TEAM ORANGE

ADONIS

Atmospheric Drag, Occultation 'N' Ionospheric Scintillation Mission

Summer School Alpbach 2013

Space Weather:
Science, Missions and Systems

July 16-25, Alpbach/Tyrol, Austria

Outline

- **Mission Statement & Objectives**
- **Mission Requirements**
- **Instrumentation**
- **Orbit**
- **Space Segment**
- **Ground Segment & Operations**
- **Development, Cost & Risks**
- **Disposal**
- **Summary**

Mission Statement & Objectives

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Mission Statement

The mission goal is to study the **dynamics of the thermosphere and ionosphere** over a **full solar cycle (i.e. 11 years)** in LEO.

Mission objectives

- Build a model for **satellite drag** in relation to Space WEather (SWE).
- Provide data to improve **ionospheric models.**
- Provide **near real-time Total Electron Content (TEC) data** in the northern polar region.
- Contribute to models for **radio communication perturbation** in relation to SWE.
- Operate for a **full solar cycle (11 years)** to gather significantly improved statistics.

Problem Statement

Why are thermosphere and ionosphere important?

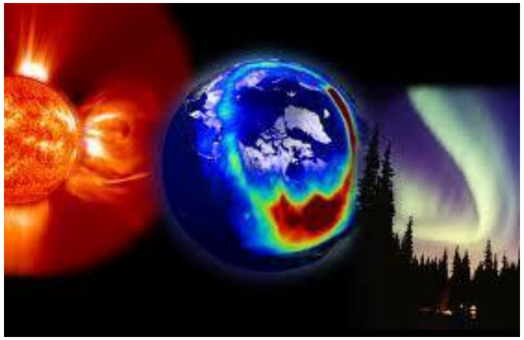
= the atmospheric layer where we experience space weather (= SWE)

Incomplete understanding of Solar activity and SWE effects on the terrestrial thermosphere and ionosphere

Ionospheric space weather causes:

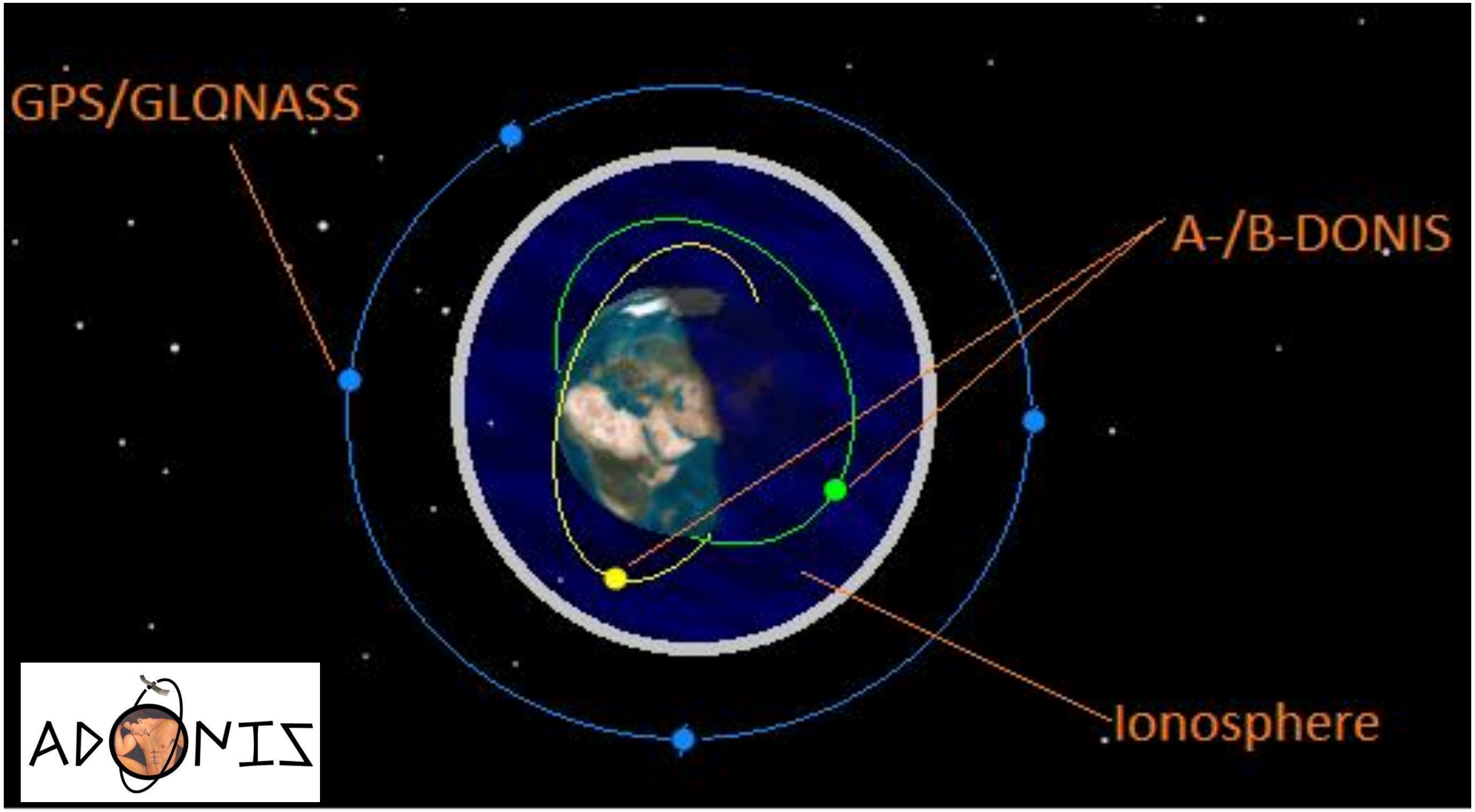
1. **Satellite drag**: better description needed for launch planning, prediction of uncontrolled re-entries, control of LEO s/c
2. **Disturbances** related to variations in Total Electron Content (TEC):
   a. in critical **HF communications** during polar flights, emergency relief, military operations
   b. in **GNSS signals**
   c. in **satellite telecommunication and broadcasting**

   ⟶ **Safety, commercial and strategic impacts**



Source: Worldpress

Overview of the ADONIS Mission



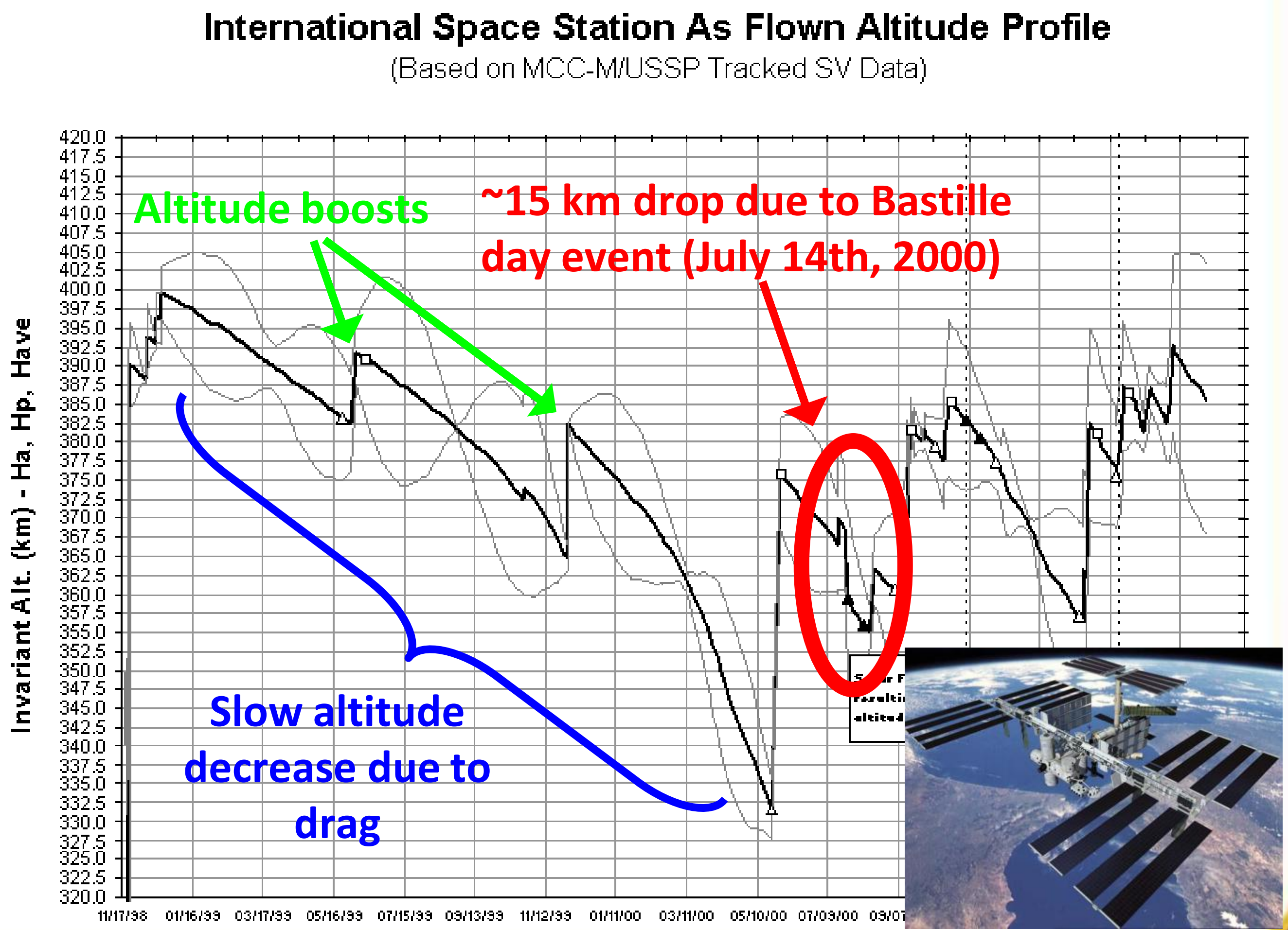
Ionosphere	GPS	GLONASS	A-/B-DONIS	LEO
100 - 1000 km	20 200 km	19 100 km	300 - 800 km	160 - 2000 km

Mission Science Goal I

Improve the quality and reliability of current **satellite drag models** by measuring in-situ both the drag and the parameters relevant to model it.

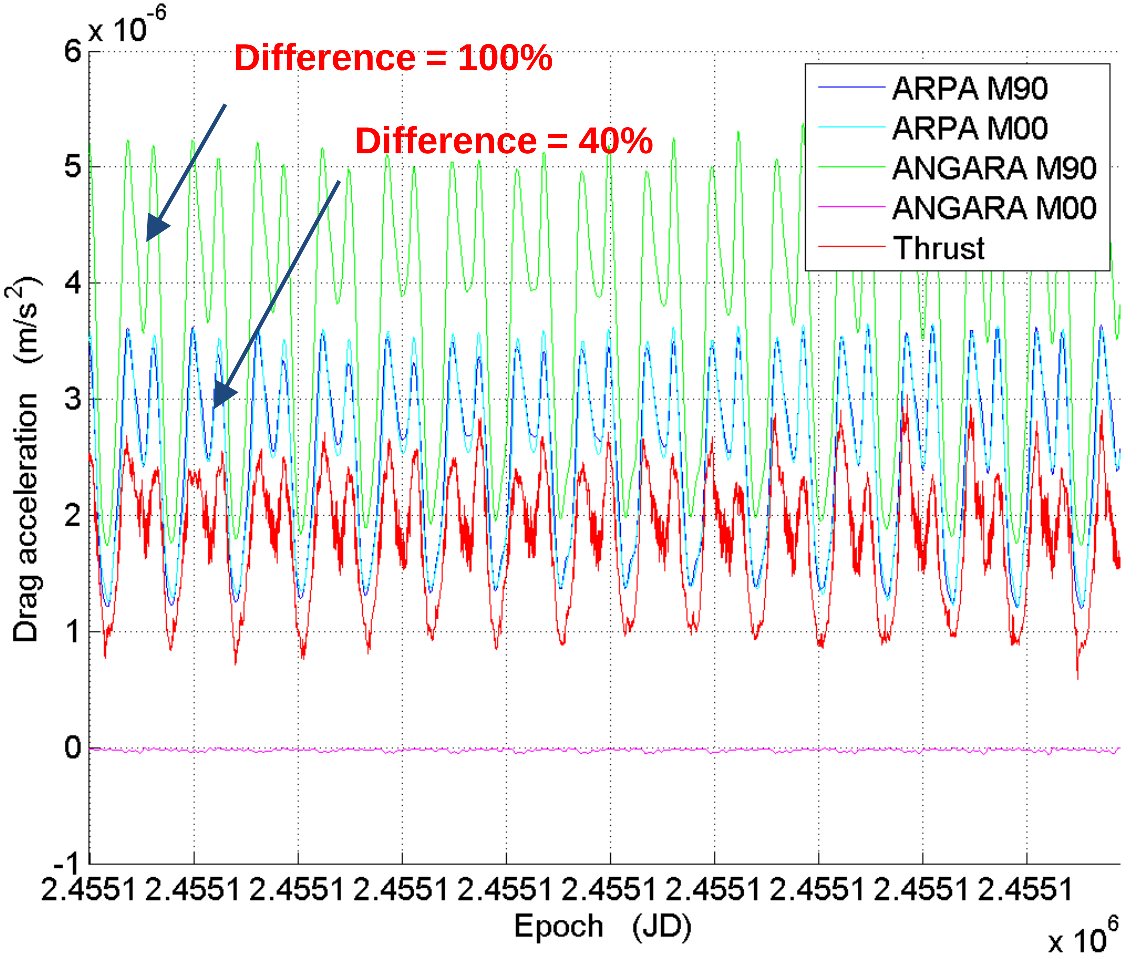
- ISS altitude from 11/1998 to 5/2001
- Drag causes constant dropping of altitude
- Altitude boosts needed
- Note drop due to Bastille day event

Source: Chammons 2001



Mission Science Goal I

- GOCE thrust compared to drag modelling
- Thrust is compensating directly the drag
- Note the difference between accelerations computed and observed

