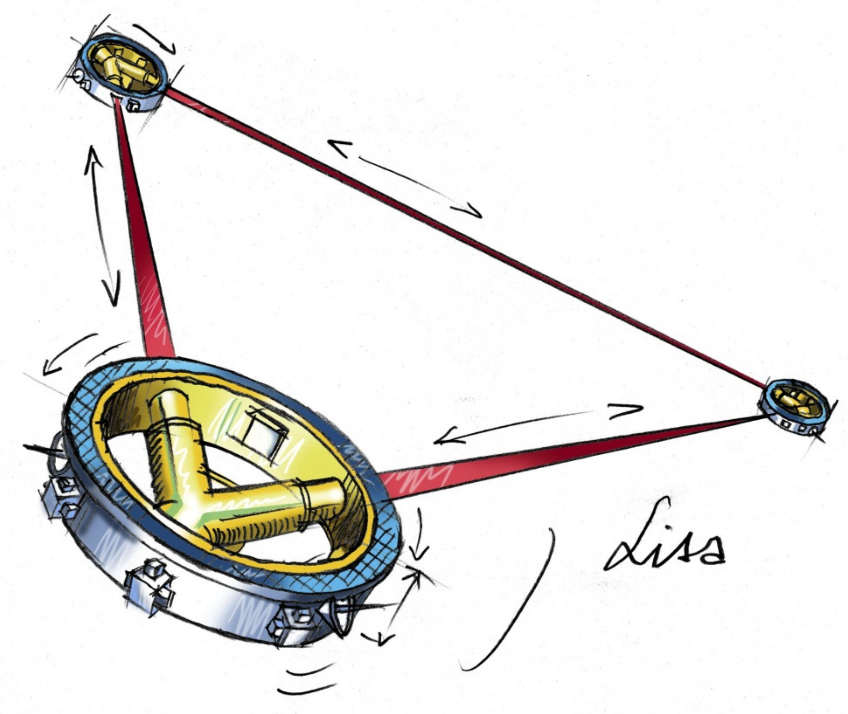
LISA



Welcome

Internal Final Presentation
ESTEC, 05-05-2017

Prepared by the CDF* Team

(*) ESTEC Concurrent Design Facility

AGENDA

Start	Duration	End	Subject
09:30	00:10	09:40	Welcome
09:40	00:10	09:50	Study objectives
09:50	00:20	10:10	Science objectives
10:10	00:50	11:00	System
11:00	00:30	11:30	Payload
11:30	00:15	11:45	**Coffee Break**
11:45	00:20	12:05	Mission analysis
12:05	00:15	12:20	GS and Ops
12:20	00:25	12:45	DFACS - AOCS
12:45	00:20	13:05	Chemical Propulsion
13:05	00:25	13:30	Electric Propulsion
13:30	01:00	14:30	**Lunch Break**
14:30	00:15	14:45	Comms
14:45	00:15	15:00	DHS
15:00	00:15	15:15	Power
15:15	00:15	15:30	Mechanisms
15:30	00:15	15:45	Configuration
15:45	00:15	16:00	Structures
16:00	00:15	16:15	**Coffee Break**
16:15	00:15	16:30	Thermal
16:30	00:15	16:45	Risk
16:45	00:15	17:00	Programatics
17:00	00:15	17:15	Conclusions
17:15	00:15	17:30	Way forward

STUDY BACKGROUND

- CDF has been requested by SCI-FM (under GSP funding) to perform a preliminary mission design for the Cosmic Vision L3 mission:

  LISA (Laser Interferometer Space Antenna)

- LISA has already been studied both in Industry and at the CDF, Europe and America

- The main goal of the mission is to detect and observe Gravitational Waves
- The sensing methodology is laser interferometry between free flying Test Masses
- A constellation of three spacecraft is required, flying in a triangle in an Earth Trailing orbit
- A significant part of the payload components have been successfully demonstrated in LISA PathFinder

CDF STUDY OBJECTIVES

- The main objectives of the present CDF study are:
  - Design a mission compatible with the updated Science Goals
  - Iterate the mission design, incl. launcher, final orbit definition and transfer trajectories
  - Define the mission architecture, including assessment of system options
  - Define the spacecraft configuration required to accommodate the payload
  - Develop a preliminary design of the payload
  - Define operational scheme
  - Define system integration and testing flows
  - Asses impact of science extension to 10 years
  - Provide risk and cost assessments

STUDY SCHEDULE

SESSION	DAY	DATE	TIME
Kick Off	Wednesday	08/03/2017	13:30-17:30 CET
#2	Friday	10/03/2017	9:30-13:30 CET
#3	Wednesday	15/03/2017	9:30-13:30 CET
#4	Friday	17/03/2017	9:30-13:30 CET
#5	Wednesday	22/03/2017	13:30-17:30 CET
#6	Friday	24/03/2017	13:30-17:30 CET
#7	Wednesday	29/03/2017	9:30-13:30 CET
#8	Friday	31/03/2017	9:30-13:30 CET
#9	Wednesday	05/04/2017	13:30-17:30 CET
#10	Friday	07/04/2017	9:30-13:30 CET
#11	Wednesday	12/04/2017	9:30-13:30 CET
#12	Wednesday	03/05/2017	9:30-13:30 CET
[illegible]al Final Presentation	Friday	05/05/2017	9:30-17:30 CET

