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JASON-3

JASON-3 CHARACTERISTICS FOR POD PROCESSING

	Date	Signature
Prepared by : **V.COUDERC**	31/08/15	[signature]
Approved by :		
For application : **G. ZAOUCHE**	31 Aug 2015	[signature]

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5.1 JASON-3 MISSION DESCRIPTION

See RD8 for more details.

The objective of JASON-3 mission is to provide a continuation of the TOPEX/Poseidon, JASON-1 and JASON-2 missions and their collection of high accuracy radar altimetry measurements for global ocean circulation and sea surface studies, without any data gaps.

Data from JASON-3 will be used to provide:

- a near-real time data (and product) service for operational activities such as marine nowcasting and numerical prediction of sea state, ocean circulation and weather.
- Offline data (and product) services to support research and operational requirements.

The JASON-3 satellite consists of a satellite-bus (PROTEUS), carrying a payload module (PIM). The payload module will include the POSEIDON-3B radar altimeter and its antenna, the Doppler Orbitography and Radiopositioning Integrated by Satellite (DORIS) receiver package, a microwave radiometer and its antenna (AMR), a laser retroreflector (LRA), a Global Positioning System (GPSP) receiver package provided by NASA. , and 2 additional experiments (non core-mission) for the radiations effect measurement (CARMEN3/AMBRE and LPT).

Nota : The GPSP is part of the Core Mission but is considered as a non critical Instrument.

The JASON-3 satellite shall be launched from the Western Range Facility (Vandenberg Air Force Base (VAFB), USA), aboard a Falcon-9 launcher (v1.1 with a payload fairing) in a single launch configuration.

After the assessment phase, the JASON-3 satellite mission objective is to operate for a period of five years. The Proteus PF design is based on a nominal life time of 3 years, the consumables will be compatible of the objective and the reliability will be computed for 5 years.

5.2 MISSION ORBIT CHARACTERISTICS

For details, see also RD2.

5.2.1 OPERATIONAL ORBIT DEFINITION

Jason-3 operational orbit has the same characteristics as TOPEX's, Jason-1's and Jason-2's. It is a near-circular and frozen orbit that follows an exact repeating ground track every 127 revolutions in a little less that ten days. The inclination of 66 deg enables to cover most of the non-frozen seas (from 66°N to 66°S in latitude).

Mean Orbital parameters of the nominal orbit :

Parameters	Value
Semi-major axis	a=7714431 m
Eccentricity	e = 9.4 E-05
Inclination	i = 66.038309 °
Argument of periapsis	90°
Reference equatorial altitude	1336 km
Cycle duration	9.91564 days
Number of revolution in a cycle	127 orbits

Jason-2 and Jason-3 must follow the same ground track, and both satellites will be on the same phased orbit.

5.2.2 STATION ACQUISITION

The main goal is to position JASON-3 on its nominal orbit (within the station-keeping margin) by correcting the launcher dispersions, the initial gap on the semi-major axis and phase the satellite.

During assessment phase and if JASON-2 is still operational, both JASON-2 and JASON-3 satellites will be phased on the same ground track with a time separation between the two satellites between 1 and 10 minutes ("tandem flight"). This gives the opportunity for cross-calibration of JASON-3 altimetry measurements with JASON-2 measurements.

The targeted injection orbit is 25 km below the mission orbit. The general strategy of orbit acquisition is detailed in RD2. According to the actual injection precision and the launch date (giving the cycle day for JASON-2 rendez-vous), several maneuvers will be performed :

- out-of-plane maneuver(s) to correct the inclination if needed
- in-plane maneuvers to correct Jason-2/Jason-3 phase lag and the launcher dispersions (correction of semi-major axis)
- a trim maneuver at the end of the station acquisition to initialize the station keeping

After a few months of instruments inter-calibration by having the 2 satellites in tandem flight, scientists would like to improve repetitiveness, by shifting JASON-2 ground track of a half-track interval (TBC) wrt JASON-3 ground track ("interleaved" orbit).

5.2.3 STATION KEEPING

Jason-3 mission requires an accurate control of an exactly repeating ground track. The orbit has to be maintained so that the ground track remains close to the reference grid. The specification is ± 1 kilometer perpendicular to the ground track at the equator.

The characteristics of each kind of orbit control maneuver is described in RD2.

5.2.4 ATTITUDE MODE

On board Jason-3, the attitude is controlled by reaction wheels and magnetotorquer bars.

The nominal attitude of the satellite is *geodetically Earth-pointed*.
Jason-3 is in *yaw steering mode* except for beta prime within +/- 15°. A yaw flip is made at 0°. Between +/- 15° the satellite is in fixed yaw.

5.2.5 CALIBRATION MANEUVERS

5.2.5.1 CROSS CALIBRATION MANEUVERS

The detailed specification of cross calibration maneuvers is provided in RD3.

Cross calibration maneuvers are performed on flight to identify the biases between the platform and the altimeter antenna pointing. It allows to identify mispointings in the pitch and roll axis with respect to the local nadir, thanks to specific manoeuvers on these two axis, while the alternate measurement derived from the altimeter provides a “half cone” measurement.

Some specific cross calibration maneuvers will also be performed for altimeter mispointing calibration purpose in assessment phase. The only differences with “regular” cross calibration maneuvers are :

- the duration of the mispointing which will be longer (5 min instead of 1 min)

- these maneuvers are performed in either roll or pitch axis in order to limit their duration.

5.2.5.2 AMR CALIBRATION MANEUVERS

The detailed specification of AMR calibration maneuvers is provided in RD4.