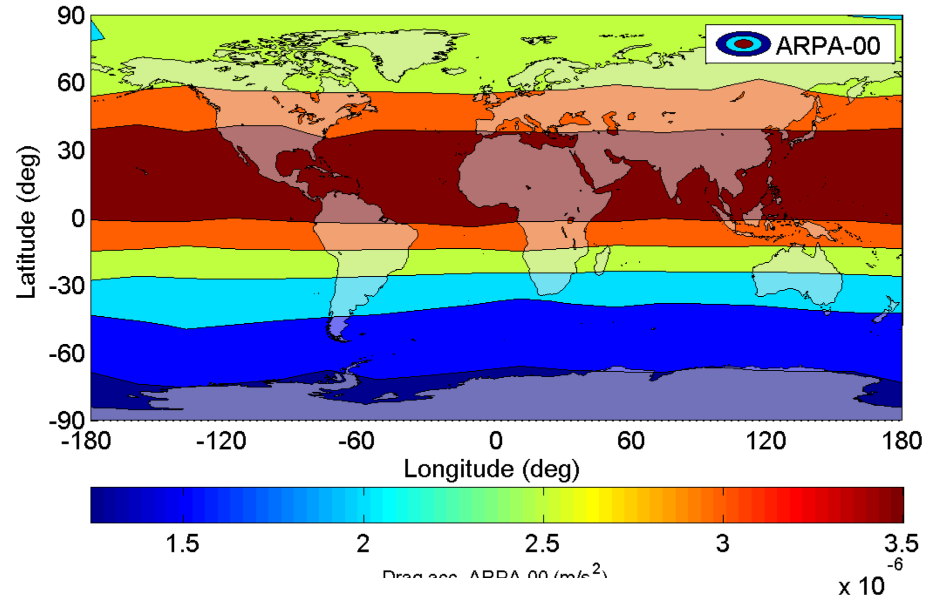


Source: F. Gini, 2013

Mission Science Goal I

Necessary parameters for derivation of the drag:

$$a = \frac{\rho A v^2}{2m} c_d(T_0, T_S, m_p, n_p)$$



T_0: atmospheric temperature

m_p: average mass of the particles

n_p: particle density

ρ: atmospheric density

v: velocity of the satellite with respect to the atmosphere

a: drag acceleration

A: cross sectional area

m: mass of the satellite

c_d: drag coefficient

T_S: spacecraft temperature

Mission Science Goal I

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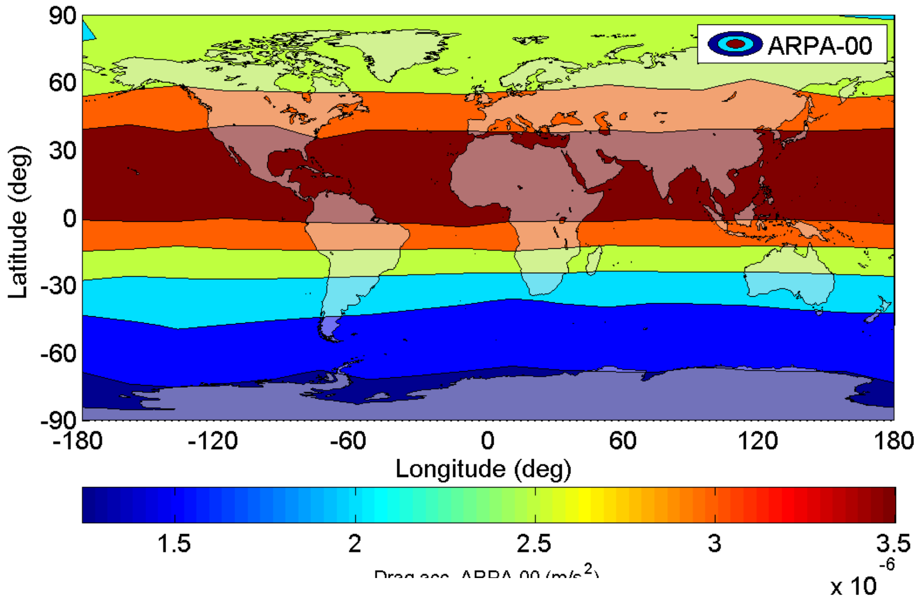


- → T_0: atmospheric temperature
- → m_p: average mass of the particles
- → n_p: particle density
- → ρ: atmospheric density
- → v: velocity of the satellite with respect to the atmosphere
- → a: drag acceleration
- A: cross sectional area
- m: mass of the satellite
- c_d: drag coefficient
- → T_S: spacecraft temperature

Instruments: Spectrometer, Thermistor, Accelerometer, Langmuir Probe, Particle Analyser

Mission Benefits I

Related missions: GOCE, QB50

Benefits

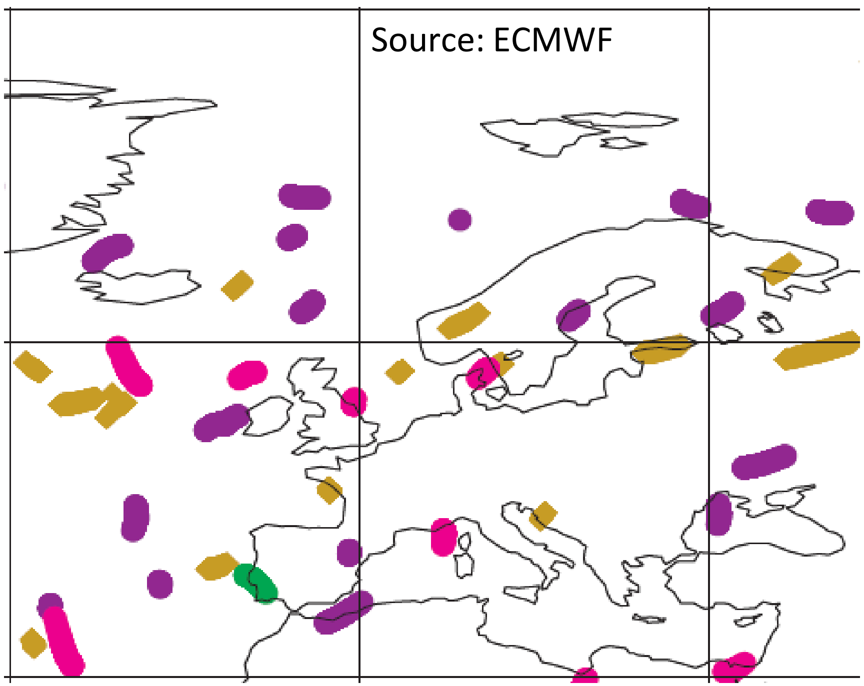
- **Observe different altitude** and **wider altitude range** than GOCE
- Measure all the parameters important for **drag in-situ**
- **Connect SWE** with drag
- Orbit design allows the study of **daily to seasonal variations**
- Long duration: large statistics for different SWE
- Continue the drag **measurements during de-orbiting** and end of mission

Mission Science Goal II

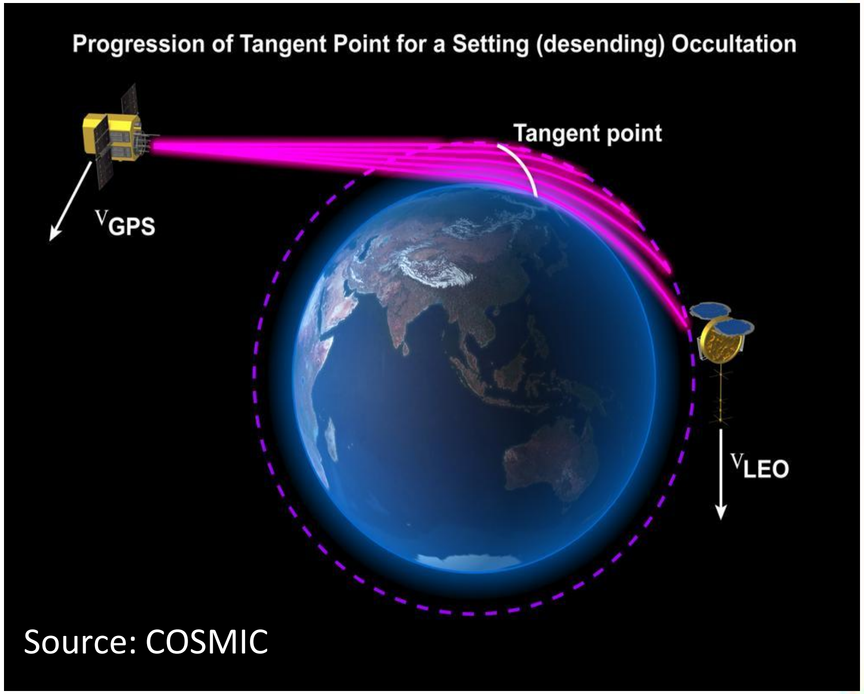
Provide additional Total Electron Content (TEC) data for ionospheric monitoring.

Radio occultation (RO)

- Doppler shift of **GNSS signal** measured on LEO polar S/C provides more **high-resolution vertical** profiles of the electron density above 80 km of **regions with sparse coverage.**
- **Near Real Time (NRT) monitoring of Arctic region** (NRT downlink to Svalbard GS)



Geograph. coverage of 6 h RO data over Europe – ROM SAF

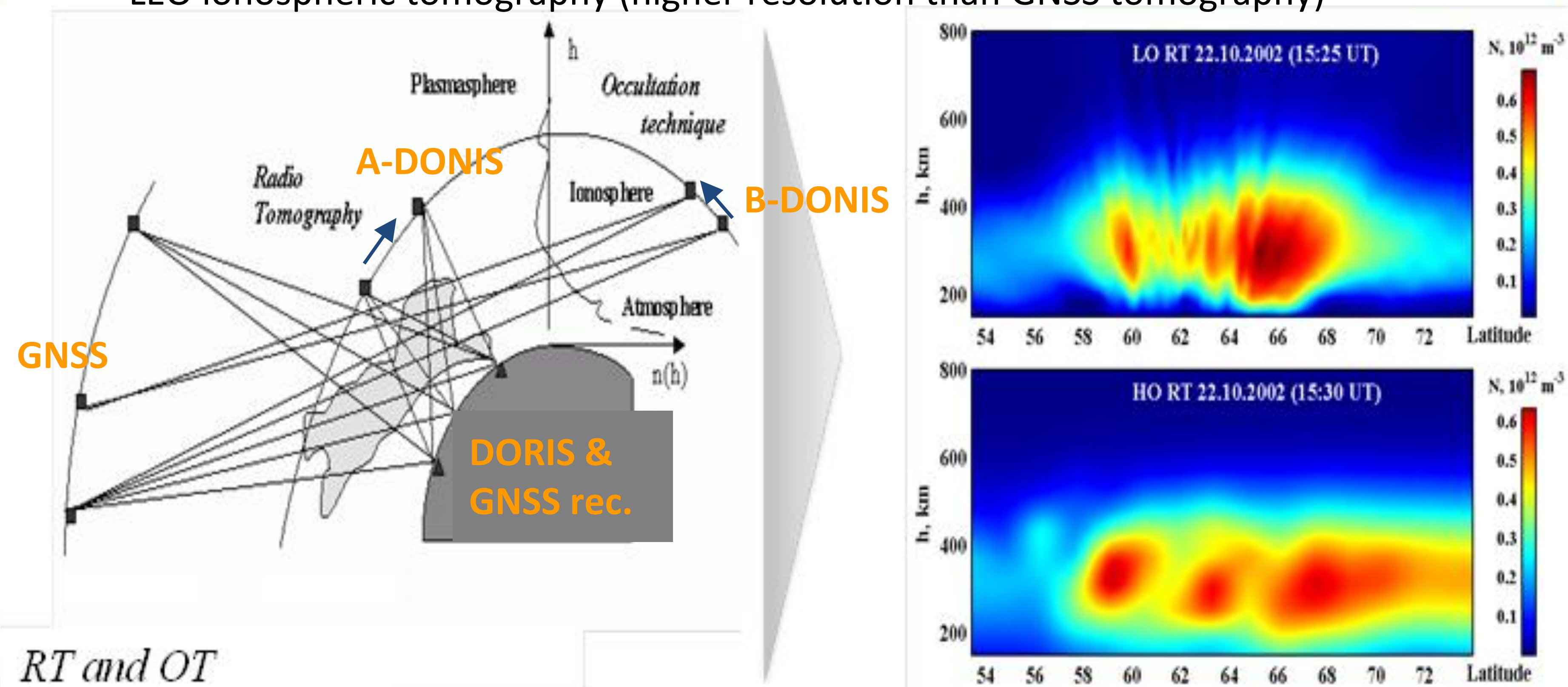


GNSS Radio Occultation principle

Mission Science Goal II

Ionospheric tomography with scintillation and occultation combined

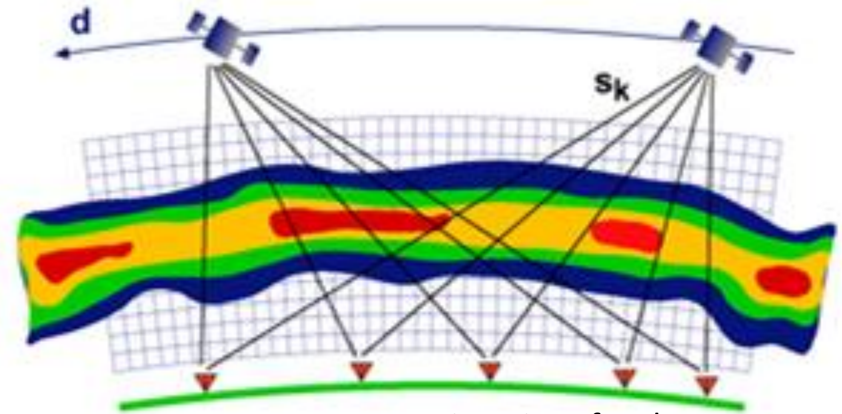
- Scintillation from **10 LEO S/C** and **56 DORIS GS beacons** (global: lat. 70°S – 80°N) interpreting ionospheric irregularities ranging from a few up to hundreds of meters
- LEO ionospheric tomography (higher resolution than GNSS tomography)



Source: Rekenthaler et al., 2013

Mission Benefits II

Related missions: COSMIC, CHAMP



Source: SGO, University of Oulu

Benefits:

- The **only available** TEC with scintillation from orbital **S/C - S/C links:**
  in average 20 contacts per day per S/C (20–40 min each, up to 200 Hz)
- Scintillation from 56 DORIS ground stations **no S/C currently provides**
- Hi-res LEO tomog., use **high ratio of frequencies**: 5.1, 2.6 (GPS use 1.3)
- 1k occultations per day in average (250 events per S/C per GNSS system)
- **3D** global TEC maps, **NRT data in the arctic region**

Mission Requirements

Mission Requirements

M.1.	Perform measurements related to S/C **drag coefficient** at LEO
M.2.	Ionospheric tomography from LEO based on **radio occultation and scintillation**
O.1.	Obtained results shall be delivered to **scientific community**
O.2.	Obtained occultation results shall enable provision of **near real-time service**
O.3.	Duration of mission shall cover **one full solar cycle** (i.e. 11 yrs)
O.4.	The S/C shall keep the same cross area **normal to the velocity vector**
D.1.	S/C shall fit into **Vega** fairing
D.2.	**On-board components** shall not interfere with each other
D.3.	S/C shall provide on-board **payload instruments** to meet the mission objectives
D.4.	GS shall provide **uplink and downlink capability**
P.1.	The **acceleration** shall be determined at least once per second

M: Mission goal; O: Operational req; D: Design req; P: Performance req.

Science Requirements

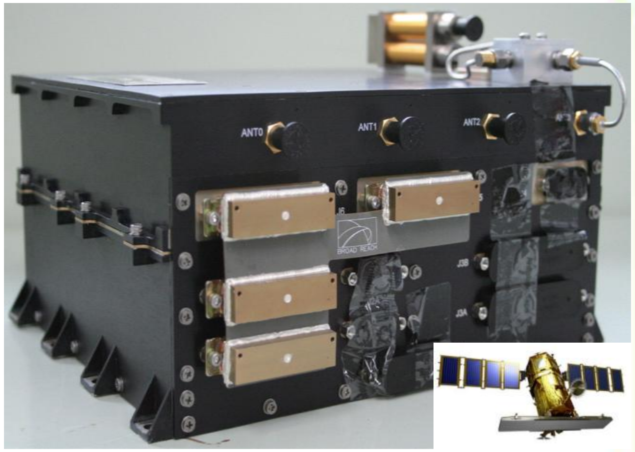
Requirements	Range/Sensitivity	Instrument
Particle composition	0–50 amu, 128 bins, 1 Hz NRT	Ion & Neutral Mass Spectrometer (**INMS**)
On-board temperature	1°C accuracy, 1 Hz	Thermistors
S/C acceleration	10^{-8} m/s², 1 Hz	Italian Spring Accelerometer (**ISA**)
Plasma velocity, temp.	10 eV–30 keV, 1 Hz NRT	3D Particle Analyser (**3DPA**)
Plasma density	10^{9}–10^{12}/m³, 1 Hz NRT	Langmuir Probe (**m-NLP**)
TEC from RO	1k RO events, 0.001 TECU rel. (3 TECU abs.), 10Hz	Radio occultation instr. **(IGOR+)**
Ionospheric scintillation	0.003 TECU rel. (1 TECU abs.) 10Hz S4	Radio tomography receiver (**CITRIS**)
Magnetic field	+/-80 µT, 0.5 nT, 0.1 Hz NRT	Flux Gate Magnetometer (**FGM**)

Drag	Ionospheric/thermospheric dynamics	Both

All TRL 9 except 3DPA (will fly in Dec. 2013)

Instrumentation

Payload: Occultation and Scintillation



IGOR+ radio occultation instrument:

- RO of 2 GNSS (GPS, GLONASS)
- 5 kg, 22 W, 21.8×24.0×14.4 cm³
- Flight heritage, modified for KOMPSAT-5 (2013)
  Tri-G (+ GALILEO) TBC follow-up mission (50 W req.)