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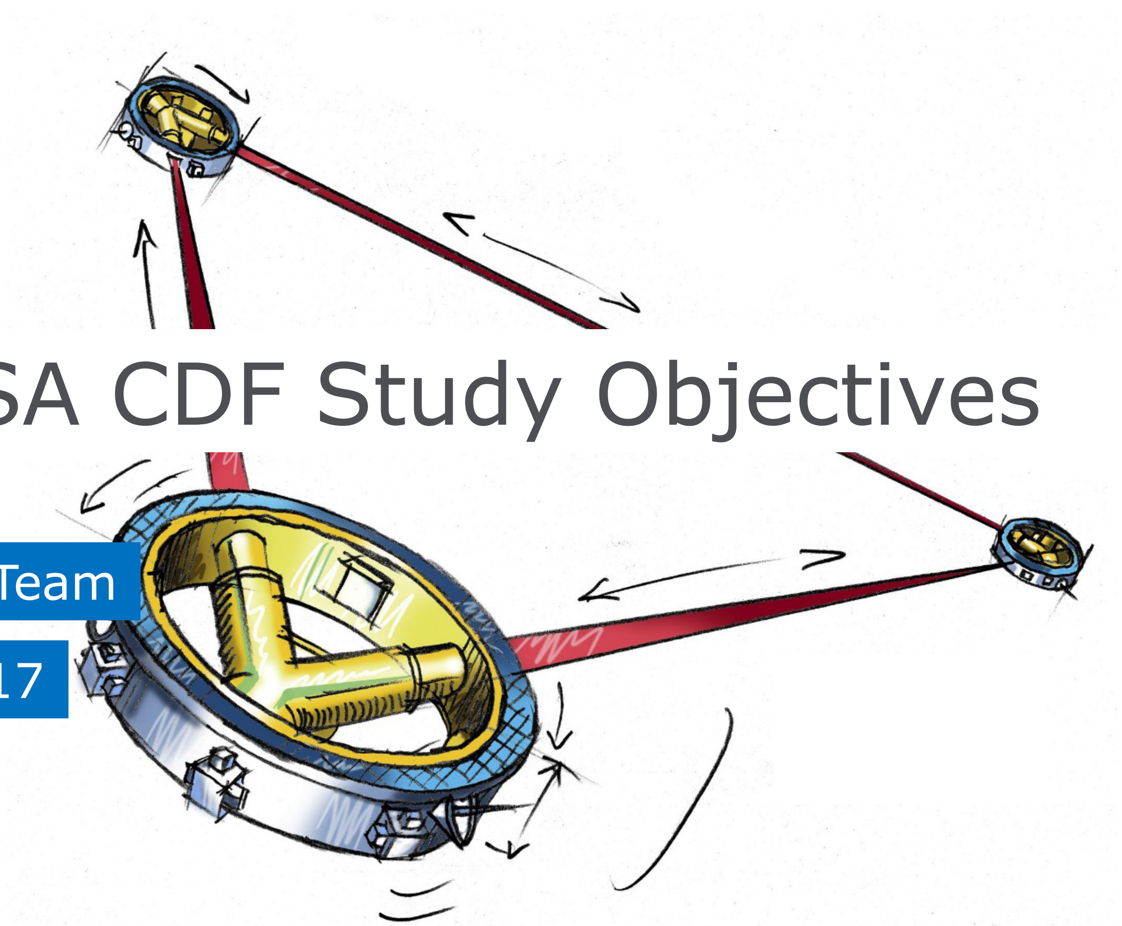
L3/LISA CDF Study Objectives



L3 Study Team

05/05/2017

L3/LISA Background Overview

- **L3/LISA: third large class mission in the science programme**
- **Call for L2, L3 themes (white paper):** March 2013 - November 2013
  L2: “The hot and energetic Universe”; L3: “The gravitational Universe”
- **Call for L3 missions:** October 2016 - January 2017
  - Selection of the L3 candidate expected by June SPC
- **LISA (Laser Interferometer Space Antenna)** is a mission proposal received in response to the call for L3 missions.

- CDF Phase 0 study requested by SCI-FM to assess the mission feasibility, taking into account changes from previous studies.
  - Input for the industrial assessment Phase-A (2018/2019)
- CDF study funded by GSP

L3/LISA Programmatic Boundary Conditions

- Europe-led mission
- Launch Date: ~**2034** (one launch!)
- Cost Envelope: **1050 MEUR CaC to ESA** (i.e. plus member state contributions and plus other contributions (e.g. NASA))
- Technology Readiness:
  - **TRL 5 / 6** for all critical subsystems (incl. payload) **by adoption (2022-2024)**
  - The earlier, the better!