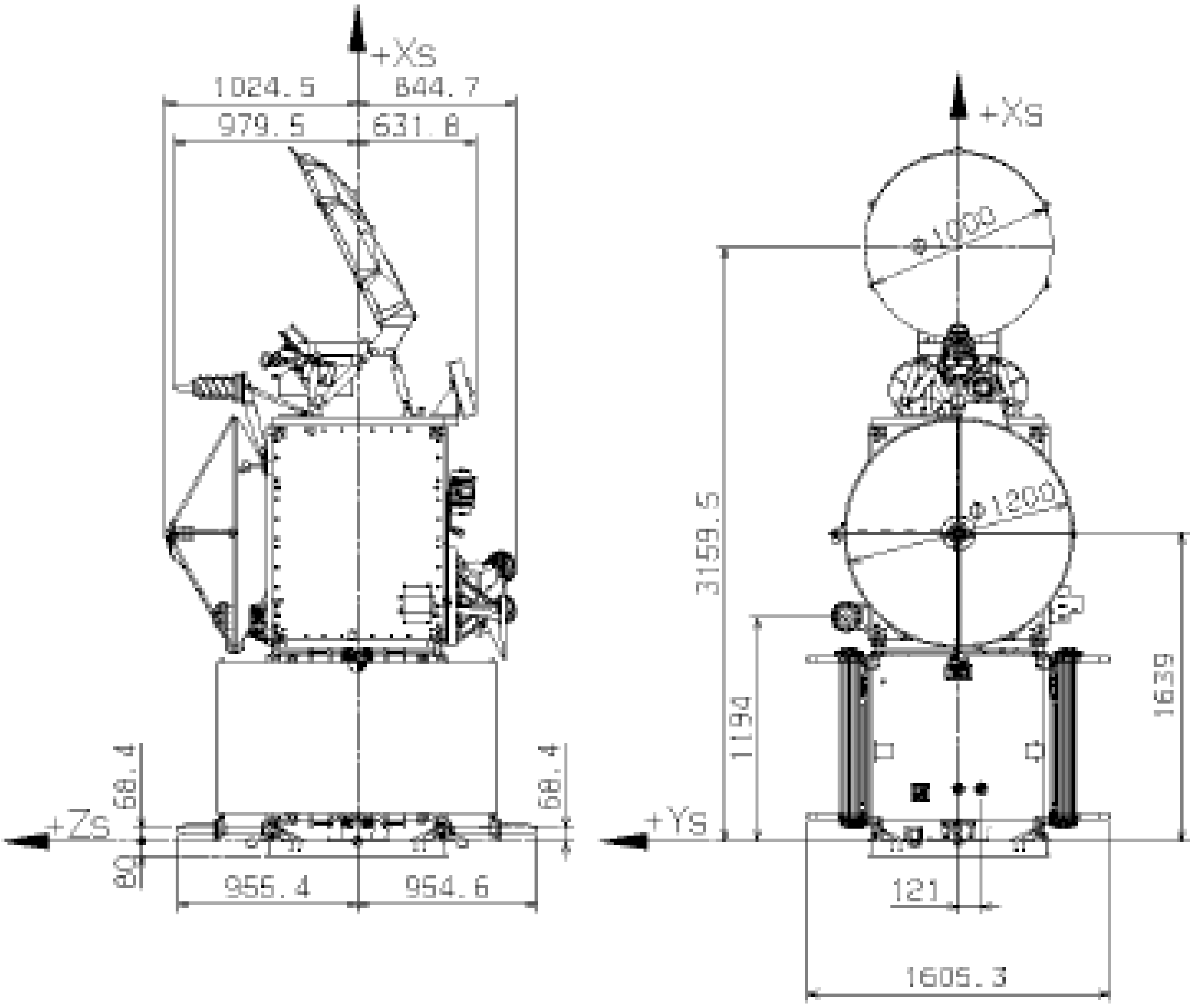
In flight, the AMR radiometer shall be calibrated by performing attitude maneuvers on the pitch axis of the satellite (around Y sat axis), with a magnitude of 80 degrees : these maneuvers allow to get a periodic cold space view through the AMR main reflector.

6. SATELLITE DESCRIPTION

The following parameters shall be used for the flight. These parameters values may be updated prior to the launch in order to provide the most accurate information available at that time. In that case, they may also be updated during the satellite lifetime, if any major change occurs.

6.1 SATELLITE VIEW AND REFERENCE FRAME

The following views of the satellite indicate the satellite reference frame (Xs, Ys and Zs directions). Zs corresponds to the nadir. During fixed yaw orbits, Ys is orthogonal to the orbital plane.



A complete set of detailed satellite pictures is available to provide details about thermo-optical properties of the external surfaces.

6.2 SATELLITE MOBILE PARTS

The position of the central body is given by the attitude information. The only mobile part that is included in the radiation model is the solar array (reaction wheels and gyros are mobile elements but not modelized).

The solar array is rotated along the Ys axis in order to be pointed towards the sun. The actual rotation angle relative to the central body is given in a dedicated file, in parallel of the attitude quaternions.

6.3 MASS PROPERTIES

The satellite mass values of :

- Beginning of Life Satellite mass (before orbit acquisition maneuver),
- Nominal Satellite total mass (beginning of mission, including moving parts if any),
- End of Life Satellite Mass

Are given (UNIT = kilograms) hereafter.

The Centre of Mass coordinates are also given in the Satellite Reference Frame. The Centre of Mass coordinates uncertainties and their evolution during the satellite are defined below.

The mass properties of JASON-3 have been measured at the end of the AIT sequence. The mass and the center of gravity (CoG) of the satellite, without propellant and without the solar wings, and with some remaining non flight items, have been measured with the following accuracies:

- Mass measurement accuracy: ± 0.11%.

  Nota : The additional variation of the total mass of Jason-3 throughout its mission is estimated to 0.411 kg corresponding to 0.0806% of nominal mass

- CoG measurement accuracy: +/- 0,54 mm on the X axis axis and Z, and to 0 on the Y axis (see §6.3.3)
- MOI measurement accuracy: +/- 1%

From these mass properties, the overall mass and global center of gravity location is computed all along the satellite life-time taking into account:

- the weight of the solar arrays measured on-ground and its center of gravity location estimated by analysis,

- the mass of the remaining hydrazine in the tank, computed on-ground by processing the outputs of the pressure transducer, and the associated center of gravity.

6.3.1 MASS BUDGET AND CENTERING DATA

The following table gives the present MCI's estimated for the JASON-3 satellite.

Platform mass data come from PROTEUS 5PF MCI post CALIPSO RQS with weighted units values, taking into account equipment modifications (MTB 180 A.m², BEU, Battery, Additional DHU board, harness modification).

Due to the payload centering, a real balancing masse of 12 kg have been taken into account (the maximum balancing mass is 20 kg).

The mass budget is based on the PROTEUS maximum propellant mass capability (28.3 kg including pressurant).

Current Best Estimate

The current best estimate (CBE) values of the JASON-3 spacecraft mass properties in the launch/separation/ flight configuration are shown below (from RD5 presented at satellite RQS, Nov 2014) :

Fig. 1 : Spacecraft MCI :

		MASS	CENTRING (mm)			INERTIA (m2Kg)					
	SA	(Kg)	X	Y	Z	Ixx	Iyy	Izz	PXY	PYZ	PXZ
DRY MASS	STOWED	481,3	1012,8	0,0	-2,2	116,4	320,5	318,6	1,7	2,6	7,3
LAUNCH	STOWED	509,6	977,1	0,0	-2,1	117,0	332,1	330,2	1,7	2,6	7,2
BOL	DEPLOYED	509,6	1002,3	0,0	-2,1	502,9	321,2	705,1	1,7	2,3	7,3
EOL (8 kg consumption)	DEPLOYED	501,6	1018,3	0,0	-2,1	502,7	317,8	701,7	1,7	2,3	7,3
EOL (total propellant	DEPLOYED	481,3	1039,5	0,0	-2,2	502,3	308,6	692,5	1,7	2,3	7,3

		MASS	CENTRING (mm)			INERTIA (m2Kg)					
	SA	(Kg)	X	Y	Z	Ixx	Iyy	Izz	PXY	PYZ	PXZ
BOL	No	467,2	1015,7	0,0	-2,2	96,7	314,0	304,0	1,7	2,7	7,3
EOL (8 kg consumption)	No	459,2	1033,4	0,0	-2,3	96,5	310,5	300,5	1,7	2,7	7,3
EOL (total propellant consumption)	No	438,9	1057,4	0,0	-2,4	96,1	300,9	290,9	1,7	2,7	7,3

Fig. 2 : Payload MCI :

PL MCI	Mass (Kg)	Center of Gravity (mm) in Fs			Inertia tensor (Kg.m²) in Fs at CoG					
	(Kg)	X	Y	Z	Ixx	Iyy	Izz	PXY	PYZ	PXZ
In Sat. ref. frame at PL CoG	204,1	1718,0	7,3	-14,0	50,5	89,6	82,7	0,8	2,4	9,9
In PL refernce frame at PL ref. frame Origin	204,1	648,0	7,3	-14,0	50,6	175,3	168,4	1,8	2,4	8,0

Note: The mass center and inertia tensor data is measured relative to the spacecraft coordinate system.