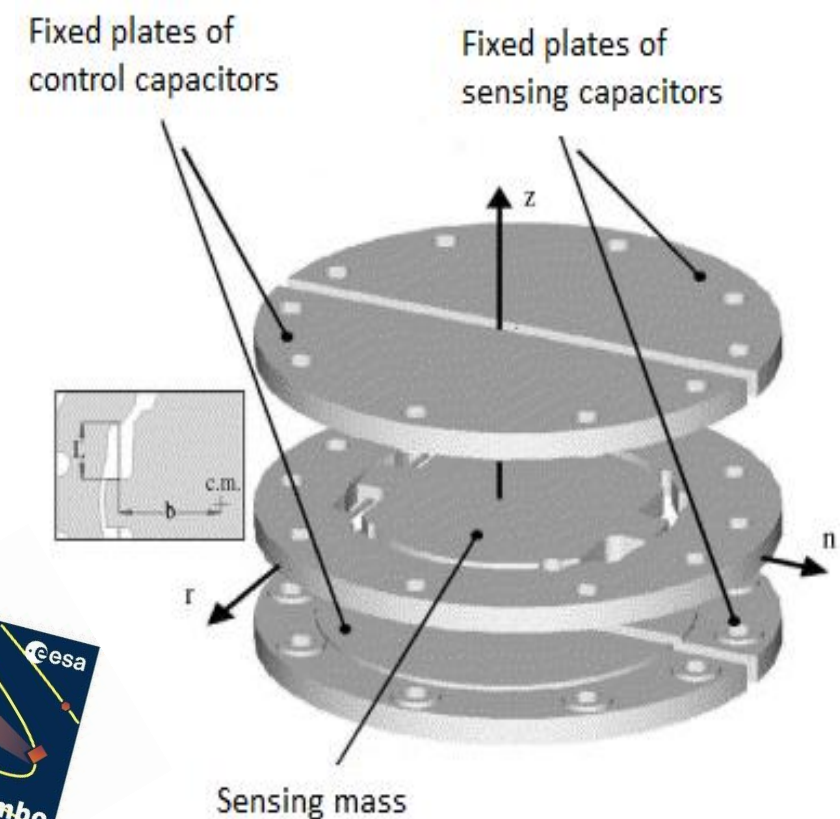
CITRIS ionospheric tomography instrument:

- Reconstruction of scintillation parameters
- Receiving 56 DORIS GS @ 401, 2036 MHz
- 10 satellite transmitter – sat. receiver links
  @150, 400, and 1067 MHz
- Sampling up to 200 Hz
- 4.5 kg, 12.3 W, 40.0×31.0×12.0 cm³
- Flown on STPSAT1 (3/2007 – 10/2009)

Both only receivers, no RF interference with plasma instrument suite

Italian Spring Accelerometer (ISA)

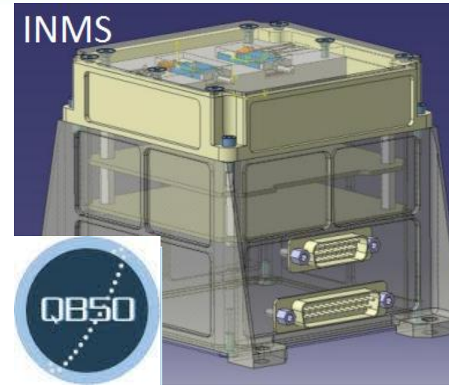
- Planned for BepiColombo
- **Full acceleration vector** of the satellite
- Sensitivity of **~10^{-8} m/s²**
- Modification needed in order to measure at higher frequencies (at 1 Hz)



Payload: INMS, 3DPA, m-NLP, FGM

Ion and Neutral Mass Spectrometer (INMS)

- Resolves ions 0.1 - 28 eV and neutrals O, O_2, N_2
- QB50 planned

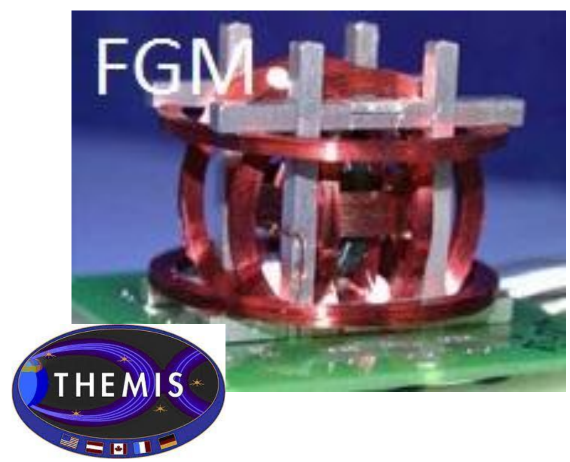


3D Particle Analyser (3DPA)

- Resolves ions and electrons few eV–30 keV
- Two detectors on top and bottom of satellite
- Test flight on sounding rocket mission ICI-4, Dec. 2013

Multi-Needle Langmuir Probe (m-NLP)

- 0.6 m booms, 3 probes
- Flown on CubeSTAR, QB50 planned



Flux Gate Magnetometer (FGM)

- +/-80 µT, resolution 0.5 nT
- Boom-mounted (1 m in front)
- Flown on Cluster, THEMIS, etc.

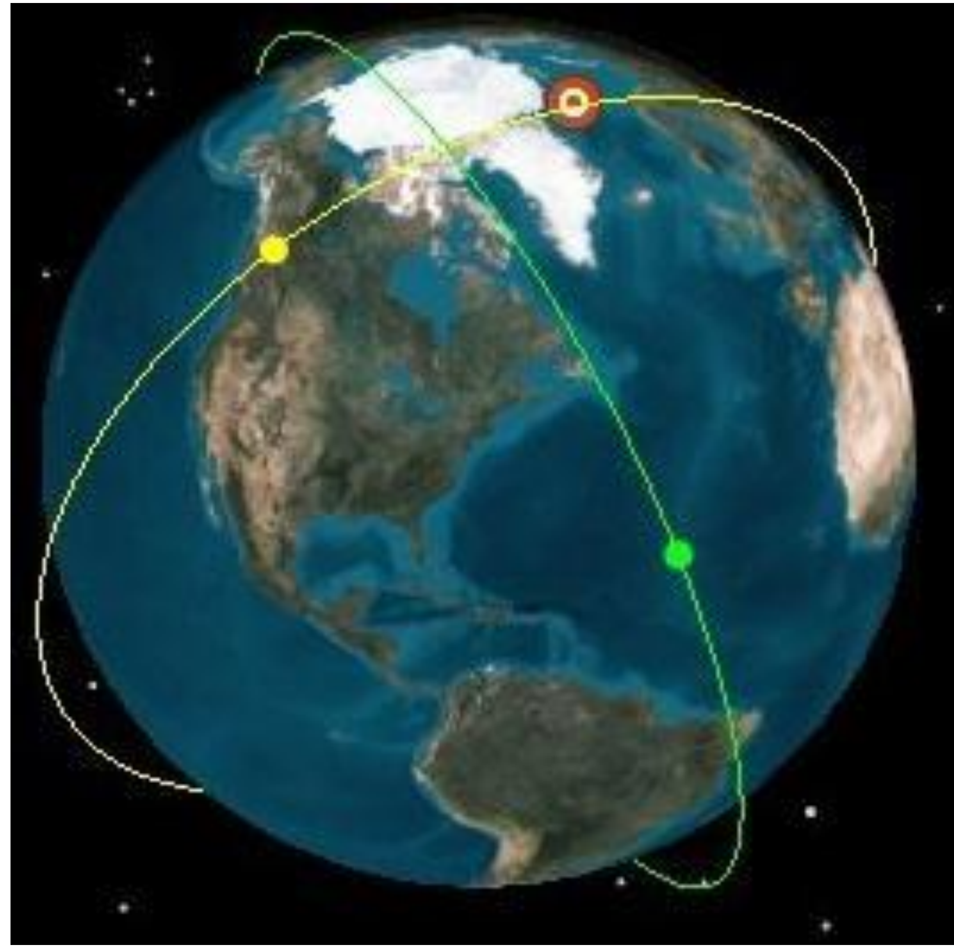
Orbit

Orbit

Final constellation:

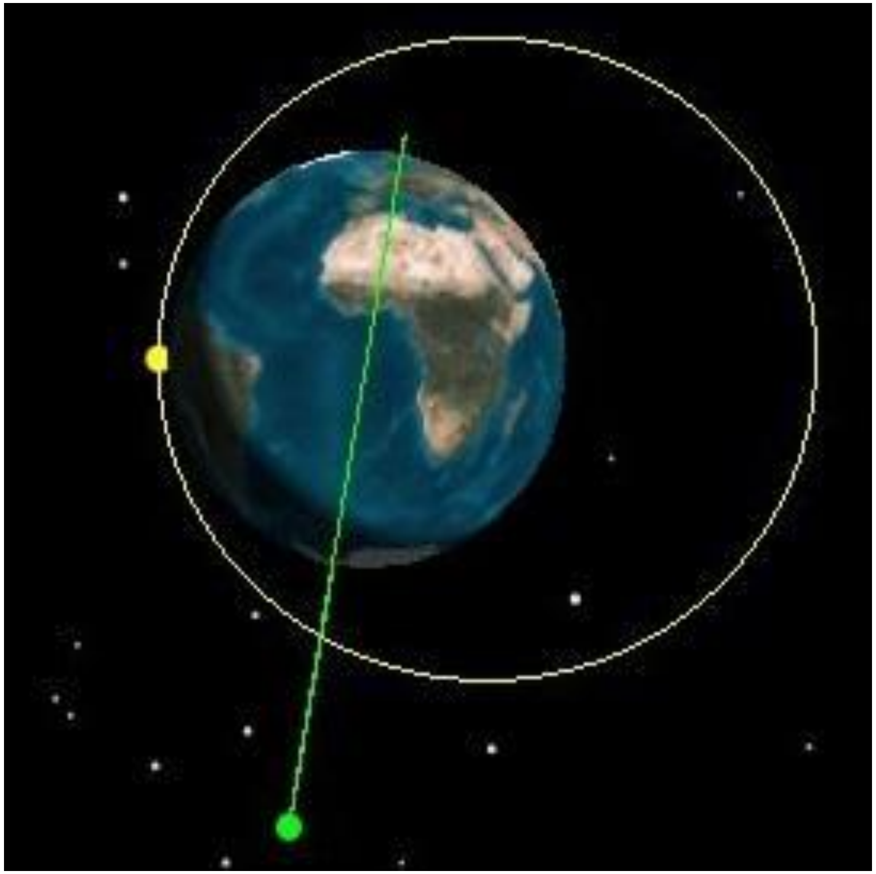
- **2 elliptical orbits** with apogee altitude 800 km, perigee altitude 300 km and **80° inclination**.

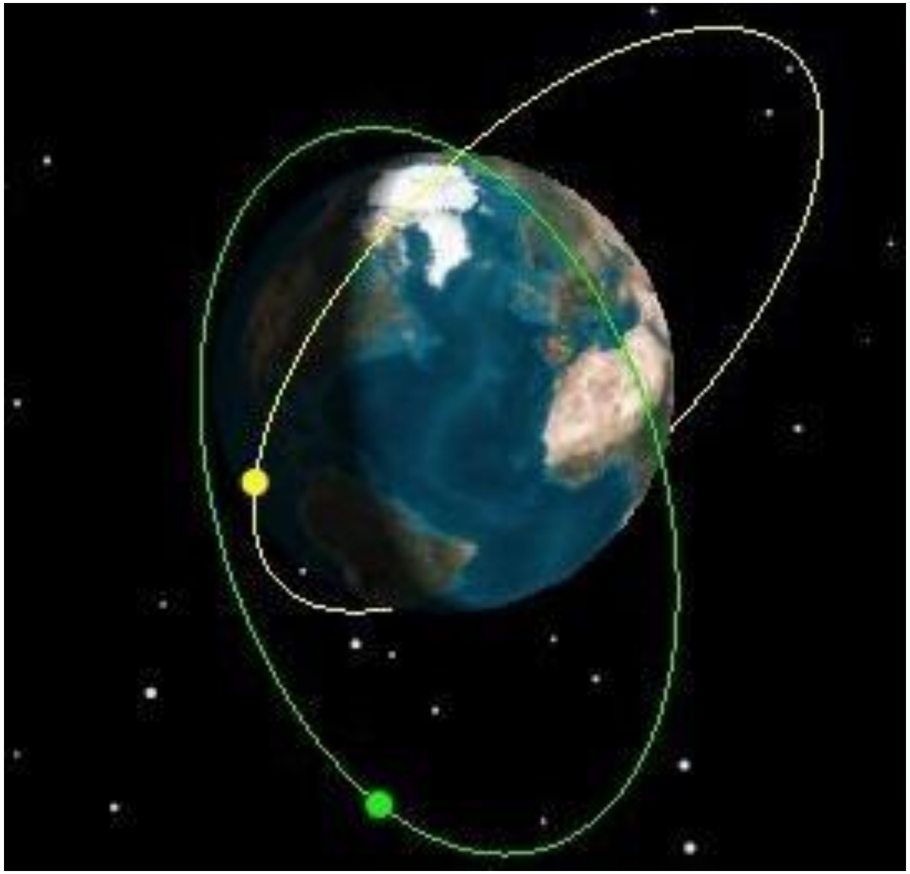
- **90° difference** in the plane of the two orbits (Right Ascension of the Ascending Node – RAAN) and 90° difference in argument of perigee.



→ Space and time resolution, different local times at passage.

Amplified View





Perigee Determination

$$a = \frac{\rho A v^2}{2m} c_d(T_0, T_S, m_p, n_p)$$

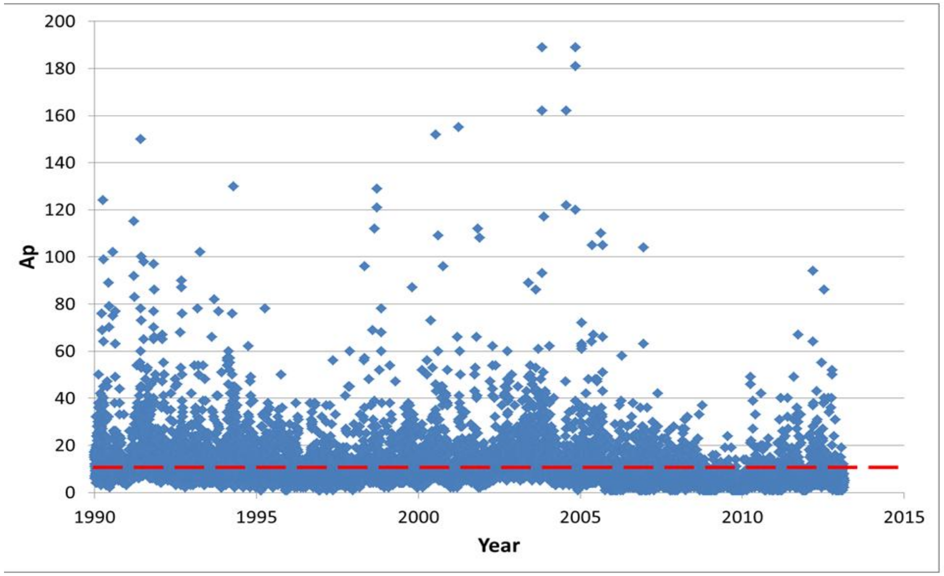
The main driver of satellite drag is
air density

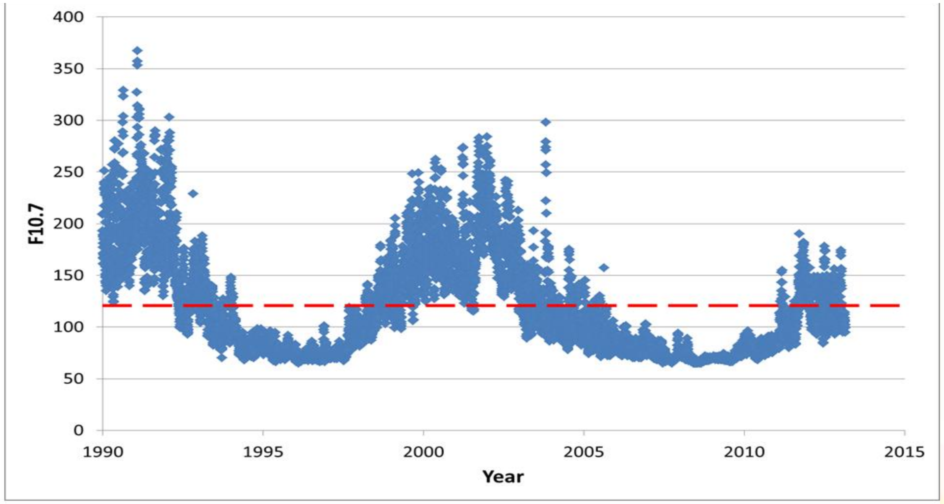
solar activity

For the following computations
an **overestimated average solar activity** was selected, and **NRLMSIS-00** was used as atmospheric model.

