	TBC	TBC	TBC
Sentinel 1A	7070.982	0.001181	98.183402	90.000	TBC	TBC	TBC
Sentinel 1B	7070.982	0.00118	98.183402	90.000	TBC	TBC	TBC

The repeat cycle and cycle length are the following:

Table 21: Repeat cycle and Cycle length

Element	Repeat cycle (days)	Cycle length (orbits)
ERS 1/2	35 / 3	504 / 43
Envisat	35 / 3	501 / 14
CryoSat	369 / 2	5344 / 29
GOCE	TBC (< 2 months)	TBC
SMOS	149	2144
ADM-Aeolus	7	109
Terrasar	TBC	TBC
EarthCare	TBC	TBC

Table 21: Repeat cycle and Cycle length

Element	Repeat cycle (days)	Cycle length (orbits)
Swarm A	TBC	TBC
Swarm B	TBC	TBC
Swarm C	TBC	TBC
Sentinel 1A	TBC	TBC
Sentinel 1B	TBC	TBC

The operational orbits will be controlled in order to keep predefined parameters within allowed tolerances.

Table 22: Orbit tolerances

Element	Ground track deviation (km)	MLST at ascending node (min)	Mean altitude (m)
ERS 1/2	TBC	TBC	TBC
Envisat	1	5	68
CryoSat	5	-	-
GOCE	-	-	-
SMOS	TBC	TBC	TBC
ADM-Aeolus	TBC	TBC	TBC
Terrasar	TBC	TBC	TBC
EarthCare	TBC	TBC	TBC
Swarm A	TBC	TBC	TBC
Swarm B	TBC	TBC	TBC
Swarm C	TBC	TBC	TBC
Sentinel 1A	TBC	TBC	TBC
Sentinel 1B	TBC	TBC	TBC

The CFI software will check the compliance of the orbit supplied on input with a set of requirements on the main osculating Kepler elements.

These requirements are less stringent than those contained in the previous table.

In fact, two categories of tolerance requirements will be checked:

- **Tight requirements**: the orbit is very close to the nominal CryoSat one.
- **Loose requirements**: the orbit is far from the nominal CryoSat one, but still is a low eccentric, quasi polar and quasi Sun-synchronous orbit, like CryoSat orbit.

Table 23: Loose tolerance requirements

Osculating Kepler element	Semi-major axis (m)	Eccentricity	Inclination (deg)
ERS 1/2	7000000 / 7300000	0.0 / 0.1	98.0 / 99.0
Envisat	7000000 / 7300000	0.0 / 0.1	98.0 / 99.0
CryoSat	1000000 / 10000000	0.0 / 0.5	60.0 / 120.0
GOCE	1000000 / 10000000	0.0 / 0.5	60.0 / 120.0
SMOS	7040000 / 7220000	0.0 / 0.1	97.1 / 99.7
ADM-Aeolus	6680000 / 6860000	0.0 / 0.1	95.7 / 98.3
Terrasar	6915000 / 7095000	0.0 / 0.1	96.6 /99.2
EarthCare	6770000 / 6900000	0.0 / 0.5	96.8 / 97.6
Swarm A	6500000 / 6975000	0.0 / 0.5	85.0 / 89.0
Swarm B	6500000 / 6975000	0.0 / 0.5	85.0 / 89.0
Swarm C	6500000 / 6975000	0.0 / 0.5	85.0 / 89.0
Sentinel 1A	7000000 / 7140000	0.0 / 0.5	97.7 / 98.7
Sentinel 1B	7000000 / 7140000	0.0 / 0.5	97.7 / 98.7

Table 24: *Tight tolerance requirements*

Osculating Kepler element	Semi-major axis (m)	Eccentricity	Inclination (deg)
ERS 1/2	7000000 / 7300000	0.0 / 0.1	98.0 / 99.0
Envisat	7000000 / 7300000	0.0 / 0.1	98.0 / 99.0
CryoSat	1000000 / 10000000	0.000 / 0.500	60.0 / 120.0
GOCE	6500000 / 6700000	0.000 / 0.500	96.0 / 97.0
SMOS	7090000 / 7170000	0.000 / 0.007	98.1 / 98.7
ADM-Aeolus	6730000 / 6810000	0.000 / 0.007	96.7 / 97.3
Terrasar	6965000 / 7045000	0.000 / 0.007	97.60 / 98.20
EarthCare	6800000 / 6870000	0.000 / 0.007	96.9 / 97.5
Swarm A	6500000 / 6925000	0.000 / 0.007	85.85 / 88.15
Swarm B	6550000 / 6925000	0.000 / 0.007	85.85 / 88.15
Swarm C	6550000 / 6925000	0.000 / 0.007	85.85 / 88.15
Sentinel 1A	7035000 / 7105000	0.000 / 0.007	97.8 / 98.6
Sentinel 1B	7035000 / 7105000	0.000 / 0.007	97.8 / 98.6