6.3.2 CENTER OF GRAVITY KNOWLEDGE

The uncertainty of the in-flight satellite center of gravity is depending on:

- uncertainty on mass and CoG measured in AIT,
- uncertainty on the estimation of the consumed hydrazine mass, uncertainty of the center of gravity location versus the filling ratio,
- uncertainty of the measured mass of the solar array sub-assembly and center of gravity location, taking into account potential bending effects.

M0 mass of satellite w/o hydrazine and solar wings,

G0 CoG location of satellite w/o hydrazine and solar wings, in satellite reference frame,

M1 mass of solar wings,

G1 CoG location of solar wings assembly in satellite reference frame,

M2 mass of hydrazine,

G2 CoG location of hydrazine in satellite reference frame,

M mass of the complete satellite = M0+M1+M2,

G center of gravity of the complete satellite,

the overall satellite center of gravity is given by:

$$G = \frac{M0.G0 + M1.G1 + M2.G2}{M}$$

and the uncertainty on the overall satellite center of gravity determination is derived as follows:

$$\Delta G = \frac{G0.(M1+M2).\Delta M0 - M0.G0.\Delta M1 - M0.G0.\Delta M2}{M^2} + \frac{M0.\Delta G0}{M} + \frac{G1(M0+M2).\Delta M1 - M1.G1.\Delta M2 - M1.G1.\Delta M0}{M^2} + \frac{M1.\Delta G1}{M} + \frac{G2.(M0+M1).\Delta M2 - M2.G2\Delta M0 - M2.G2.\Delta M1}{M^2} + \frac{M2.\Delta G2}{M}$$

The following data have been considered:

- ΔM0, uncertainty of satellite mass measurement in AIT: 0.478 kg (0.1% of 480,1 kg)

- ΔG0, uncertainty of satellite CoG measured in AIT: 2 mm,

- ΔM1, uncertainty of solar array mass measured on-ground: 0.100 kg,

- ΔG1, uncertainty of solar array CoG in life: 21 mm. The major part of this item is due to the solar array manufacturing and assembly. Knowledge partly achieved during AIT,

- ΔM2, uncertainty on the hydrazine remaining mass: 0.411 kg (see § 3.4.2.2),

- ΔG2, uncertainty on the location of center of gravity of the hydrazine: 4 mm. For memory, the tank supplier announced being compliant with a requirement of knowledge of the center of gravity location with 4 mm uncertainty, whatever the filling ratio is, under JASON-3 Normal Operational Mode accelerations conditions.

With the formula and hypotheses mentioned here above, the worst case is, combining linearly all **contributors:**

Delta COG	
δX_{COG}	4.75 mm
δY_{COG}	3.7 mm
δZ_{COG}	3.7 mm

6.3.3 CENTER OF GRAVITY POSITION VARIATION KNOWLEDGE

Concerning the variation of the Center of Gravity location, the uncertainty is mainly due to the solar array center of gravity position variation throughout the orbit.

2 identified contributors in the variation of SA COG location :

- Thermoelastic effects : the SA COG knowledge can vary according to on-board thermal conditions
- Mechanical effect due to the inter-panels articulations deformation uncertainty

The SA CoG knowledge is the following (see also RD36)

Xsa = -15,3 mm with an uncertainty of 1,6 mm during the day,

Xsa = -19,3 mm with an uncertainty of 1,6 mm during the night,

Ysa = 0 (global CoG of the 2 wings), uncertainty negligible,

Zsa = 0 mm with an uncertainty of 3.2 mm.

The Xsa variation knowledge between day and night can be transferred to the S/C by the ratio of mass:

during the day: , -15,3*42.3/509.6= -1,27 mm

during the night:, -19,3*42.3/509.6 = -1,6 mm

In order to simplify the logic we can consider day values as reference and the maximum uncertainty as follow: Xsa = -15,3 +/- 5,6 mm with 5,6 = Delta Day/Night + uncertainty.

Maximum uncertainty around Ysa is : $(5,6^2+3,2^2)^{1/2}$ =6,45 mm (all included : day/night effect , measurement uncertainties, alignment bias with Day values as reference).

This uncertainty can be expressed in S/C axes, using a projection of this variation on the S/C axes under the rotation by an angle α of the SA around Ysa:

Uncertainty Xsc = 6,45 x 42.3/509.6 x cos α = +/- 0,54 cos α

Uncertainty Ysc = 0

Uncertainty Zsc = 6,45 x 42.3/509.6 x sin α = +/- 0,54 sin α

Considering this calculation, the uncertainty variation knowledge of the satellite center of mass location is equal to a maximum 0,54 mm on the X axis axis and Z, and to 0 on the Y axis, for any position of the SA.

6.4 SATELLITE SURFACES

The satellite external surfaces are defined by their area (m^2) and the corresponding normal vector and material characteristics. They are grouped in different categories :

- Satellite body (ncluding main protruding elements excepted radiometer antenna and solar panels) : -X/+X, -Y/+Y, -Z/+Z areas
- Radiometer antenna : areas and normal vector
- Solar panels : +GS/-GS areas

The satellite surface properties are given for the solar spectrum and the infrared spectrum:

- Specular reflectivity coefficient
- Diffuse reflectivity coefficient
- Emissivity coefficient (s.u)

6.4.1 THERMO-OPTICAL PROPERTIES

It may be noticed that the optical coefficients evolve along the life of the Satellite mainly because of the degradation of materials due to radiation and other space environment. This evolution is not very well known and it is impossible to define optical properties with the required accuracy.

Nevertheless, the following array gives the optical coefficients for the beginning of life which can be considered with an accuracy better than 10%

Thermo-optical properties (Beginning of life)

	SSM	White paint	Black paint	Black MLI	Aluminium	Kapton (MLI)	Solar cell (SA)	Carbon (SA)	external first wing (SA)
CA (coef absorption)	0.1	0.18	0.96	0.96	0.1	0.35	0.65	0.7	0.3
Specularity	1	0	0	0.5	0.5	0.5	1	0	0
CS (coef spec.)	0.9	0	0	0.02	0.45	0.325	0.35	0	0
CD (coef diffusion)	0	0.82	0.04	0.02	0.45	0.325	0	0.3	0.7
Emissivity	0.78	0.84	0.88	0.88	0.04	0.77	0.82	0.95	0.84

The following array gives an estimation of the same coefficients for the end of life. In this case, the accuracy cannot be guaranteed

Thermo-optical properties (End of life)

	SSM	White paint	Black paint	Black MLI	Aluminium	Kapton (MLI)	Solar cell (SA)	Carbon (SA)	external first wing (SA)
CA (coef absorption)	0.16	0.32	0.96	0.96	0.31	0.45	0.85	0.9	0.38
Specularity	1	0	0	0.5	0.5	0.5	1	0	0
CS (coef spec.)	0.84	0	0	0.02	0.345	0.275	0.15	0	0
CD (coef diffusion)	0	0.68	0.04	0.02	0.345	0.275	0	0.1	0.59
Emissivity	0.78	0.84	0.88	0.88	0.04	0.77	0.82	0.95	0.84