Ap = 11.8
F10.7 = 118.2E22 W/m²Hz

For elliptical orbits the drag exponentially increases with decreasing altitude of the perigee.



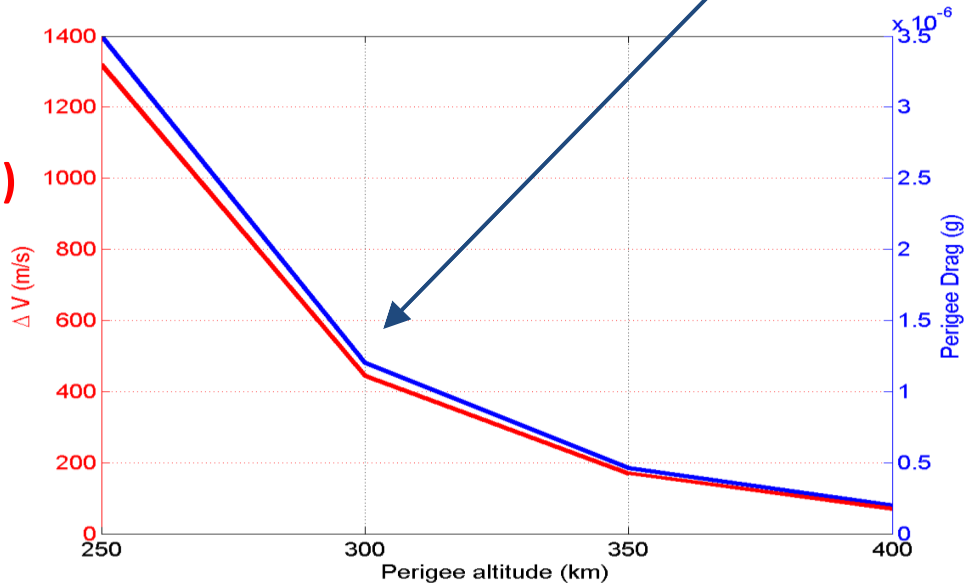


Perigee Determination

h apogee	h perigee	Drag acceleration (maximum value)	Drag ΔV (after 11 years)	m prop (Isp = 300 s)
(km)	(km)	(g)	(m/s)	(kg)
800	400	1.75E-7	80	10
800	350	4.25E-7	185	20
800	**300**	**1.11E-6**	**480**	**45**
800	250	3.30E-6	1400	150

Selected solution

Total ΔV due to drag is computed to design the orbit (perigee)

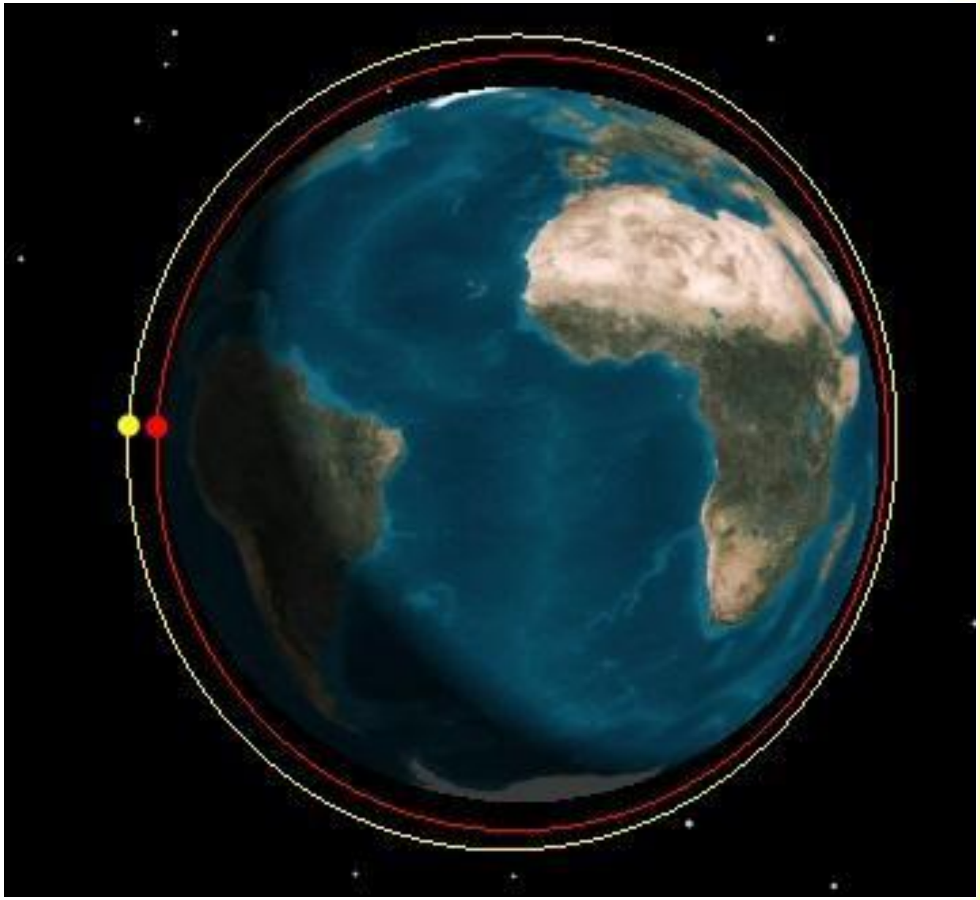


Orbit

Constellation build up:

- Begin with 2 satellites in 800×300 km, 80° inclination orbit.
- A-DONIS: **ΔV = 0.14 km/s** to make one orbit **circular (800 km),** with same inclination.
- B-DONIS: **ΔV = 0.09 km/s** to make one orbit **elliptical (300×500 km)**, with same inclination.

→ provides different precession rates of RAAN

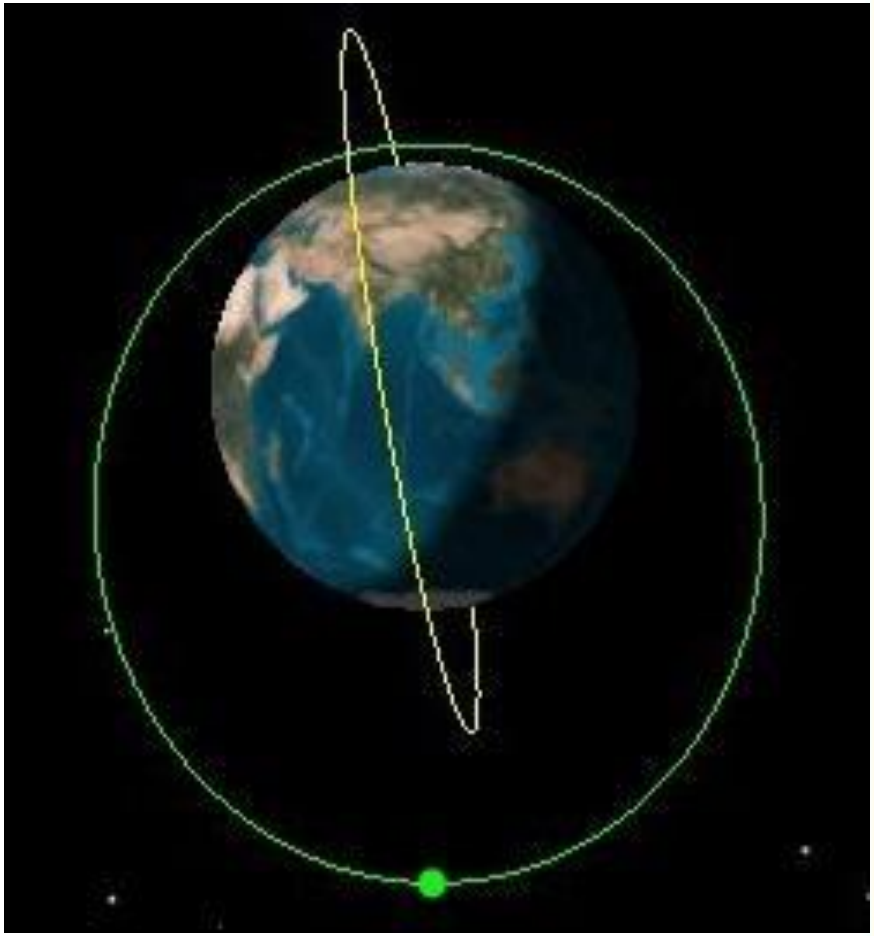


Start constellation

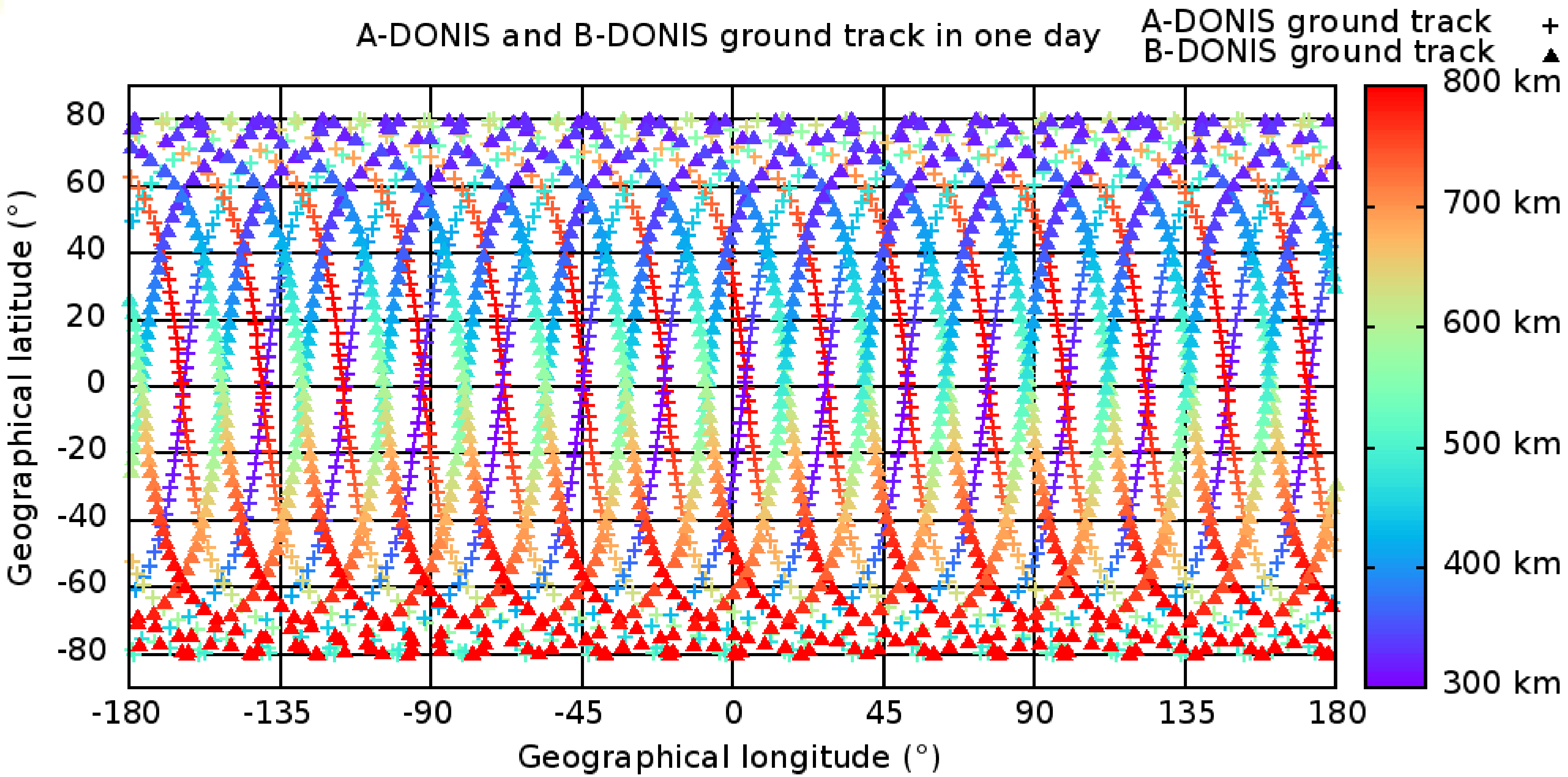
Orbit

- Relative precession rate of ascending node: 0.255°/day
- Difference in RAAN becomes equal to 90° (340 days after launch)

  → apply the same ΔV (**apogee kick burn**) to make the two orbits identical, but with **different orbital planes**



Ground Track

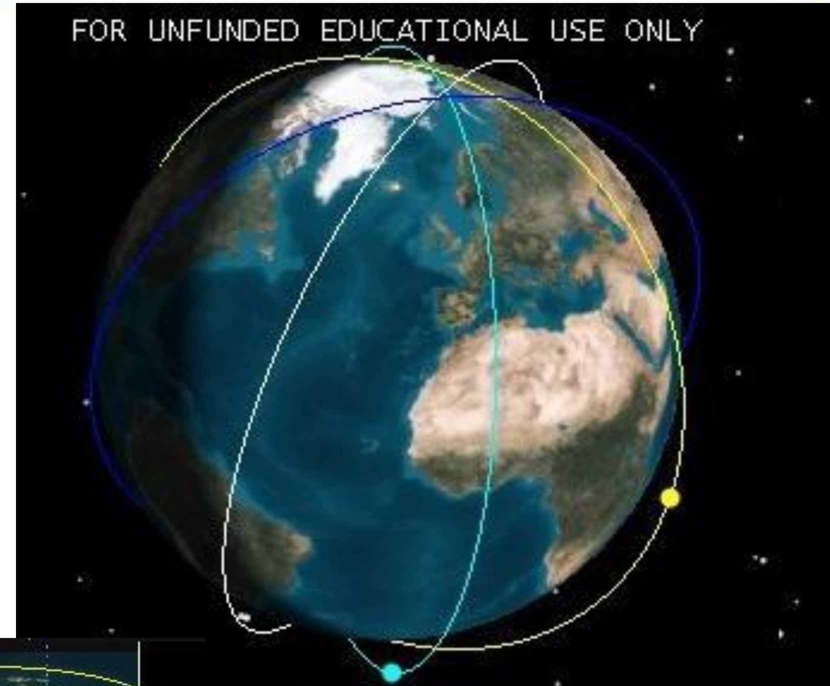


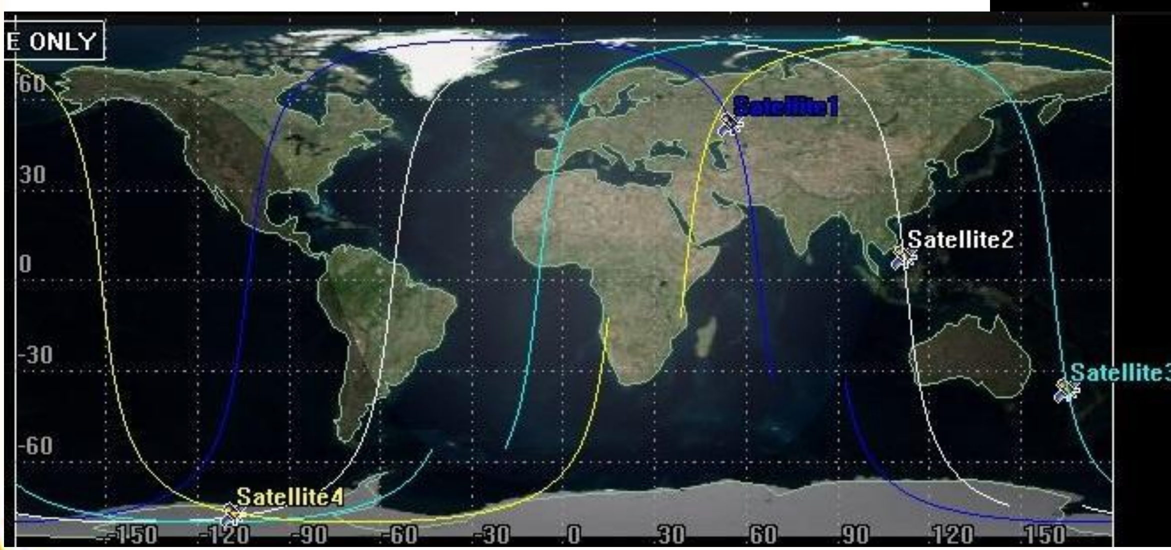
Improvement of spatial sampling via:

- **90° difference in the argument of perigee**
- **Temporal changes in the orientation of the perigee at same precession rates for both satellites**

Orbit

The **mission can be expanded** by increasing the coverage with two more satellites (45° difference between the planes).