Study Objectives

- The main objectives of the L3/LISA CDF study were:
  - Design a mission compatible with the updated Science Goals
  - Iterate the mission design, incl. launcher, final orbit definition and transfer trajectories
  - Define the mission architecture
  - Define the spacecraft configuration required to accommodate the payload
  - Develop a preliminary design of the payload
  - Define system integration and testing flows
  - Provide preliminary development plans
  - Define operational scheme
  - Asses impact of life extension to 10 years
  - Provide risk and cost assessments

Gravitational Wave Astronomy: Sounds from the Dark Side of the Universe!

Prof. Dr. Karsten Danzmann

Albert-Einstein-Institut:

Max-Planck-Institut für Gravitationsphysik

und

Institut für Gravitationsphysik der Leibniz Universität Hannover

We have written Science History!

Prof. Dr. Karsten Danzmann

Max-Planck-Institut für Gravitationsphysik
und
Institut für Gravitationsphysik der Leibniz Universität Hannover

"All the News
That's Fit to Print"

The New York Times

Late Edition

Today, some sunshine giving way to times of clouds, cold, high 28. **Tonight,** a flurry or heavier squall late, low 15. **Tomorrow,** windy, frigid, high 21. Weather map, Page A19.

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Clinton Paints Sanders Plans As Unrealistic

New Lines of Attack at Milwaukee Debate

By AMY CHOZICK and PATRICK HEALY

MILWAUKEE — Hillary Clinton, scrambling to recover from her double-digit defeat in the New Hampshire primary, repeatedly challenged the trillion-dollar policy plans of Bernie Sanders at their presidential debate on Thursday night and portrayed him as a big talker who needed to "level" with voters about the difficulty of accomplishing his agenda.

Foreign affairs also took on unusual prominence as Mrs. Clinton sought to underscore her experience and Mr. Sanders excoriated her judgment on Libya and Iraq, as well as her previous praise of former Secretary of State Henry A. Kissinger. But Mrs. Clinton was frequently on the offensive as well, seizing an opportunity to talk about leaders she admired and turning it against Mr. Sanders by bashing his past criticism of President Obama — a remark that Mr. Sanders called a "low blow."

With tensions between the two Democrats becoming increasingly obvious, the debate was full of new lines of attack from Mrs. Clinton, who faces pressure to puncture Mr. Sanders's growing popularity before the next nomi-

CALTECH-M.I.T.-LIGO LABORATORY

A worker installed a baffle in 2010 to control light in the Laser Interferometer Gravitational-Wave Observatory in Hanford, Wash.

WITH FAINT CHIRP, SCIENTISTS PROVE EINSTEIN CORRECT

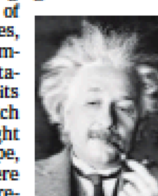
A RIPPLE IN SPACE-TIME

An Echo of Black Holes Colliding a Billion Light-Years Away

By DENNIS OVERBYE

A team of scientists announced on Thursday that they had heard and recorded the sound of two black holes colliding a billion light-years away, a fleeting chirp that fulfilled the last prediction of Einstein's general theory of relativity.

That faint rising tone, physicists say, is the first direct evidence of gravitational waves, the ripples in the fabric of space-time that Einstein predicted a century ago. It completes his vision of a universe in which space and time are interwoven and dynamic, able to stretch, shrink and jiggle. And it is a ringing confirmation of the nature of black holes, the bottomless gravitational pits from which not even light can escape, which were the most foreboding (and



Long in Clinton's Corner, Blacks Notice Sanders

Courted Hard in South Carolina, Loyalists Listen Closely

By RICHARD FAUSSET

ORANGEBURG, S.C. — When Helen Duley was asked whom she would vote for in the South Carolina primary, she answered

candidate she barely knew. "It makes me feel good," she said, chuckling, "that young people are listening to the elderly people." She now said she was an undecided voter and planned to do some homework on Mr. Sanders.

Last Occupier In Rural Oregon Is Coaxed Out

We have detected Gravitational Waves!

Observation of Gravitational Waves from a Binary Black Hole Merger

B. P. Abbott *et al.*[*]

(LIGO Scientific Collaboration and Virgo Collaboration)

(Received 21 January 2016; published 11 February 2016)

On September 14, 2015 at 09:50:45 UTC the two detectors of the Laser Interferometer Gravitational-Wave Observatory simultaneously observed a transient gravitational-wave signal. The signal sweeps upwards in frequency from 35 to 250 Hz with a peak gravitational-wave strain of 1.0×10^{-21}. It matches the waveform predicted by general relativity for the inspiral and merger of a pair of black holes and the ringdown of the resulting single black hole. The signal was observed with a matched-filter signal-to-noise ratio of 24 and a false alarm rate estimated to be less than 1 event per 203 000 years, equivalent to a significance greater than 5.1σ. The source lies at a luminosity distance of 410^{+160}_{-180} Mpc corresponding to a redshift $z = 0.09^{+0.03}_{-0.04}$. In the source frame, the initial black hole masses are $36^{+5}_{-4}M_\odot$ and $29^{+4}_{-4}M_\odot$, and the final black hole mass is $62^{+4}_{-4}M_\odot$, with $3.0^{+0.5}_{-0.5}M_\odot c^2$ radiated in gravitational waves. All uncertainties define 90% credible intervals. These observations demonstrate the existence of binary stellar-mass black hole systems. This is the first direct detection of gravitational waves and the first observation of a binary black hole merger.