And the following array gives a summary of thermo-optical properties of the satellite thermal control elements :

		ε moy	α min	α max
MLI PF	**Kapton 50 mm alu 1 face / coté brut (MLI externe)**	**0,77**	**0,32**	**0.49**
	Kapton 25 mm alu 1 face / coté brut (MLI interne)	**0,62**	**0,36**	**0,49**
Radiateur	**SSM Argenté Sheldhal**	**0,76**	**0,10**	**0,16**
	PSG 120 (peinture blanche)	**0,84**	**0,18**	**0,32**
Peinture noire	**Chemglaze Z306**	**0,88**	**0.96**	**0.96**
Traitement aluminium	**AOC sur yokes et prises ombilicales**	**0.49**	**0.43**	**0.67**
	Alodine 1200 sur AU4G1 (Poussoirs)	**0,35**	**0,30**	**0,43**
	Alodine sur Pieds de gerbage	**0.45**	**0.30**	**0.43**
Adaptateur	**Aluminium nickelé (I/F Lanceur)**	**0.11**	**0.42**	**0.60**
	Aluminium doré (I/F Lanceur)	**0.04**	**0.13**	**0.30**
GS et CSS	**Face externe des CSS**	**0,78**	**0.91**	**0.91**
	Cellules solaires (face active)	**0.82**	**0.75**	**0.85**
	Cellules solaires (face non active)	**0.70**	**0.92**	**0.92**

The following array gives the min/max temperature of the satellite radiative surfaces along 2 orbits with eclipses:

- a cold case with a 0° Sun orbital illumination (eclipse max),

- a hot case with a 15° Sun orbital illumination (beginning of yaw steering maneuver).

The last columns show the IR flux emitted by the radiative surfaces.

Remark : JASON-3 radiative surfaces are considered as identical to those of JASON-2. JASON-3 temperatures and emitted fluxes are different for the PL panels radiators, due to T2L2 passenger not reconducted on JASON-3.

	Jason-2 radiator temperatures (°C)				Srad (m²)	Jason-2 IR flux emitted (W)			
Sun orbital illumination	0° Cold case		15° Hot case			0° Cold case		15° Hot case	
Localisation	Min	Max	Min	Max		Min	Max	Min	Max
PF [-Ys] panel	0	1	13	17	0.27 (0.49*)	66	67	80	84
PF [+Ys] panel	-6	-3	9	10	0.36 (0.22*)	81	85	101	102
PF [+Zs] panel	7	9	20	23	0.43 (0.48*)	117	120	140	146
PF [-Zs] panel	19	21	22	25	0.06 (0.08*)	19	20	20	21
STA radiator	-5	-4	20	25	0.03 (0.03*)	7	7	10	10
PL [-Ys] panel	-16	-15	0	5	0.39 (*)	75	76	96	103
PL [+Ys] panel	-6	-4	3	5	0.54 (*)	121	125	139	143
MLI	Adiabatic equilibrium with the environment				-	-	-	-	-
Solar arrays	-90 (**)	70 (**)	-90 (**)	70 (**)	-	-	-	-	-

- (*) the values in brackets are estimated using the pictures of Jason 3 taken during the satellite integration, in order to have a better estimation of the true radiating surfaces. The surfaces for PF and PL have been cumulated for y axis.
- (**) These are worst cases temperatures, for solar arrays thermal modelisation the values that should be used out of eclipse are :
  - 55°C sun face /52°C shadow face (best case)
  - 55°C sun face /45°C shadow face (worst case)

6.5 EXTERNAL GEOMETRY

To help the modelling, some pictures are given hereafter, showing the satellite during integration.

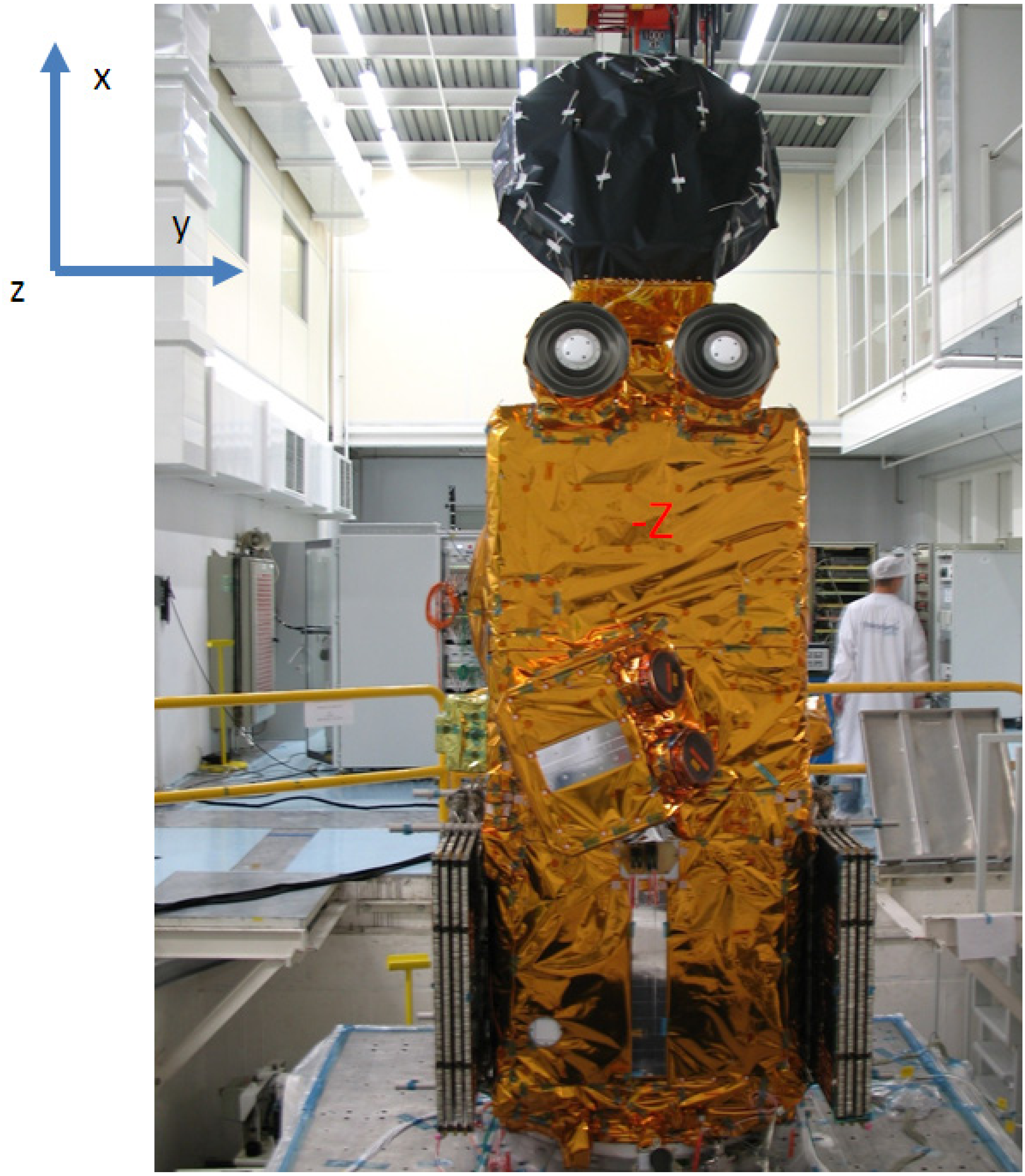


Figure : surface –Z (zenith)