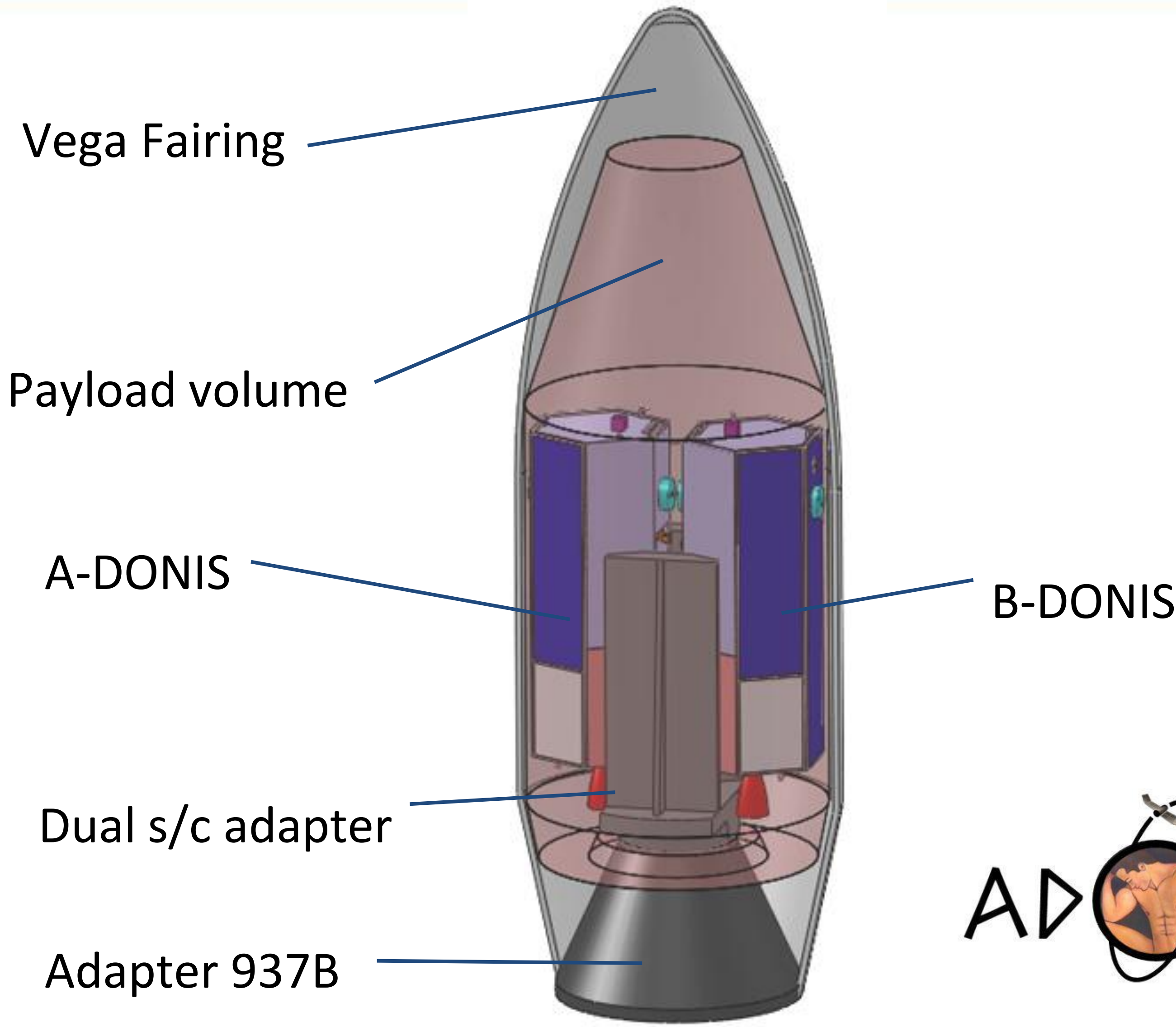
Space Segment

Launcher: Vega

Arianespace Vega

- Cost: **35 M€**
- Liftoff mass: **137 tonnes**
- Payload: **1.5 tonnes** to 800 km altitude
- Mission with **2 s/c**
- Launch site: Guiana Space Centre, **Kourou**, French Guiana

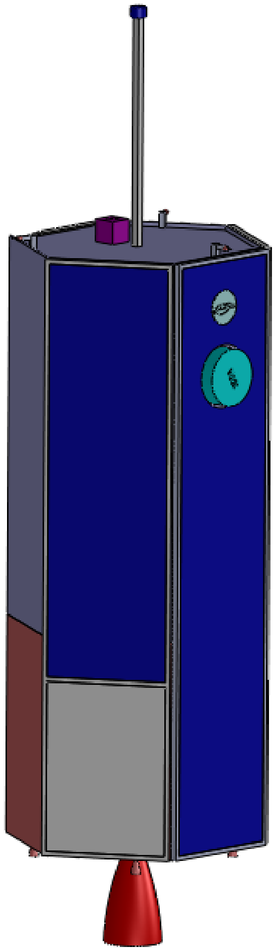
Configuration of S/C in VEGA Fairing



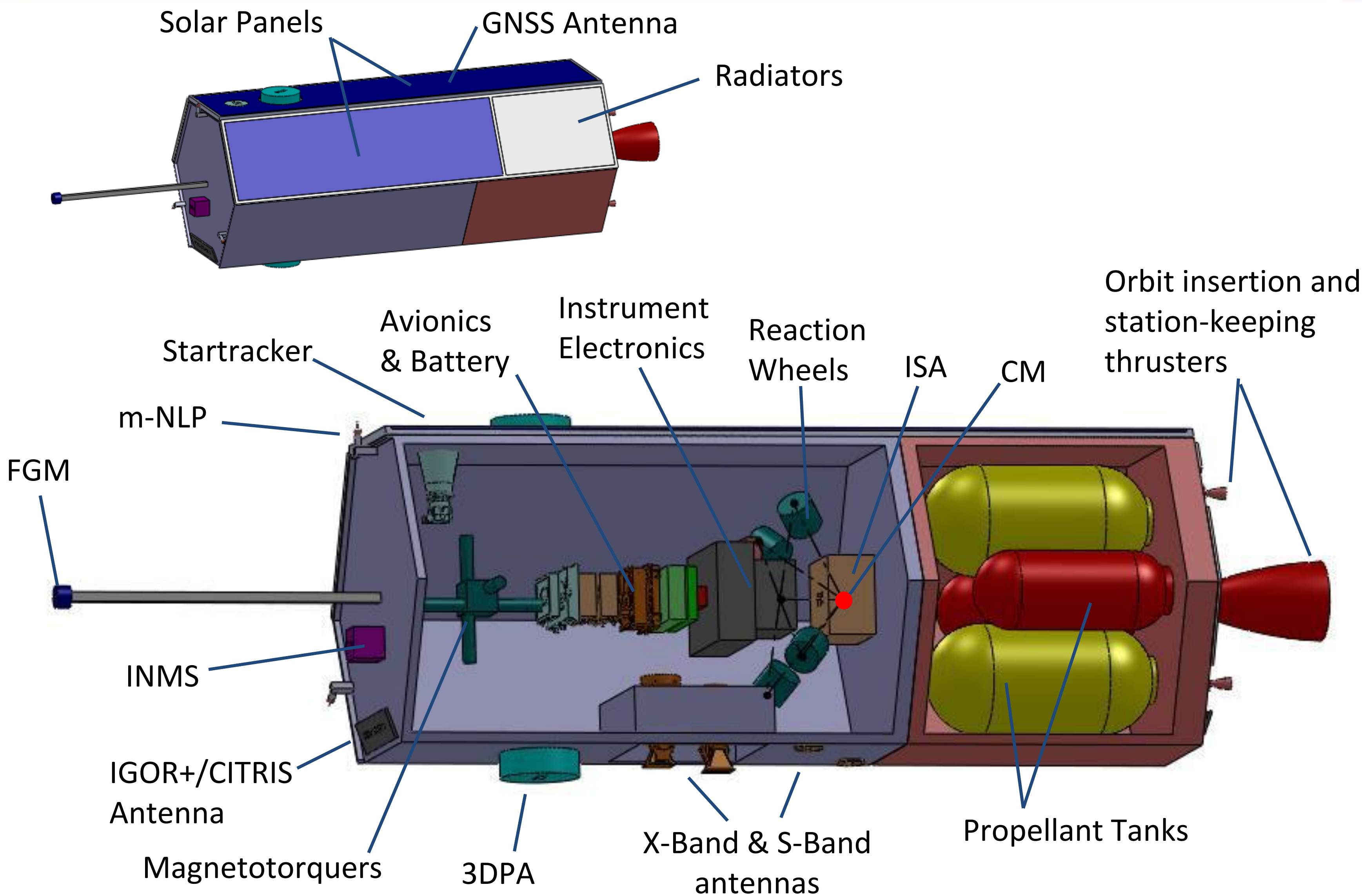
Overview

Based on **commercial Surrey SSTL-300 platform,** customised to meet mission requirements:

- **Small frontal area** and **simple shape**
- **Propulsion** for orbit insertion and station-keeping
- Increased lifetime for **full solar-cycle** coverage
- **Lightweight structure** to meet payload capacity of Vega launcher



Overview


Solar Panels
GNSS Antenna
Radiators
Startracker
Avionics
& Battery
Instrument
Electronics
Reaction
Wheels
ISA
CM
Orbit insertion and
station-keeping
thrusters
m-NLP
FGM
INMS
IGOR+/CITRIS
Antenna
Magnetotorquers
3DPA
X-Band & S-Band
antennas
Propellant Tanks

Spacecraft Product Tree

Space Segment

Structure
- Aluminum and aluminum-skinned honeycomb panels

Thermal
- Radiators
- Multilayer Insulation
- Peltier Coders
- Thermal Heaters

AOCS & Propulsion
- Reaction wheels
- Magneto-torquers
- Laser gyro
- Orbit control
- Star tracker
- Sun sensors
- Magneto -meters
- Bipropellant thrusters

Avionics
- On Board Computer 750
- High Speed Data Recorder
- Power System 150
- CAN BUS
- Propulsion Control

Power
- GaAs cells solar panels
- Li-ion cells battery

Commu-nications
- S-Band receiver
- S-Band transmitter
- S-Band Patch Antenna

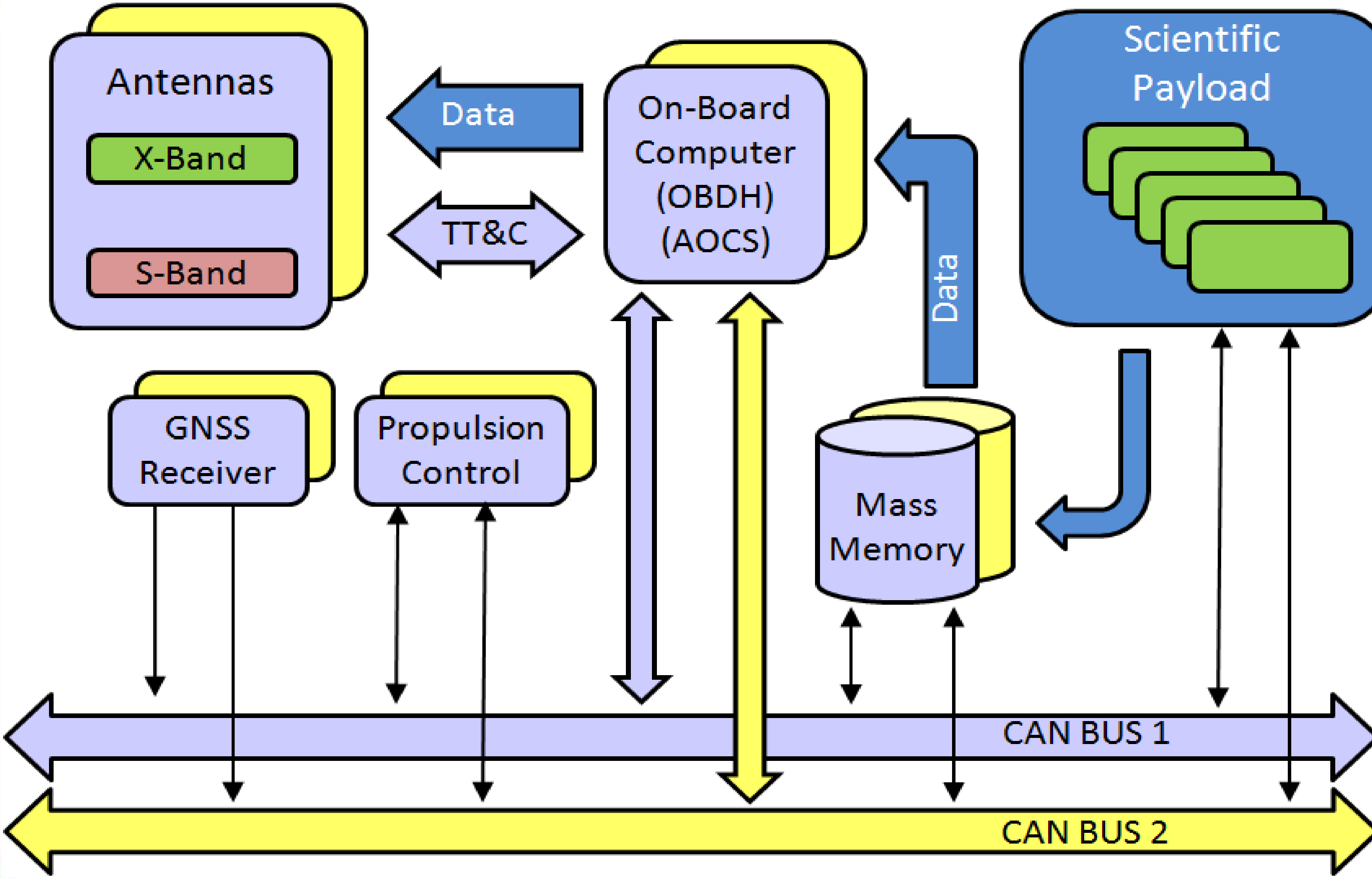
- X-Band science data downlink
- XTX 400 Transmitter
- High Gain Pointing Antenna