If the tight tolerance requirements are not satisfied, but the loose ones are, then a warning will be returned by the CFI software. If even the loose tolerance requirements are not satisfied, then an error will be returned.

A.4.2 Orbit Types

The different missions utilise a sub-set of those orbit types presented in the section 6.2. The table below shows the orbit types applicable to each particular one sorted from less to most accuracy:

Table 25: Orbit types usage

Orbit Type	ERS 1/2	Envisat	CryoSat	GOCE	SMOS (TBC)	ADM-Aeolus (TBC)
Reference	X	X	X	-	X	X
FOS Predicted	X	X	X	X	X	X
DORIS Navigator	X	X	X	-	X	X
FOS Restituted	X	X	-	-	-	-
DORIS Preliminary	X	X	X	-	X	X
DORIS Precise	X	X	X	-	X	X
GOCE GPS	-	-	-	X	-	-

A.4.3 Propagation Models

The different missions utilise a sub-set of those orbit propagation models presented in the section 6.4. The table below shows the orbit propagation models applicable to each particular one:

Table 26: Orbit propagation models usage

Orbit Propagation Model	ERS 1/2	Envisat	CryoSat	GOCE	SMOS (TBC)	ADM-Aeolus (TBC)
Simulation	X	X	-	X	X	X
Operational	X	X	-	X	X	X
Non-Sun-synchronous simulation	-	-	X	-	-	-
Non-Sun-synchronous operational	-	-	X	-	-	-

A.4.4 Attitude Control

The different missions utilise a sub-set of those attitude control modes presented in the section 8.1. The table below shows the attitude control modes applicable to each particular one:

Table 27: AOCS modes usage

AOCS Mode	ERS 1/2	Envisat	CryoSat	GOCE	SMOS (TBC)	ADM-Aeolus (TBC)
Coarse Acquisition Mode (CAM)	X	X	-	-		
Coarse Pointing Mode (CPM)	-	-	X	-		
Fine Acquisition Mode 1 (FAM1)	X	X	-	-		
Fine Acquisition Mode 2 (FAM2)	X	X	-	-		
Fine Acquisition Mode 3 (FAM3)	X	X	-	-		
Fine Pointing Mode (FPM)	X	X	X	-		
Orbit Control Mode (OCM)	X	X	X[a]	-		
Stellar Yaw Steering Mode (SYSM)	X	X	-	-		
Yaw Steering Mode (YSM)	X	X	X	-		
Stellar Fine Control Mode (SFCM)	X	X	X[b]	-		
Fine Control Mode (FCM)	X	X	X[b]	-		
Satellite Safe Mode (SSM)	X	X	X[c]	-		
Local Normal Pointing (LNP)	-	-	X	-		

a. with different requirements from those listed: TBD bias pointing in pitch and roll, and up to 180 degrees yaw slew

b. covered with OCM

c. identical to CPM

A.4.5 DRS-Artemis satellites

The different missions utilise a sub-set of those DRS satellites presented in the section 8.2. The table below shows the DRS satellites applicable to each particular one:

Table 28: DRS satellite usage

DRS Satellite	ERS 1/2	Envisat	CryoSat	GOCE	SMOS (TBC)	ADM-Aeolus (TBC)
DRS-Artemis	X	X	-	-	X	X

A.4.6 Earth

A.4.6.1 Earth Geometry

The different missions utilise a sub-set of those Earth geometry definitions presented in the section 8.3.2. The table below shows the Earth geometry definitions applicable to each particular one:

Table 29: Earth ellipsoid definition usage

Earth Geometry	ERS 1/2	Envisat	CryoSat	GOCE	SMOS (TBC)	ADM-Aeolus (TBC)
WGS 84	X	X	X	-	X	X