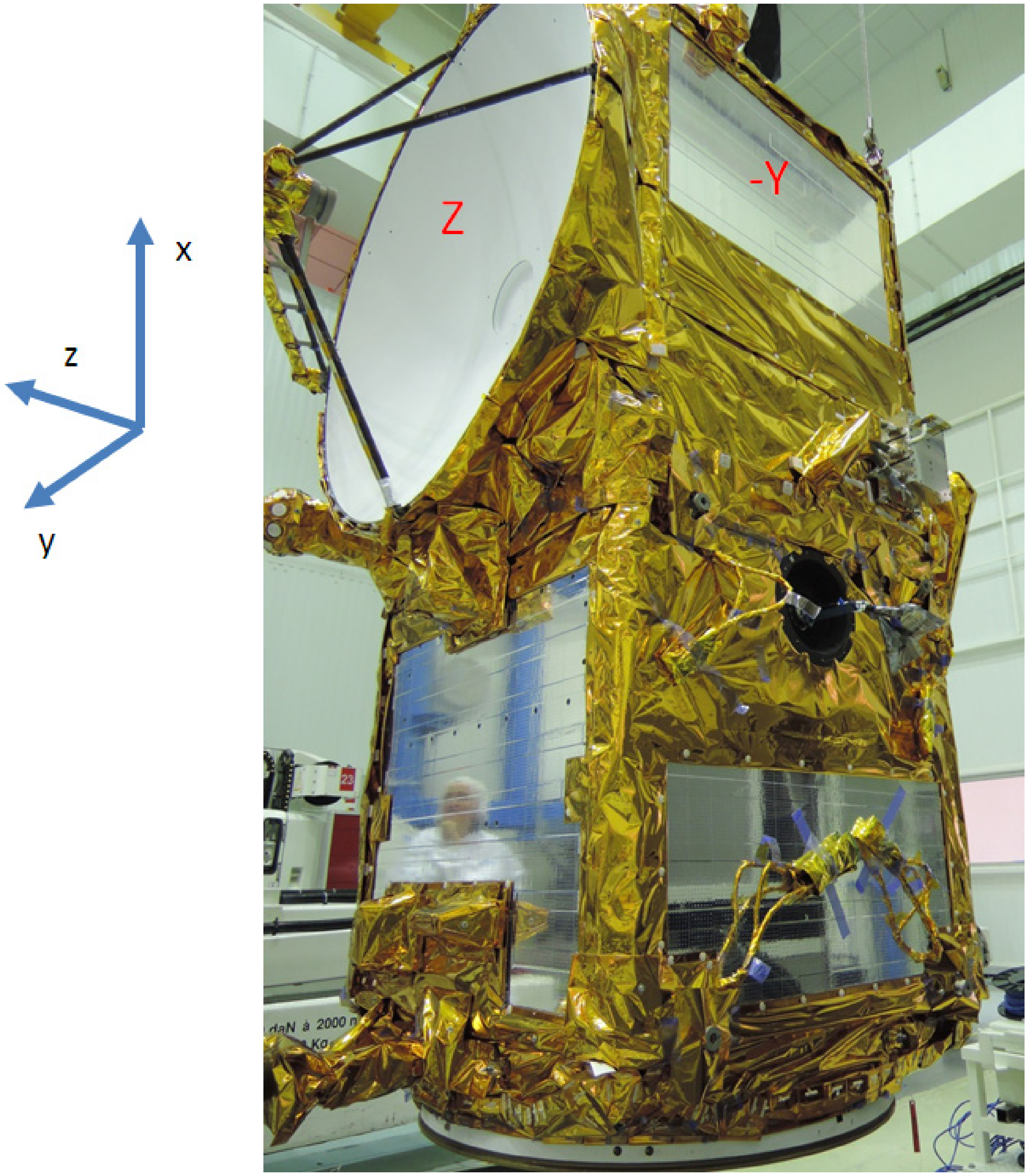


Figure : surfaces for +Z and -Y

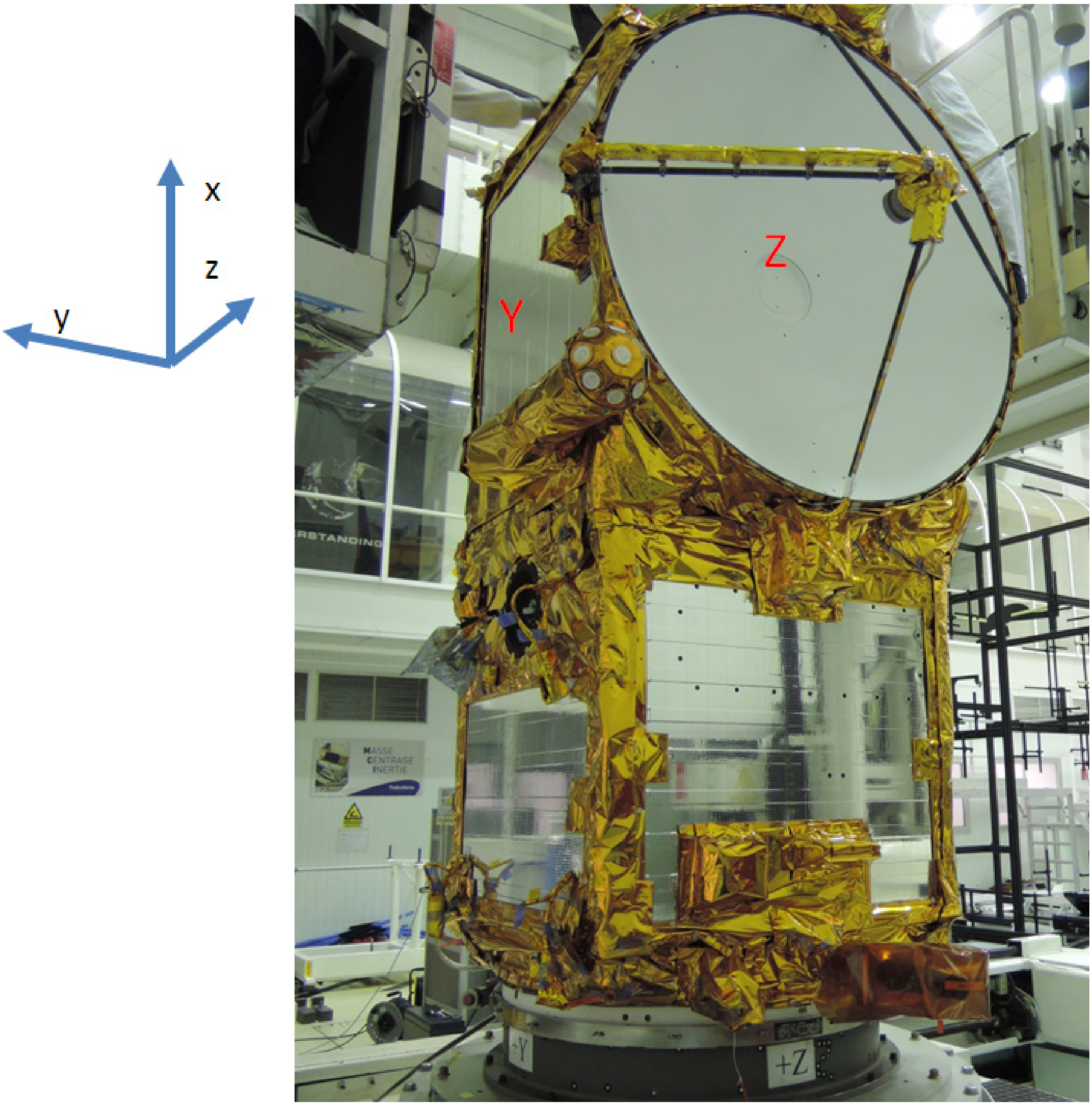


Figure : surfaces for +Z and +Y

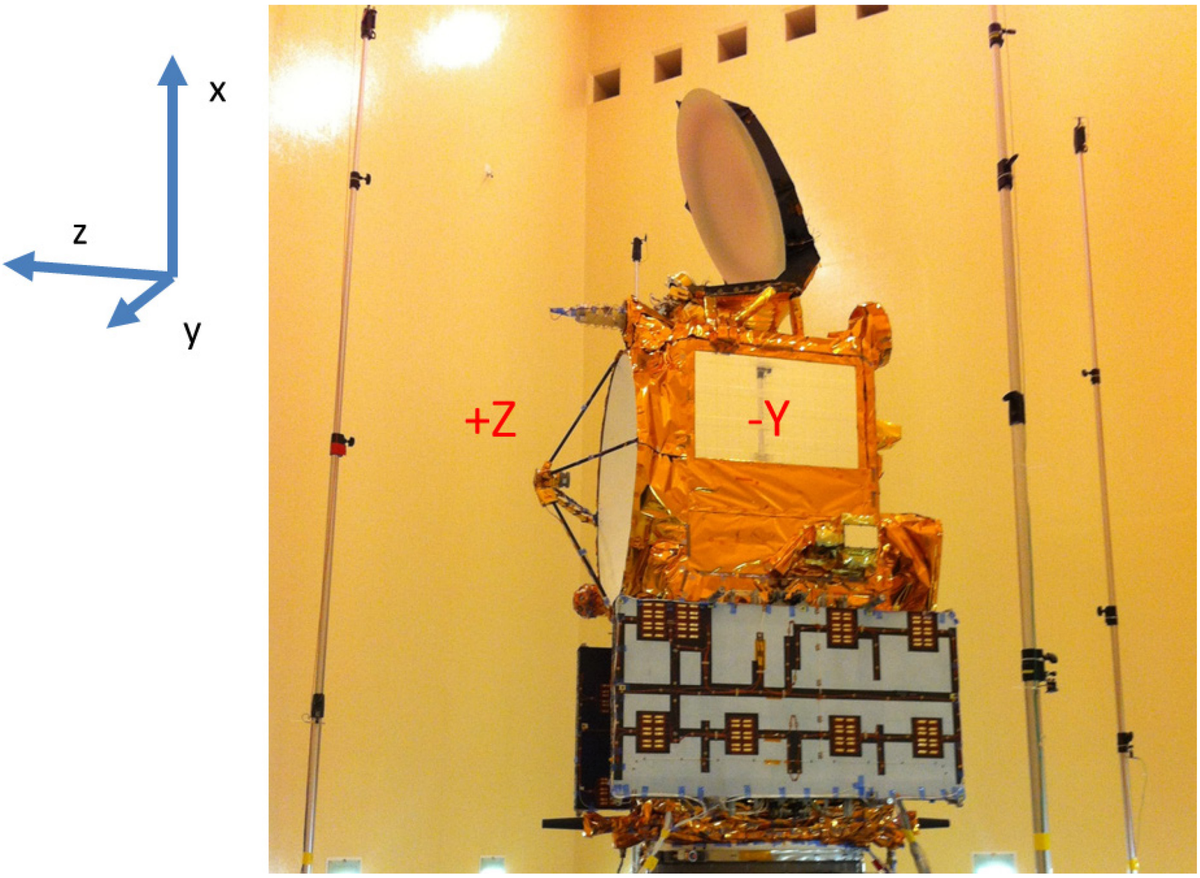


Figure : surfaces for +Z and -Y

The satellite central part dimensions are 0.885 m along z and y and 2.300 m along x. The following tables give the surfaces (unit=m^2) and the corresponding materials for the external characteristics of the satellite body and solar array. For the altimeter antenna, the surface corresponding to the exceeding parts in –y and +y (diameter is 1.200 m and platform width is 0.885) have been added as mli parts on –z surface. The radiometer is defined as an independent plate with the corresponding normal direction (30 degrees inclination toward +x).

Surfaces in m2 for each modelized S/C element

	S/C Axis	MLI	Black MLI	SSM	Aluminium	White paint	Solar cells	Carbon	external 1rst wing SA	***total***
PF + PL box (**)	-X	0.683			0.1					0.783
	+X	0.783								0.783
	-Y	1.16		0.88						2.04
	+Y	1.28		0.76						2.04
	-Z	2.11		0.07	0.14					2.32
	+Z	0.71		0.48		1.13				2.32
Radiometer reflector	-Zr (*)		0.906							0.906
	+Zr					0.906				0.906
GS	GS+				0.1		8.773	0.763		9.636
	GS-				0.1			7.152	2.384	9.636

(*) inclination of 30 degrees wrt +X axis (xOz plan) : [0.5; 0.; -0.866]

(**) these figures also include protruding elements like AMR electronic seen from Y or Z axis, STR seen from X or Y axis, and POS3B antenna seen from X or Y axis

6.6 INSTRUMENTS REFERENCE POINTS

These tables provide the *mechanical* reference point of each instrument antenna (instrument reference point for LRA) in the spacecraft reference frame (from RD6). The spacecraft reference frame is located at the bottom of the S/C, at the center of the bottom propulsion panel.

Reference points are given by three components in the satellite reference frame:

DORIS antenna RF axis is parallel to the +Z satellite axis and is nominally nadir pointed. GPS antennas are zenith pointed with a 15 degrees tilt towards the +X axis.