Scientific Payload
- INMS Ion & Neutral composition
- 3DPA Plasma Analyser
- m-NLP langmuir probe
- Thermistors surface temperature
- IGOR Radio Occultation
- ISA Italian Spring Accelerometer
- FGM Flux Gate Magnetometer

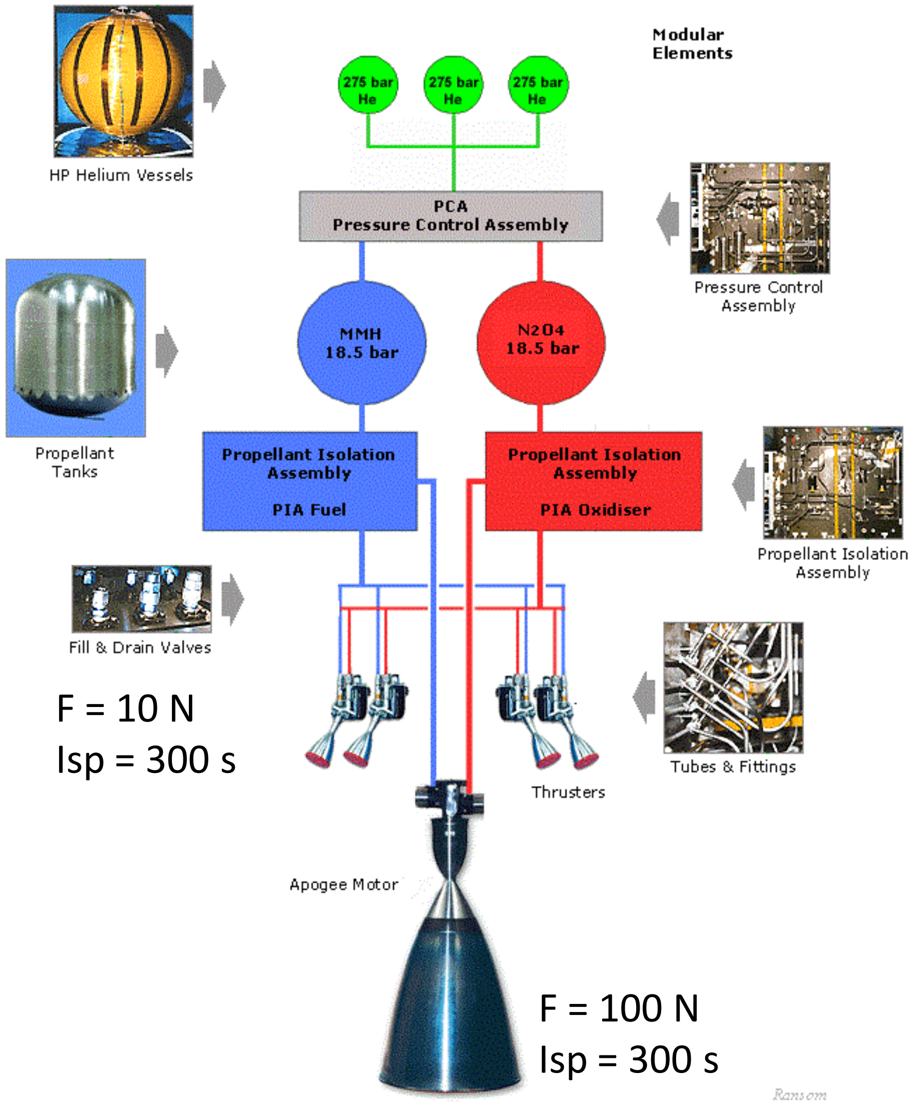
SSTL-300 modified BUS

ADONIS

Spacecraft Avionics Architecture



Propulsion System



F = 10 N
Isp = 300 s

F = 100 N
Isp = 300 s

Maneuver	A-DONIS	B-DONIS
Orbital injection	ΔV = 2×140 m/s → **58 kg** (25% margin)	ΔV = 2×93 m/s → **32 kg** (25% margin)
Temporary orbit correction	ΔV = 5 m/s → **2 kg** (50% margin)	ΔV = 100 m/s → **28 kg** (25% margin)
Elliptical orbit correction	ΔV = 500 m/s → **75 Kg** (40% margin)	ΔV = 500 m/s → **75 Kg** (40% margin)
Avoidance maneuvers	**15 kg**	**15 kg**
Sum	**150 kg**	**150 kg**

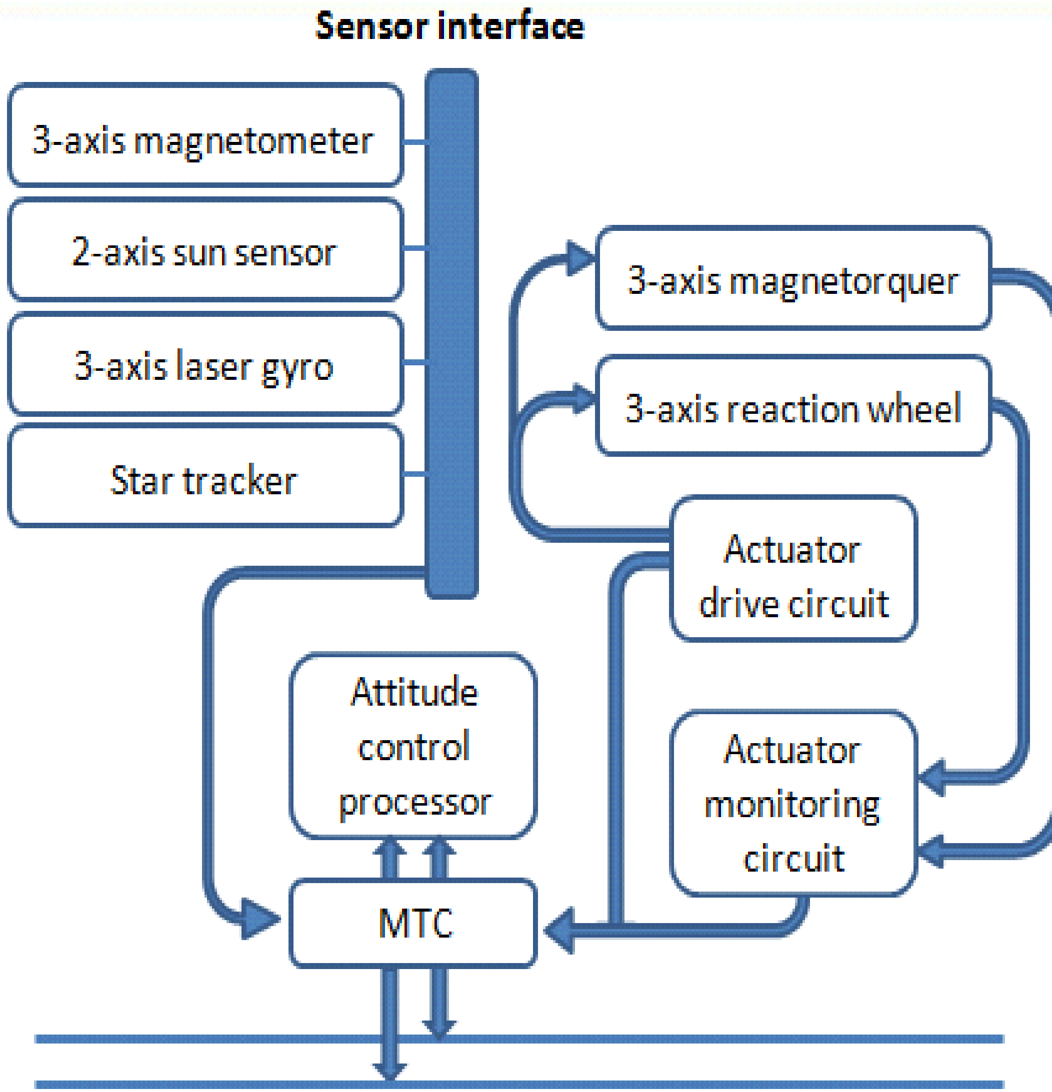
Attitude Control

Attitude control

- Reaction wheels (4 wheels, 0.02 Nm)
- Desaturation with magnetorquers (3 rods,110 Am^2)

Attitude determination

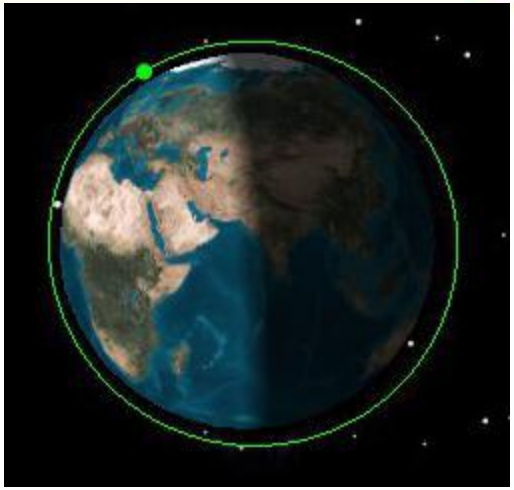
- 2 axis Sun sensors
- 3 axis magnetometers
- Star tracker
- 3 axis gyroscopes (laser gyros)



Power

Solar Panels (SP) design: worst case scenario

	Time spent pr. orbit	Power needed
Sunlight	T_d = 48 min	P_d = 115 W
No SP exposed to sun	T_e = 48 min	P_e = 90 W



$\eta_A = 0.95$ Assembly loss

$\eta_S = 0.95$ Shadowing loss

$\eta_T = 0.85$ Temperature loss

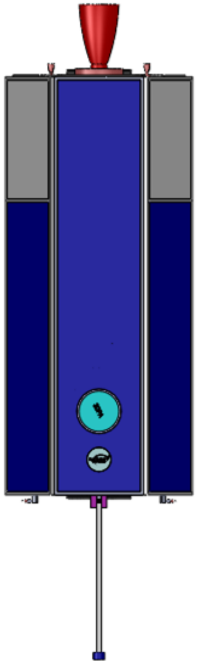
$I_d = \underbrace{\eta_A * \eta_S * \eta_T} \cong 0.77$ Inherent degradation

Selected SP cells: (Power Positive)

- NeXt Triple Junction (XTJ): **P_0 = 398 W/m²** **A_{SP} = 2.5 m²**
- GaInP2/GaAs/Ge **y_{deg} = 2.75% → L_d = 0.72 (after 12 years)**

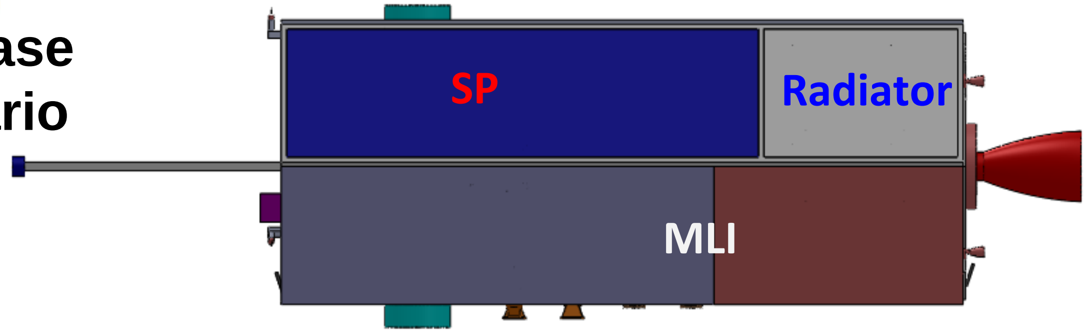
Lithium Ion Batteries: (Power Negative)

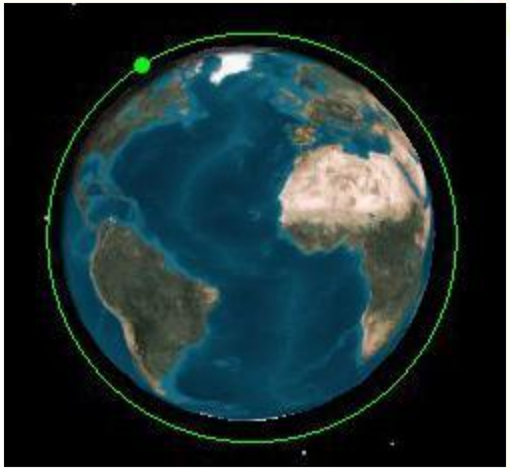
- **Capacity: 38 Ah** **Nominal voltage: 3.6 V**
- **Max. Current : 38 A** **1 battery with mass = 950 g (75% discharge)**



Thermal

Hot Case Scenario





Passive Thermal control

- **Multilayers Insulation (MLI):** exposed surfaces and underneath Solar Panels.
- **Radiators:** 3 radiators are needed: **2 at the sides** and **1 in the back.**

$$P_{BOL-SA} + \alpha S \left(\Phi_S + \Phi_{E_{alb}}\right) + \varepsilon S F_{SP} \Phi_{E_{IR}} = n \varepsilon S \sigma T^4$$

$S_{rad} = 0.4\ m^2$ ← The area was selected to optimize the S/C length

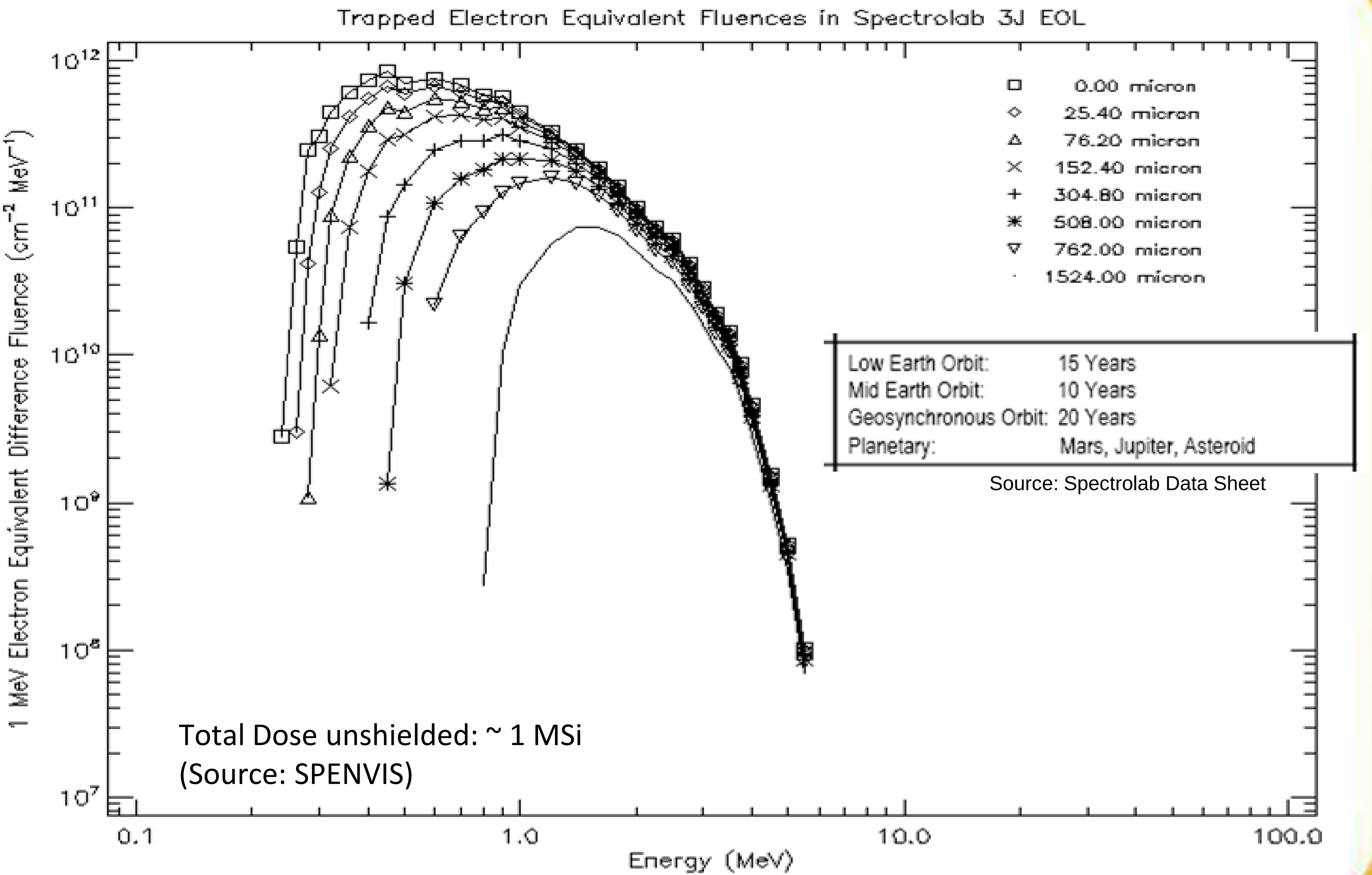
Each radiator will be equipped with louvers.