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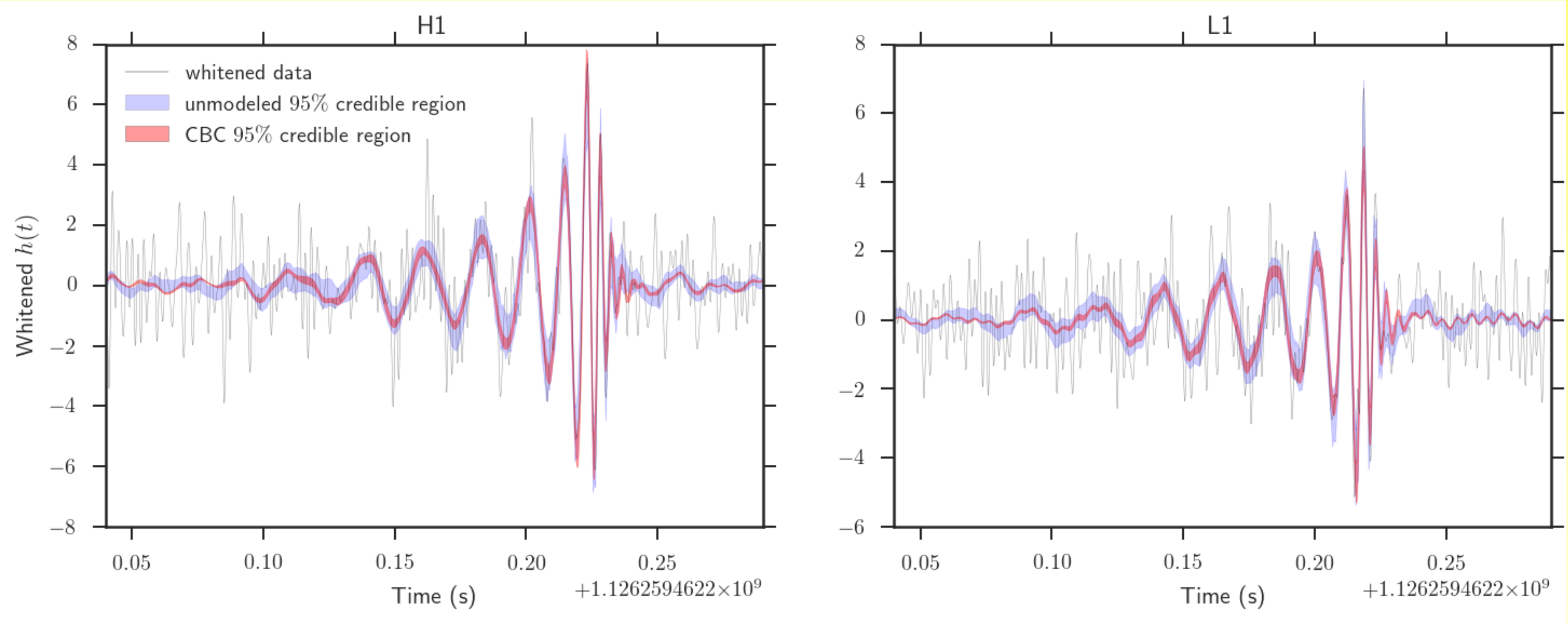