		S/C Reference Frame		
		X sat (mm)	Y sat(mm)	Z sat(mm)
Equpt frame origin		2412.8	-132.5	608.5
DORIS antenna	Xequpt	0	+1	0
	Yequpt	+1	0	0
	Zequpt	0	0	-1

		S/C Reference Frame		
		X sat (mm)	Y sat(mm)	Z sat(mm)
Equpt frame origin		2396.194	-217	-521.790
GPSPA antenna -Y	Xequpt	0.966	0	0.259
	Yequpt	0	-1	0
	Zequpt	0.259	0	-0.966

		S/C Reference Frame		
		X sat (mm)	Y sat(mm)	Z sat(mm)
Equpt frame origin		2396.194	217	-521.790
GPSPB antenna +Y	Xequpt	0.966	0	0.259
	Yequpt	0	-1	0
	Zequpt	0.259	0	-0.966

		S/C Reference Frame		
		X sat (mm)	Y sat(mm)	Z sat(mm)
Equpt frame origin		1639	0	455.09 *(*)*
POS3B antenna	Xequpt	+1	0	0
	Yequpt	0	+1	0
	Zequpt	0	0	+1

(*) See details in annex 1

		S/C Reference Frame		
		X sat (mm)	Y sat(mm)	Z sat(mm)
Equpt frame origin		1194	598	706.180
LRA	Xequpt	+1	0	0
	Yequpt	0	+1	0
	Zequpt	0	0	+1

7. DORIS PARAMETERS USED FOR POD PROCESSING

7.1 DORIS ANTENNA PHASE CENTER

Both antenna phase centres are assumed to be on the antenna Z axis. The following table gives the distances from the DORIS antenna *mechanical* reference point (antenna base plate) to the DORIS MV22 antenna *center of phase* (for each frequency):

Frequency	**X (mm)**	**Y (mm)**	**Z (mm)**	**Accuracy**
401.25	0.	0.	147mm	+/- 5 mm
2036.25	0.	0.	315mm	+/- 2 mm

The antenna phase centers for both 400 MHz and 2 GHz channels shall be then translated into spacecraft reference frame according to the table given in §6.6.

7.2 DORIS ANTENNA PHASE LAWS

Azimut and Elevation Antenna phase laws are described according to the phase center defined here above.

Fig 1 : Variation of phase in azimuth (FM22)

ε : Maximum difference compared to a law of linear phase y (Φ) = K Φ ± ε (K = constant)

Frequency (MHz)		401.25	2036.25
Specification ε (- 180° ≤ Φ ≤ + 180°)		≤ ± 4°	≤ ± 2°
ε obtained values for following values of φ	10°	±0.5°	±2°
	20°	±0.5°	±2°
	30°	±1.5°	±2°
	40°	±1.5°	±2°
	56°	±2°	±2°
	60°	± 2.5°	± 2°

Fig 2 : variation of phase in elevation (FM22)

ε : Maximum difference compared to a law of constant phase y(θ) = K ± ε

Frequency (MHz)		401.25	2036.25
Objective ε (- 56° ≤ θ ≤ 56°)		≤ ± 4°	≤ ± 2°
Valeurs ε obtenues	0°	± 1.7°	±2.5°
	22,5°	±1.2°	±2.5°
	45°	±1.4°	±1.5°
	67,5°	±1.8°	±2.5°
	90°	±2°	±3°
	112,5°	±2°	±3°
	135°	±2.1°	±2°
	157,5°	± 2.3°	±4°

LSB. = ± 2.0 deg.

Phase(Az) = Az +/- 2° TBC

Phase(El) = 0 +/- 2° TBC

8. GPSP PARAMETERS USED FOR POD PROCESSING

This paragraph describes the GPSP antenna phase center and phase law.

The phase center data for the Jason-3 GPS antennas is provided below with reference to the antenna *mechanical* coordinate frame (equipment frame origin) for the two delivered antennas (SN003 and SN006), and for both L1 and L2 GPS frequencies.

Nota : SN003 antenna is linked with GPSPA instrument, and SN006 antenna is linked with GPSPB instrument.

Data was derived from the test reports generated during the OSTM phase center measurement campaign (GPSP antennas are exactly identical on OSTM/JASON-2 and JASON-3).

Along with phase center data, uncertainties are derived in each axis based on the measurements and associated formal error in the estimation process plus mechanical uncertainty.

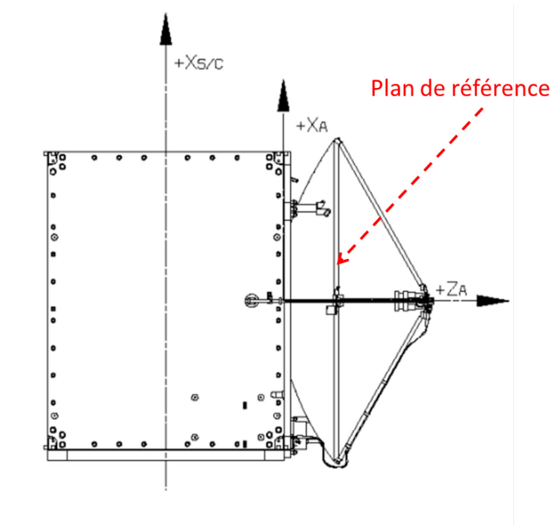
	X [mm]	Y [mm]	Z [mm]
	L1 Phase Data		
SN003	-1.8	0.4	82.1
SN003 Uncertainty	0.8	0.8	0.3
SN006	-2.1	0.3	82.2
SN006 Uncertainty	0.8	0.8	0.3
	L2 Phase Data		
SN003	-0.9	1.0	104.6
SN003 Uncertainty	0.8	0.8	0.3
SN006	-0.8	1.5	104.4
SN006 Uncertainty	0.8	0.8	0.3

The antenna phase centers coordinates for each channel shall be translated into spacecraft reference frame according to the tables given in §6.6.

Regarding the GPSP, the standard POD process uses as measurements the ionosphere-free pseudo-range and ionosphere-free phase combinations (PC~2.54*P1-1.54*P2 for pseudo-range, values in meters, and LC~0.48*L1-0.37*L2, LC in meters and L1,L2 in cycles, rinex notations). Other C1,P1,L1,P2,L2 rinex observables combinations are also used for preprocessing or ambiguity fixing (geometry free, widelane...).

9. POS3B PARAMETERS USED FOR POD PROCESSING

For POS3B antenna phase center, we use a virtual reference plan which is located between the antenna mechanical reference point and the feed.



The coordinates of the antenna reference plan with reference to the antenna *mechanical* coordinate frame is (only Z axis is used) :

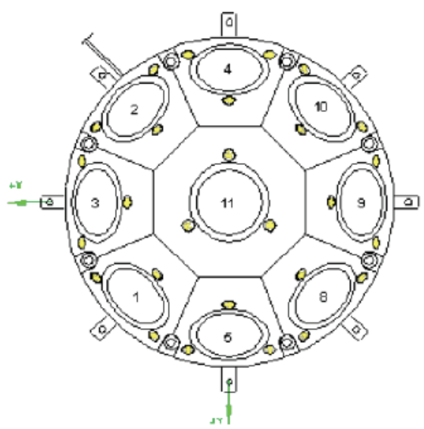
X (mm)	Y (mm)	Z (mm)
0	0	209.302 mm

The reference plan coordinates shall be translated into spacecraft reference frame according to the tables given in §6.6.