P_{BOL-SA}	:	Power to dissipate ~ 275 W
α	:	Radiator Absorption coefficient ~ 0.1
S	:	Radiator area
Φ_S	:	Solar flux $\sim$1366 W/m²
$\Phi_{E_{alb}}$	:	Earth flux due to albedo $\sim 239W/m^2$
$\Phi_{E_{IR}}$	:	Earth flux due to Infra-Red $\sim 400W/m^2$
σ	:	Stephan-Boltzmann constant
ε	:	Emissivity $\sim$0.8
F_{SP}	:	Shape factor ~ 0.4
n	:	number of radiations=3

Active Thermal Control

- **Peltier Coolers** will be used to "move" the internal heat toward the radiators.
- **Thermal Heaters** during eclipses.

Radiation Dose and Solar Cell Degradation



Spacecraft Mass Budget

Subsystem	Total (kg)
Structure & Subsystems	325
Avionics & Communications	22
Fuel	165
INMS	3.6
3DPA	1.2
m-NLP	0.3
Boom	3
Thermistors	0.036
CITRIS	5.4
IGOR+	6.96
ISA	9.78
FGM	1.8
TOTAL	**545**

Mass for launch (kg)	
2 x s/c	1090
Vega capacity	1500
Margin for dual s/c adapter	**410**

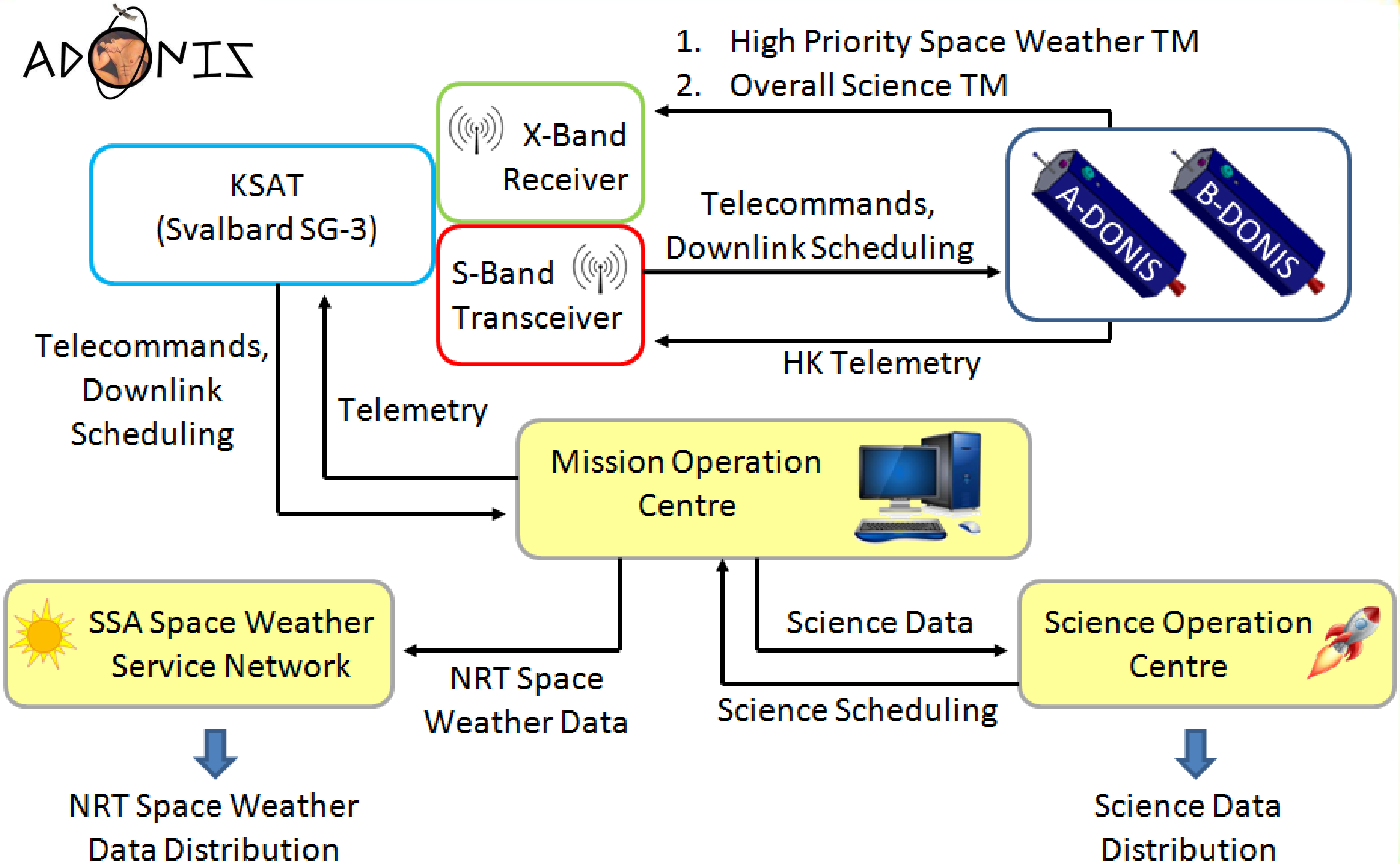
10% margin
20% margin

Spacecraft Power and Data Budget

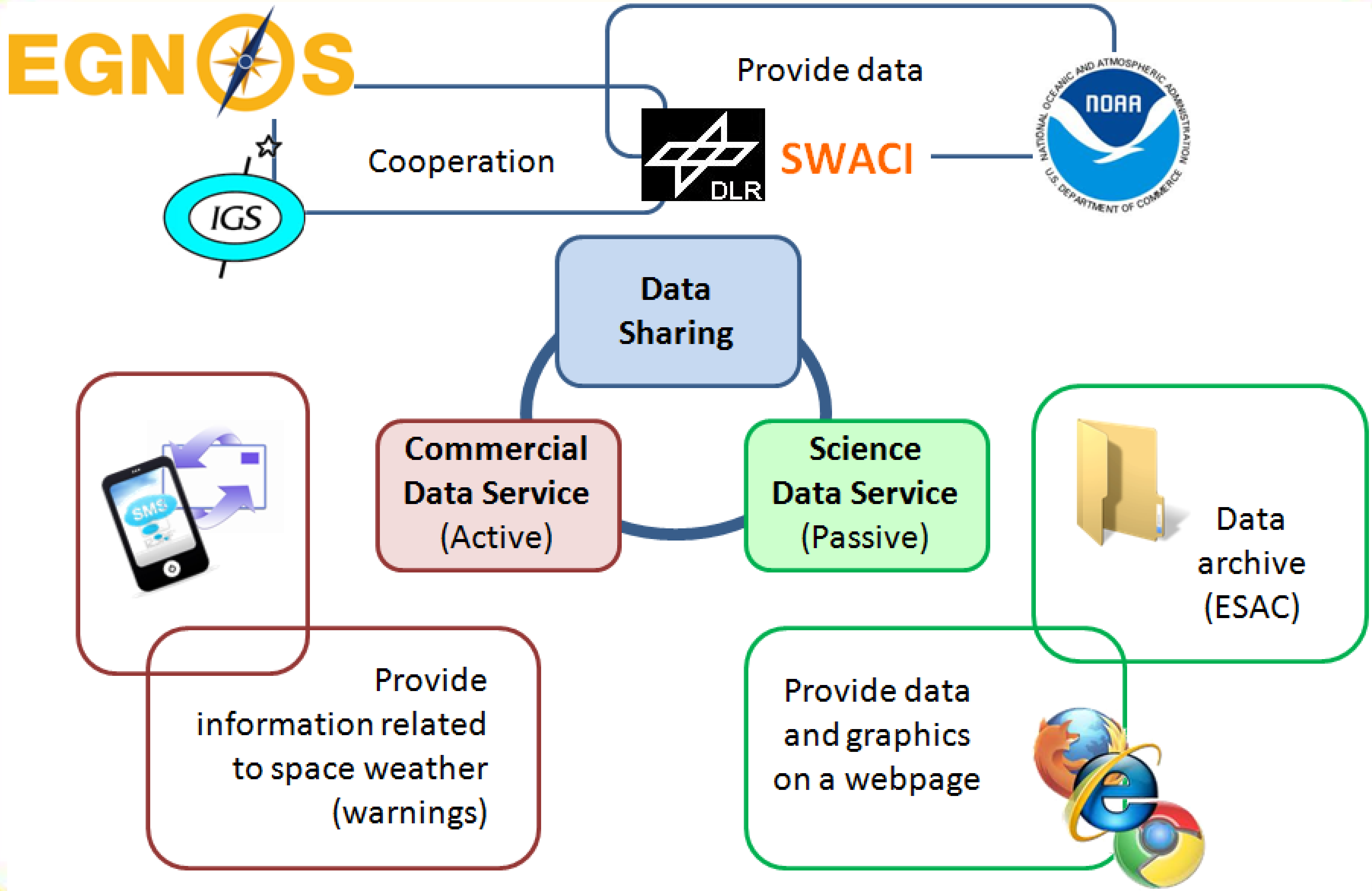
Subsystem	Operation (W)	Data rate (bits/s)
Power provided by Bus	115 (180 peak)	N/A
Avionics	40 (61 peak)	N/A
Communication	15 (50 peak)	N/A
INMS	3	2048
3DPA	1	58000
m-NLP	3.5	1900
Thermistors	0.01	96
CITRIS	12.3	15000
IGOR+	22	20000
ISA	12.1	9600
FGM	0.8	400
TOTAL	**110 W (166 W peak)**	**107 kbps**

Ground Segment & Operations

Ground Segment & Operations



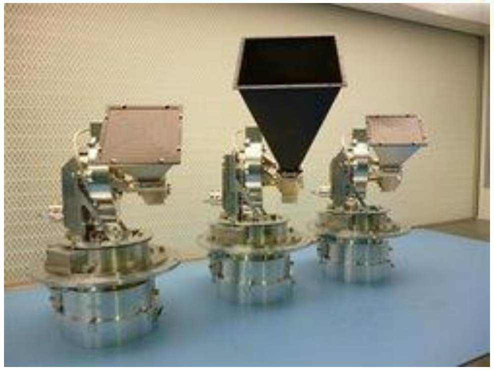
ADONIS Ground Services



Radio Systems Overview

Parameter	X-Band (down)	S-Band (down)	S-Band (up)
Data rate	105 Mbit/s	38.4 kbit/s	19.2 kbit/s
Frequency	8.5 GHz	2.2 GHz	2.1 GHz
Tx Power	5 W	0.5 W	~15W
Tx Antenna	10 cm horn	8 cm patch	13 m dish
Rx Antenna	13 m dish	13 m dish	8 cm patch
Rx G/T	32 dB	23 dB	-25 dB
Link Margin	**6.0 dB**	**15.4 dB**	**40.6 dB**

- Calculated using 2200 km slant range (Determined from STK simulation)



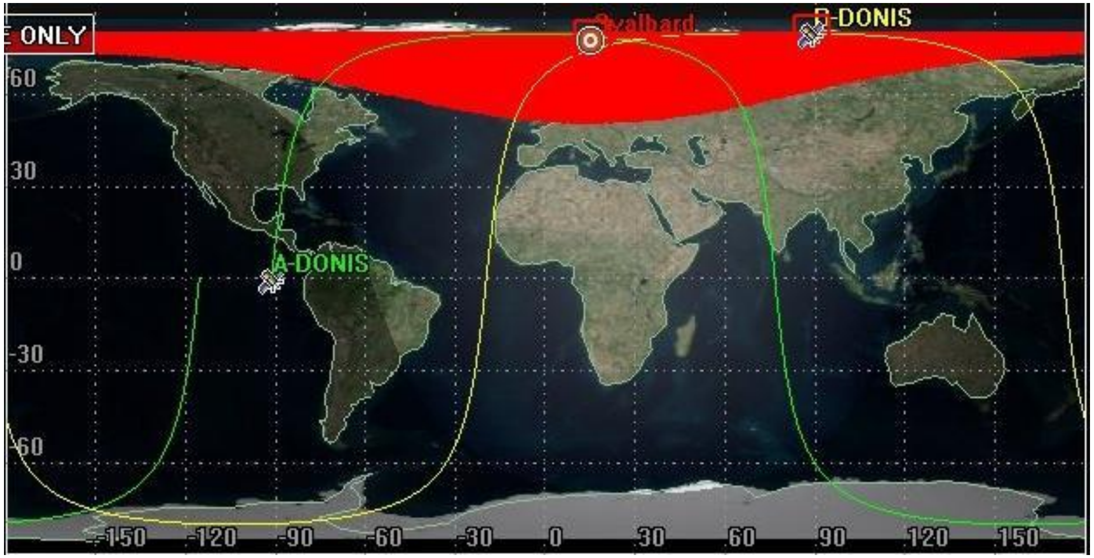
Ground Coverage I

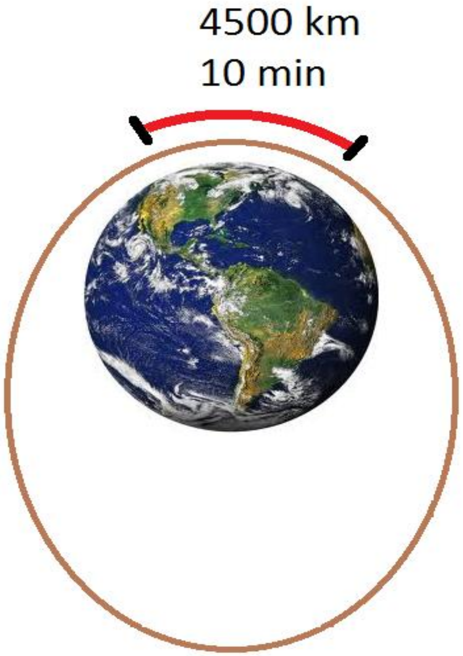
Ground Station (GS) Location: Svalbard (KSAT SG-3)

Orbits with GS coverage: 12.5 of 15.15/day (85%)
Worst-case unusable consecutive passes: 4

- Average ground pass time: 10.5 min/orbit
- Worst-case ground pass time: 4.0 min/orbit

Suggested GS allocation: 4 h/day (for both satellites)