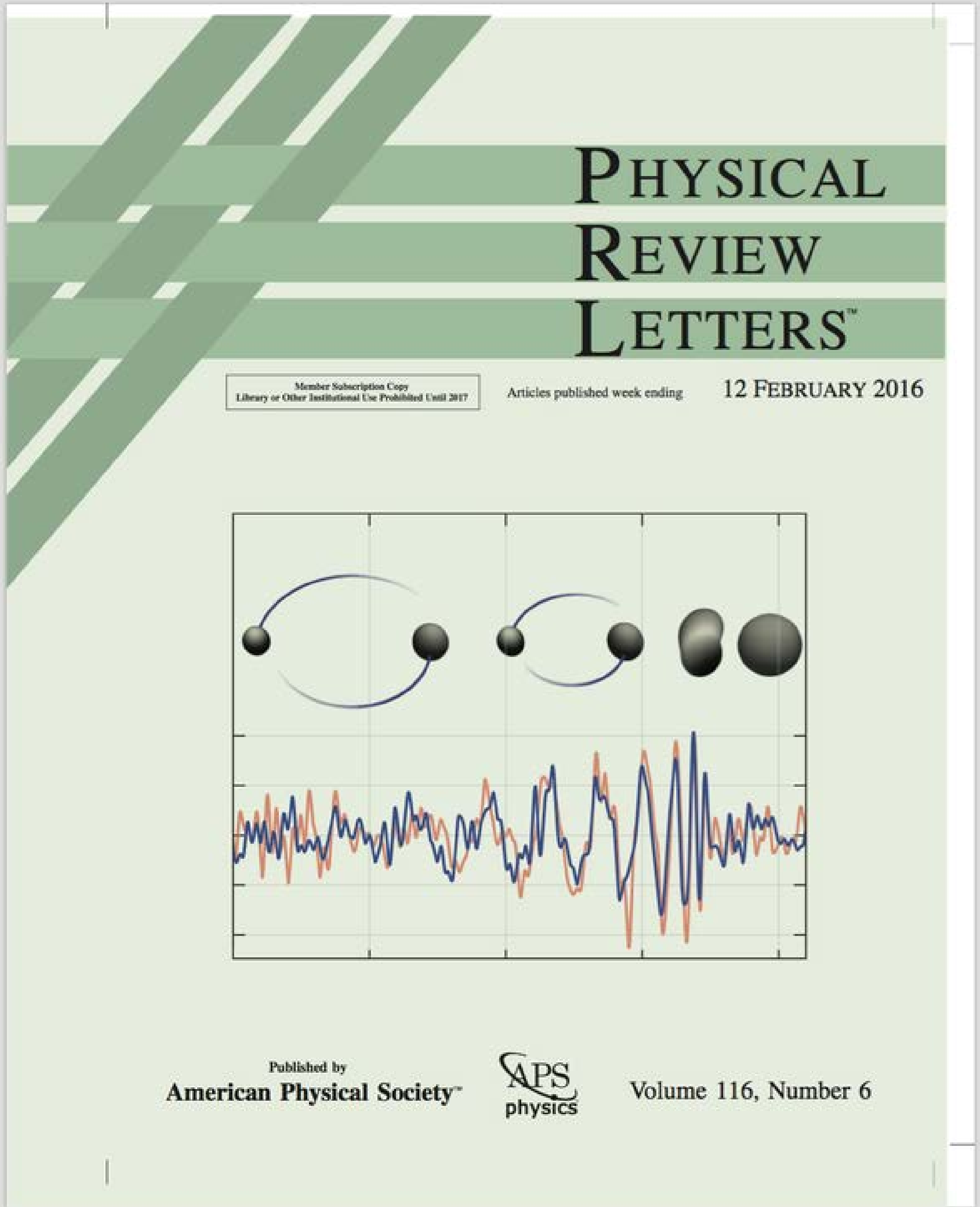

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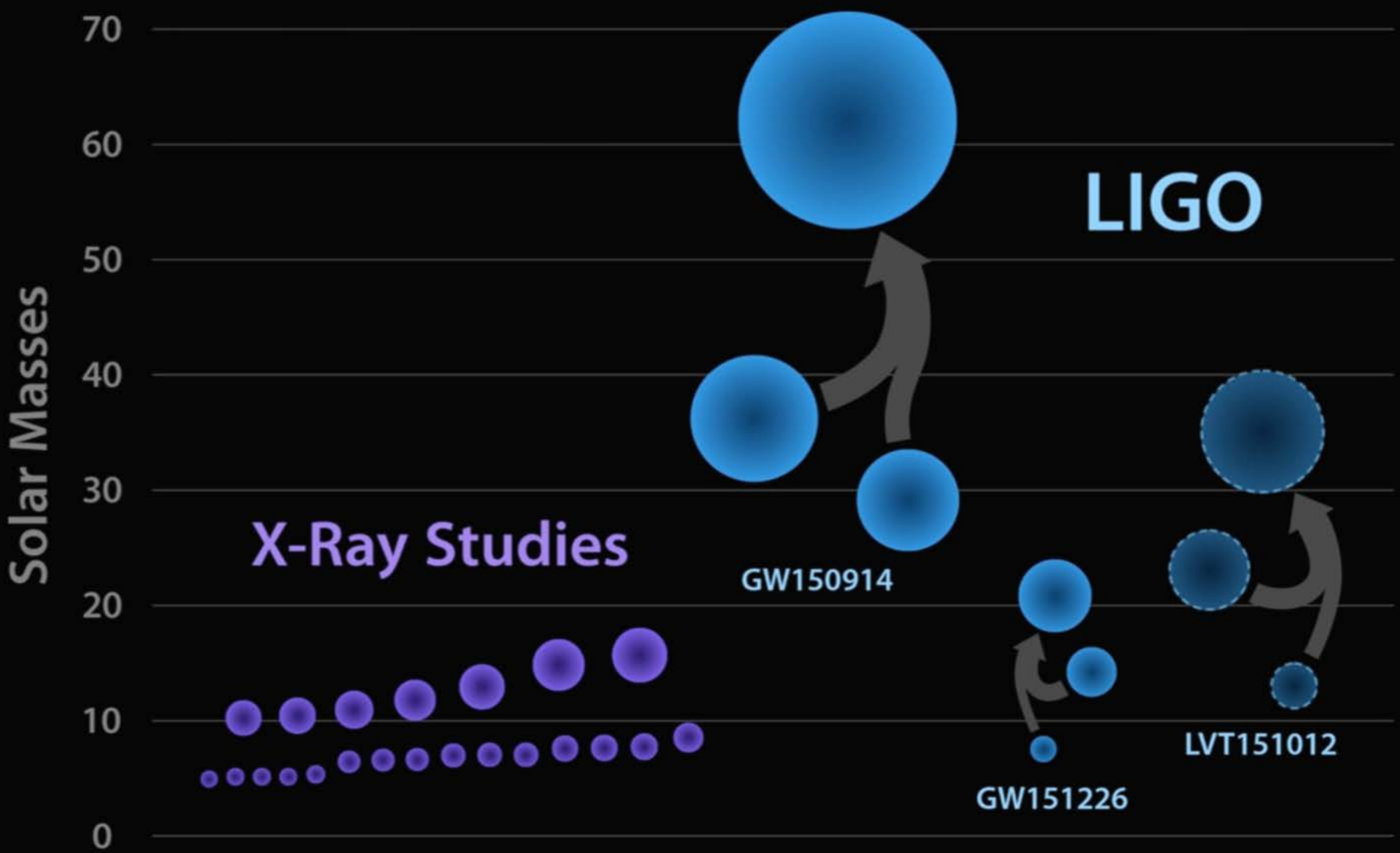
Black Holes of Known Mass