10. LRA PARAMETERS USED FOR POD PROCESSING

10.1 LRA OPTICAL CENTER

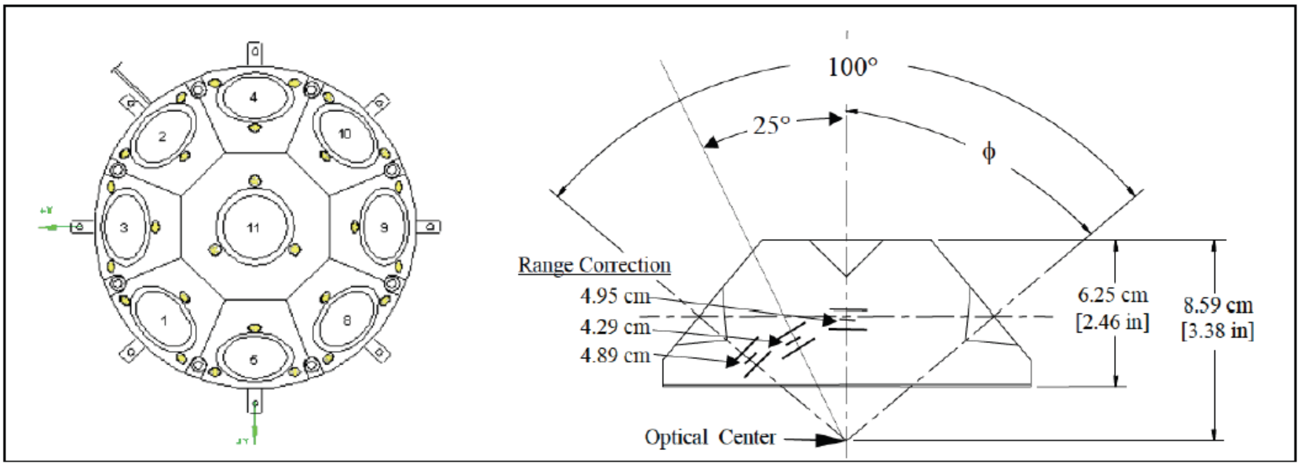
The array is radially symmetrical about its Z axis which is perpendicular to the front face of the center cube. The figure shows the array viewed from nadir on the left with the axes as shown. The numbers in the figure represent the serial numbers of the cubes.
The surface of optical reflection is a 3 dimensional shape approximating a bumpy hemisphere (since the array itself approximates a hemisphere). The table provides the coordinates of the front faces of the cubes



Cube SN	X	Y	Z	θ	ϕ
3	0.06324	0.00000	0.05306	0.00000	0.87266
1	0.04472	0.04472	0.05306	0.78540	0.87266
5	0.00000	0.06324	0.05306	1.57080	0.87266
8	-0.04472	0.04472	0.05306	2.35619	0.87266
9	-0.06324	0.00000	0.05306	3.14159	0.87266
10	-0.04472	-0.04472	0.05306	3.92699	0.87266
4	0.00000	-0.06324	0.05306	4.71239	0.87266
2	0.04472	-0.04472	0.05306	5.49779	0.87266
11	0.00000	0.00000	0.08255	0.00000	0.00000

The optical center coordinates shall be translated into spacecraft reference frame according to the table given in §6.6.

10.2 LRA RANGE CORRECTION



The range correction value is an adjustment to the measured range that will move the point of optical reflection to the optical center of the array. The optical center defined in the figure is the center of a sphere on which the front faces of the retroreflector cubes are tangent. Corrections are also shown in the figure, representing the error window for a given line of sight or incidence angle (θ,φ) on the array.
Adding the range correction to the measured range adjusts the apparent point of reflection to the optical center of the retroreflector array. Alternatively, and if desired, the incidence angle dependence could be removed by using instead an average range correction of 4.6 cm across the entire array.

ANNEX 1 : POS3B ANTENNA MECHANICAL POSITION IN S/C REFERENCE FRAME

For the computation of POS3B antenna Z coordinate in S/C reference frame, the mechanical position given by the instrument ICD is corrected with the antenna shimming results from AIT.

The shimming was done between -1,13 mm and 0 :

Correction calage dans le plan de pose

X	Y	Z
000,00	000,00	000,00 mm
000,00	000,00	001,13 mm
000,00	000,00	000,40 mm
000,00	000,00	000,10 mm
000,00	000,00	000,08 mm
000,00	000,00	000,18 mm
000,00	000,00	001,08 mm
000,00	000,00	001,11 mm
000,00	000,00	001,10 mm
000,00	000,00	000,39 mm
000,00	000,00	000,46 mm
000,00	000,00	000,45 mm

~ Parametres d'usinage de la cale : POS3 complet

Epaisseur de cale à obtenir (e = epaisseur nominale de la cale):

Point 0	**A2**	**Ep = e -1,13 mm**
Point 1	**G7**	**Ep = e**
Point 2	**G1**	**Ep = e -0,73 mm**
Point 3	**A1**	**Ep = e -1,03 mm**
Point 4	**A3**	**Ep = e -1,05 mm**
Point 5	**A4**	**Ep = e -0,95 mm**
Point 6	**G5**	**Ep = e -0,05 mm**
Point 7	**G6**	**Ep = e -0,02 mm**
Point 8	**G8**	**Ep = e -0,02 mm**
Point 9	**G2**	**Ep = e -0,73 mm**
Point 10	**G3**	**Ep = e -0,67 mm**
Point 11	**G4**	**Ep = e -0,67 mm**

so the mean wedge thickness is e-0.5875 mm with e=10 mm + 1.26 mm

The Z coordinate for POS3B antenna mechanical reference point wrt S/C reference point is :

454.42 + 1.26 - 0.59 = 455.09 mm

DOCUMENTATION CHANGE RECORD

Issue.	Rev.	Dates	Pages	Modifications	Visa
PR	0	22/11/11	All	Preliminary issue	
1	0	17/09/2013	All	Update of document (figures from satellite CDR, additional elements about LRA, DORIS AIT tests results).	
1	1	19/02/2015	All	First consolidated issue : satellite figures from SQR, DORIS last AIT results, adding of a chapter about POS3B	
1	2	31/08/2015	All	Precision about altimeter “mispointing maneuvers” (specific “cross-calibration” maneuvers) Precision about the description of satellite surfaces Update of thermo-optical properties tables	

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DESAI S.	*Shailen.d.desai@jpl.nasa.gov*	X
FERNANDEZ D.	*daniel.esteban-fernandez@jpl.nasa.gov*	
FU L.	*llf@jpl.nasa.gov*	
GALLAGHER M.	*michael.p.gallagher@jpl.nasa.gov*	
HADDOX E.	*eric.m.haddox@nasa.gov*	
HAINES B.	*bruce.j.haines@jpl.nasa.gov*	X

DIFFUSION JPL and NASA		
Noms	**mail**	**Ex.**
JAGDMANN J.	*jason.jagdmann@nasa.gov*	
KIM P.	*peter.kim@jpl.nasa.gov*	
LAHAYE N.	*Nicholas.J.Lahaye@jpl.nasa.gov*	
L'HEUREUX K.	*karan.l.lheureux@nasa.gov*	
LINDSTROM E.	*eric.j.lindstrom@nasa.gov*	
MERTZ M.	*mark.a.mertz@nasa.gov*	
NEECK S.	*Steven.neeck@nasa.gov*	
OSWALD J.	*john.e.oswald@jpl.nasa.gov*	
OYAKE A.	*amalaye.oyake@jpl.nasa.gov*	
PILOTO A.	*armando.piloto@nasa.gov*	
PIQUERO J.	*jorge.l.piquero@nasa.gov*	
SAVINELL C.	*Christopher.savinell@nasa.gov*	
SEYBOLD Calina	*calina.c.seybold@jpl.nasa.gov*	
VAZE P.	*Parag.V.Vaze@jpl.nasa.gov*	
WILLIS J.	*joshua.k.willis@jpl.nasa.gov*	

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