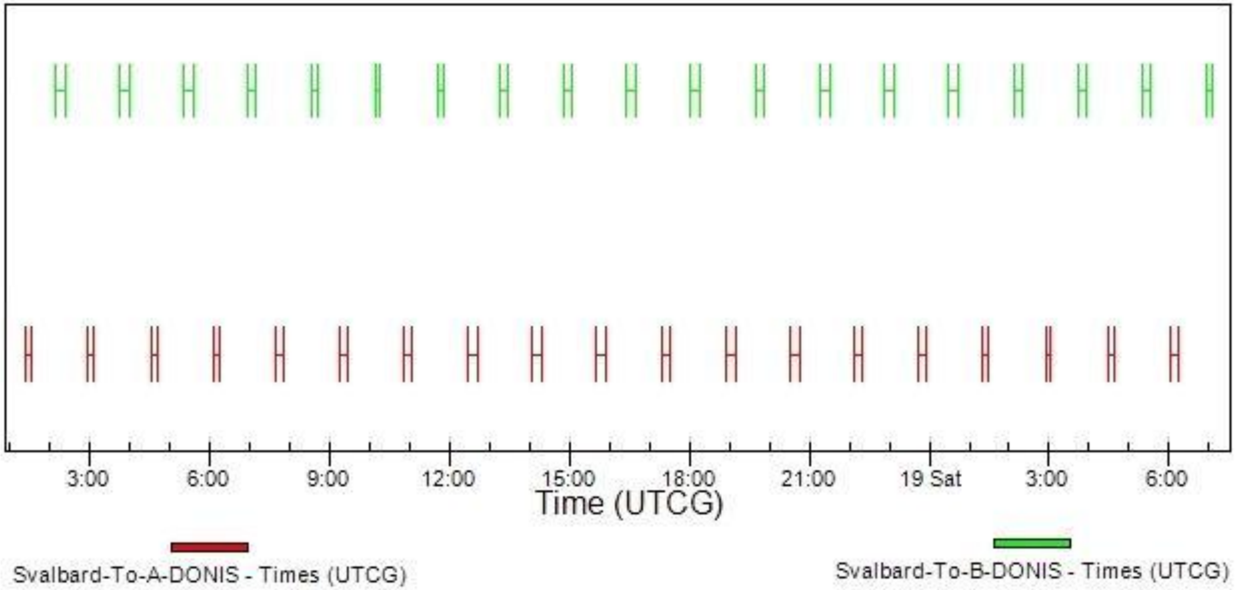
Ground Coverage II

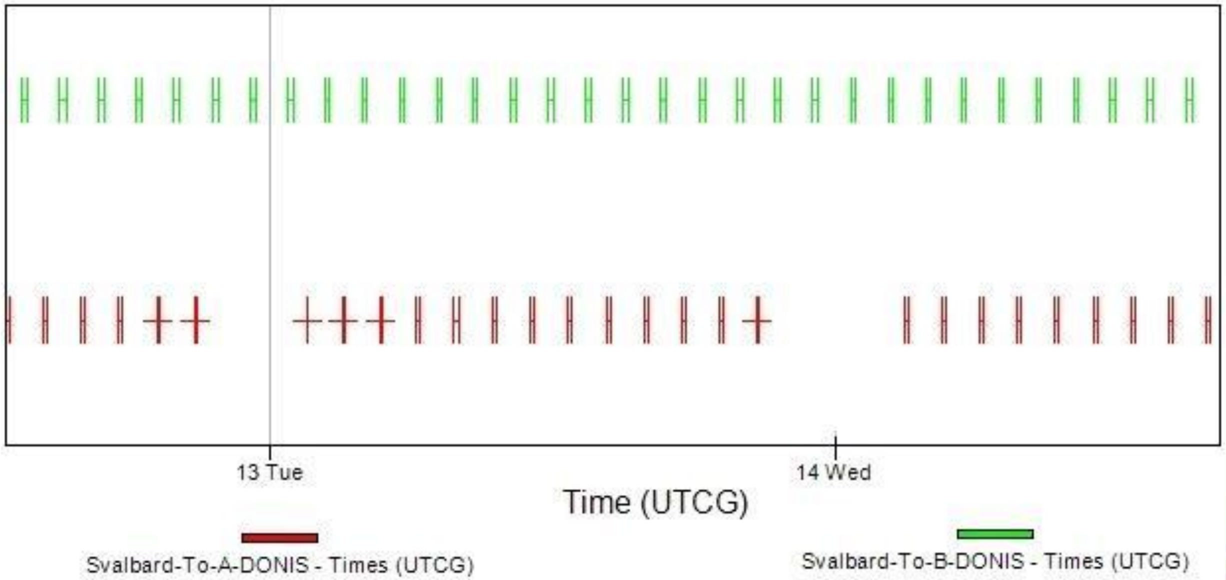
Best case

- ~48 min between downlinks
- >95% of orbits (both satellites)

Worst case

- ~95 min between downlinks
- ~95% for B-DONIS
- ~85% for A-DONIS (alternating)





Science data generation

Parameter	Data generated
Science generated per orbit	720 Mbit
Science generated in 5 orbits (worst-case)	3600 Mbit
Downlink capacity per orbit	4500 Mbit
Worst case data downlink margin	**720 Mbit**

- Mass memory capacity: 128 Gbit
- Total generated scientific data during mission: 40 Tbit

Development, Costs & Risks

Total Costs

Cost Item	Expected Cost (M€)
Arianespace VEGA launcher	35
2× Spacecraft Standard Platform SSTL	50
Customisation for SSTL	60
2× Propulsion Module	35
2× Total Payload	50
Ground Operations	45
Total	**275**

Descope Options:

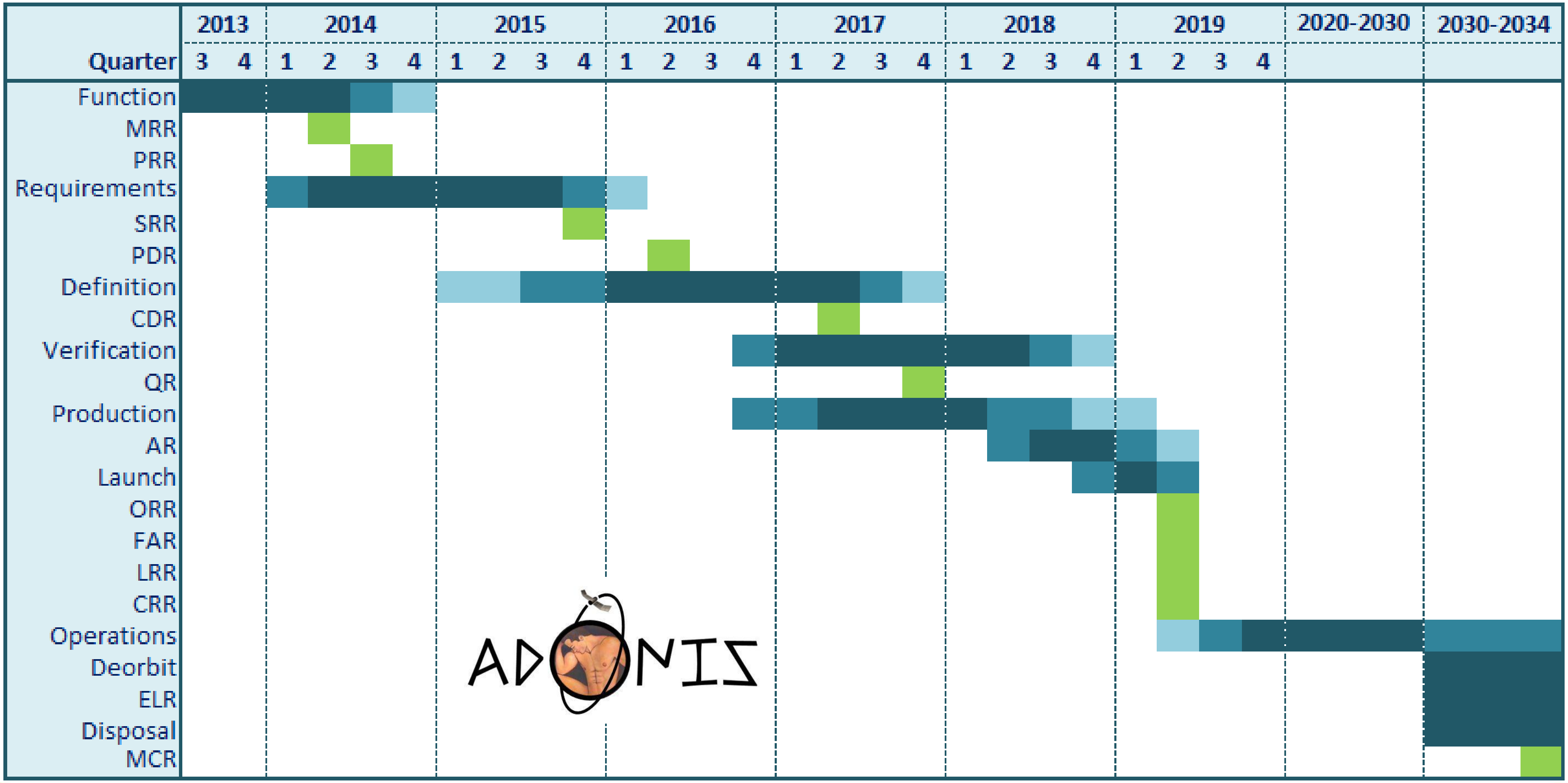
- One spacecraft, less resolution, total descope mission cost: 165 M€
- Shorter mission (5 years), total mission cost: 251 M€

Mission Operations Costs

Parameter	Yearly cost
Satellite tracking	750 k€ / year
Mission Control	2000 k€ / year
Science Operations	1000 k€ / year
NRT Operations	300 k€ / year
Total cost	**4050 k€ / year**

Total missions operations cost (11 years): 45 M€

Mission Timeline

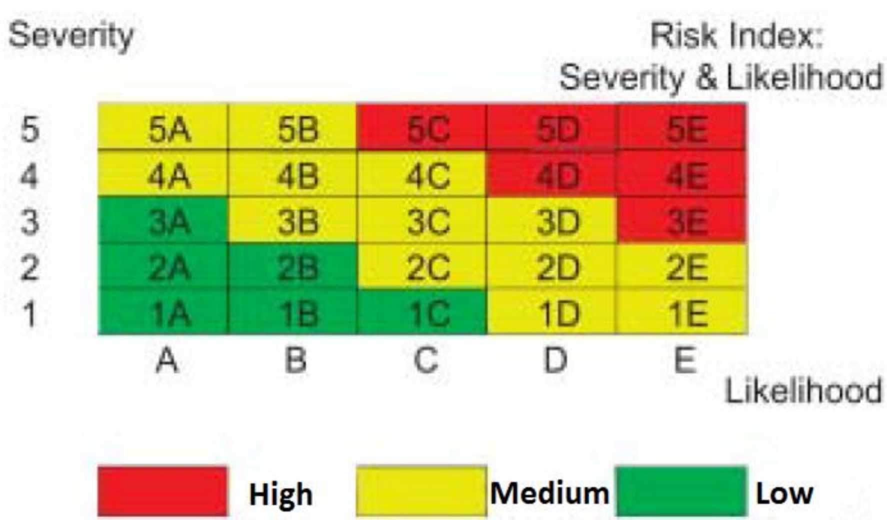


LEGEND			
MDR Mission Design	**CDR** Critical Design	**FAR** Flight Acceptance	**MCR** Mission Close Out
PRR Preliminary Requirements	**QR** Qualification	**LRR** Launch Readiness	
SRR System Requirements	**AR** Acceptance	**CRR** Commisioning Result	Milestones
PDR Preliminary Design	**ORR** Operational Readiness	**ELR** End Of Life	

Risks

Identified risk	Severity	Likelihood	Impact	Mitigation
Launcher failure	5	C	performance	accepted
Plasma analyser not space proven	2	C	performance	replace with existing simpler instrument
Space weather	3	C	performance	accepted
Bus customisation	3	C	cost	replace with different

→ No higher risk than an average Low Earth Orbit mission



P. Falkner, Alpach 2013

Disposal

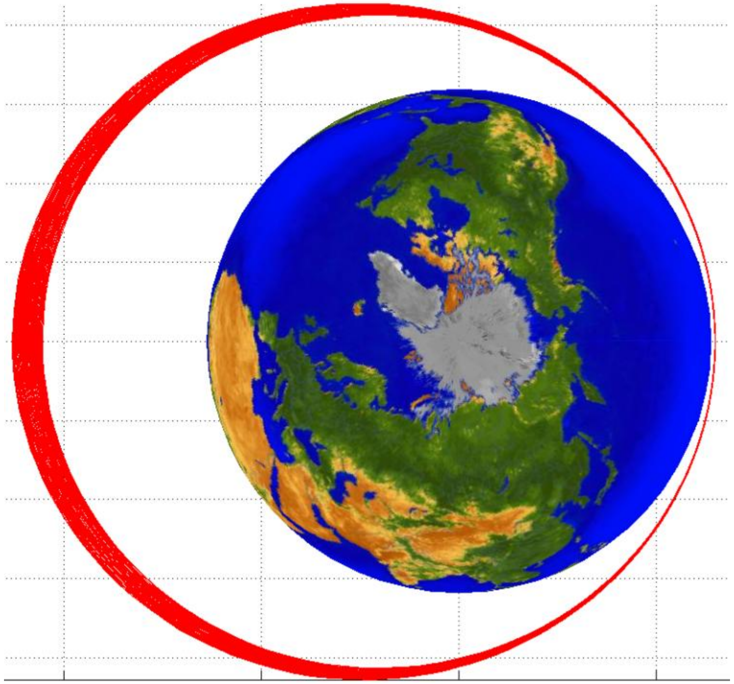
Disposal

ESA requests the removal of space systems in the **LEO region** not later than **25 years after the end of the mission**.

The **orbital decay** will be part of the **scientific phase**, and will allow to:

1. study the drag at **300 km with different perigee velocities** (circularisation phase);
2. study of the drag **below 300 km** until re-entry of the satellites (spiralisation phase);

Disposal Strategy:
Controlled re-entry using thrusters over unpopulated areas as a part of the mission.

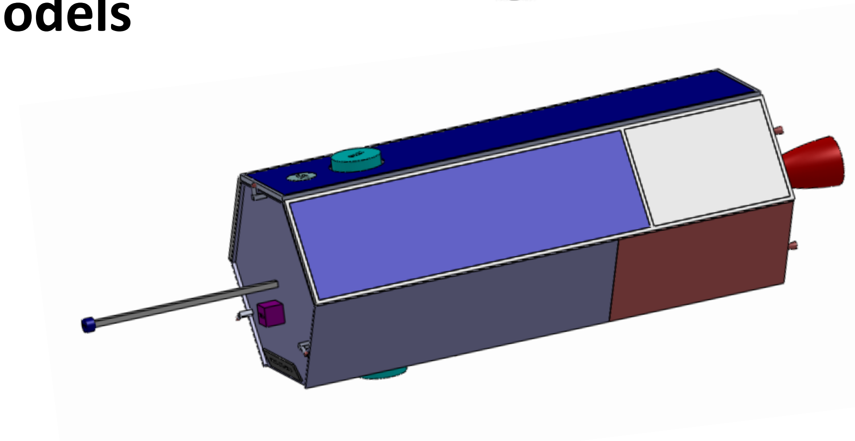


Summary

Summary – Novelties – Usages

Summary:

- **In-situ** measurement of **drag parameters**
- New altitudes, long duration
- Correlation with SWE
- Data to **improve ionospheric models**
- NRT data near polar regions



Benefits:

- **Launch** optimisation
- **De-orbiting**
- **Fuel** estimates for s/c
- Reduce environmental **pollution** (rerouting polar flights + 30% fuel)
- Optimisation of broadcast **power**
- **TEC map** improvement

Thank you for your attention!