Credit: LIGO

PRL **116,** 201301 (2016) PHYSICAL REVIEW LETTERS week ending 20 MAY 2016

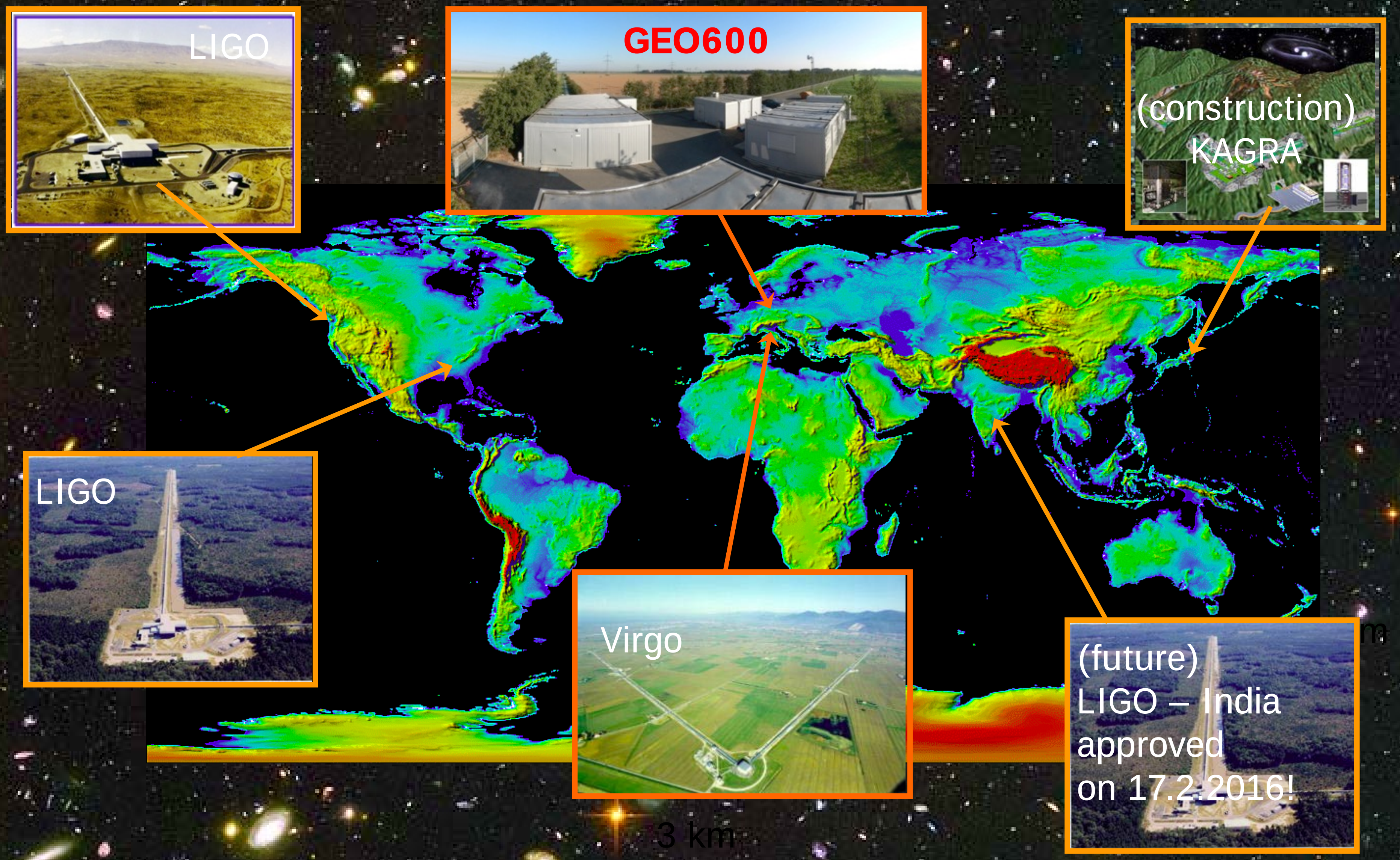
Did LIGO Detect Dark Matter?

Simeon Bird,* Ilias Cholis, Julian B. Muñoz, Yacine Ali-Haïmoud, Marc Kamionkowski,
Ely D. Kovetz, Alvise Raccanelli, and Adam G. Riess
Department of Physics and Astronomy, Johns Hopkins University,
3400 North Charles Street, Baltimore, Maryland 21218, USA
(Received 4 March 2016; published 19 May 2016)

We consider the possibility that the black-hole (BH) binary detected by LIGO may be a signature of dark matter. Interestingly enough, there remains a window for masses $20M_\odot \lesssim M_{\rm bh} \lesssim 100M_\odot$ where primordial black holes (PBHs) may constitute the dark matter. If two BHs in a galactic halo pass sufficiently close, they radiate enough energy in gravitational waves to become gravitationally bound. The bound BHs will rapidly spiral inward due to the emission of gravitational radiation and ultimately will merge. Uncertainties in the rate for such events arise from our imprecise knowledge of the phase-space structure of galactic halos on the smallest scales. Still, reasonable estimates span a range that overlaps the 2–53 $\mathrm{Gpc}^{-3}\,\mathrm{yr}^{-1}$ rate estimated from GW150914, thus raising the possibility that LIGO has detected PBH dark matter. PBH mergers are likely to be distributed spatially more like dark matter than luminous matter and have neither optical nor neutrino counterparts. They may be distinguished from mergers of BHs from more traditional astrophysical sources through the observed mass spectrum, their high ellipticities, or their stochastic gravitational wave background. Next-generation experiments will be invaluable in performing these tests.

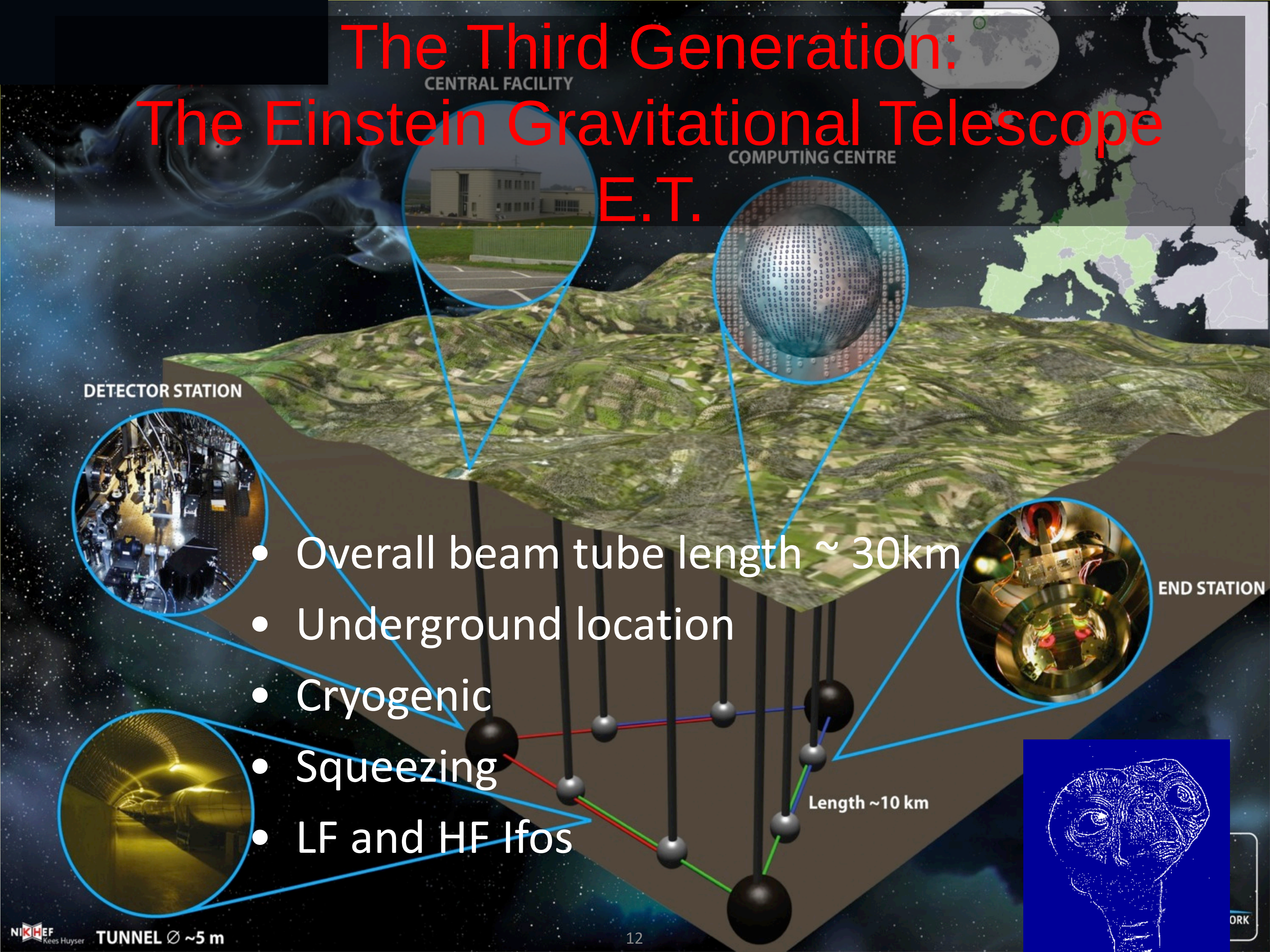
DOI: 10.1103/PhysRevLett.116.201301

World-Wide Laser Interferometric Gravitational Wave Detector Network



Source: A. Lazzarini, modified

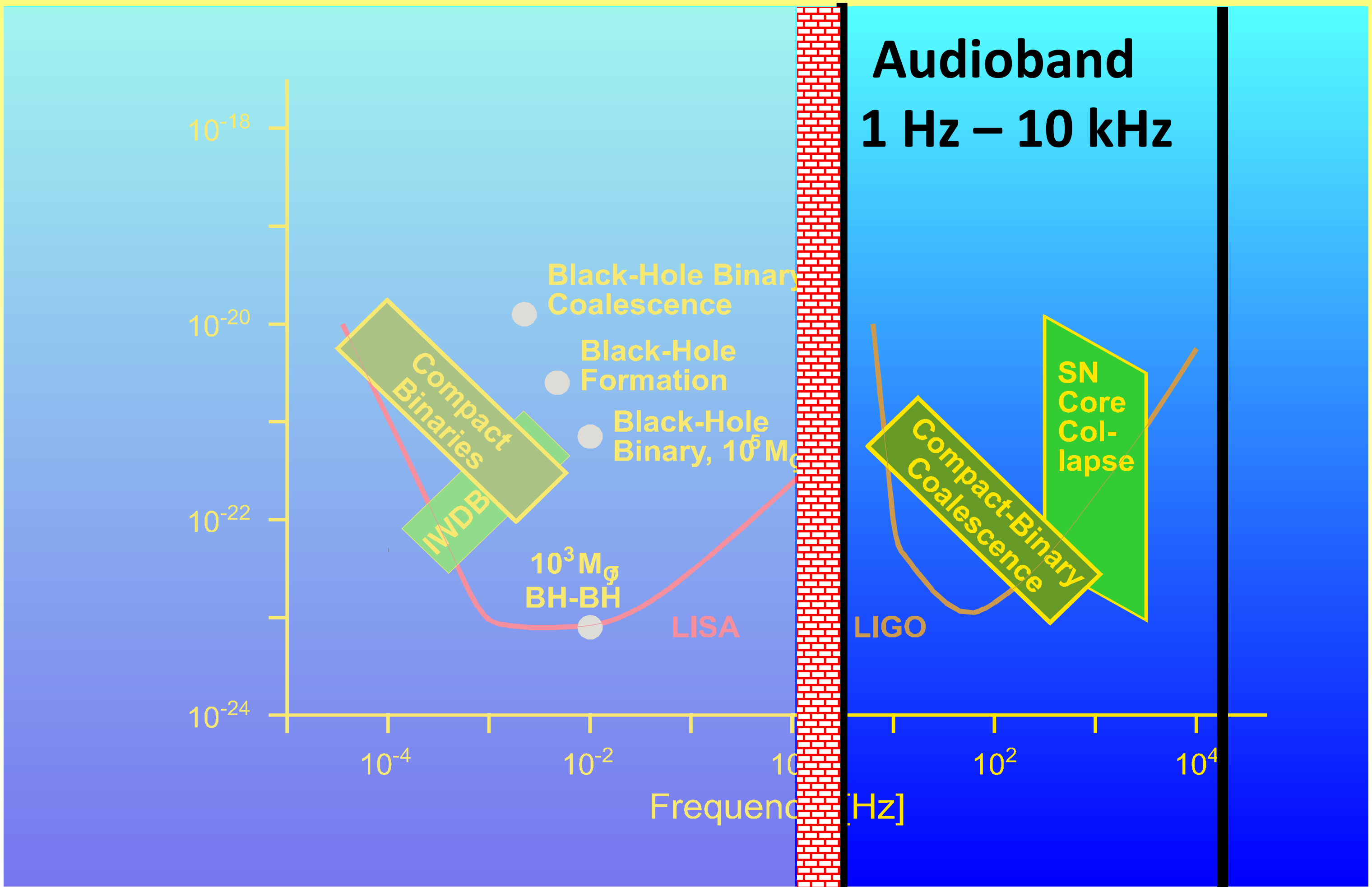
The Third Generation: The Einstein Gravitational Telescope E.T.

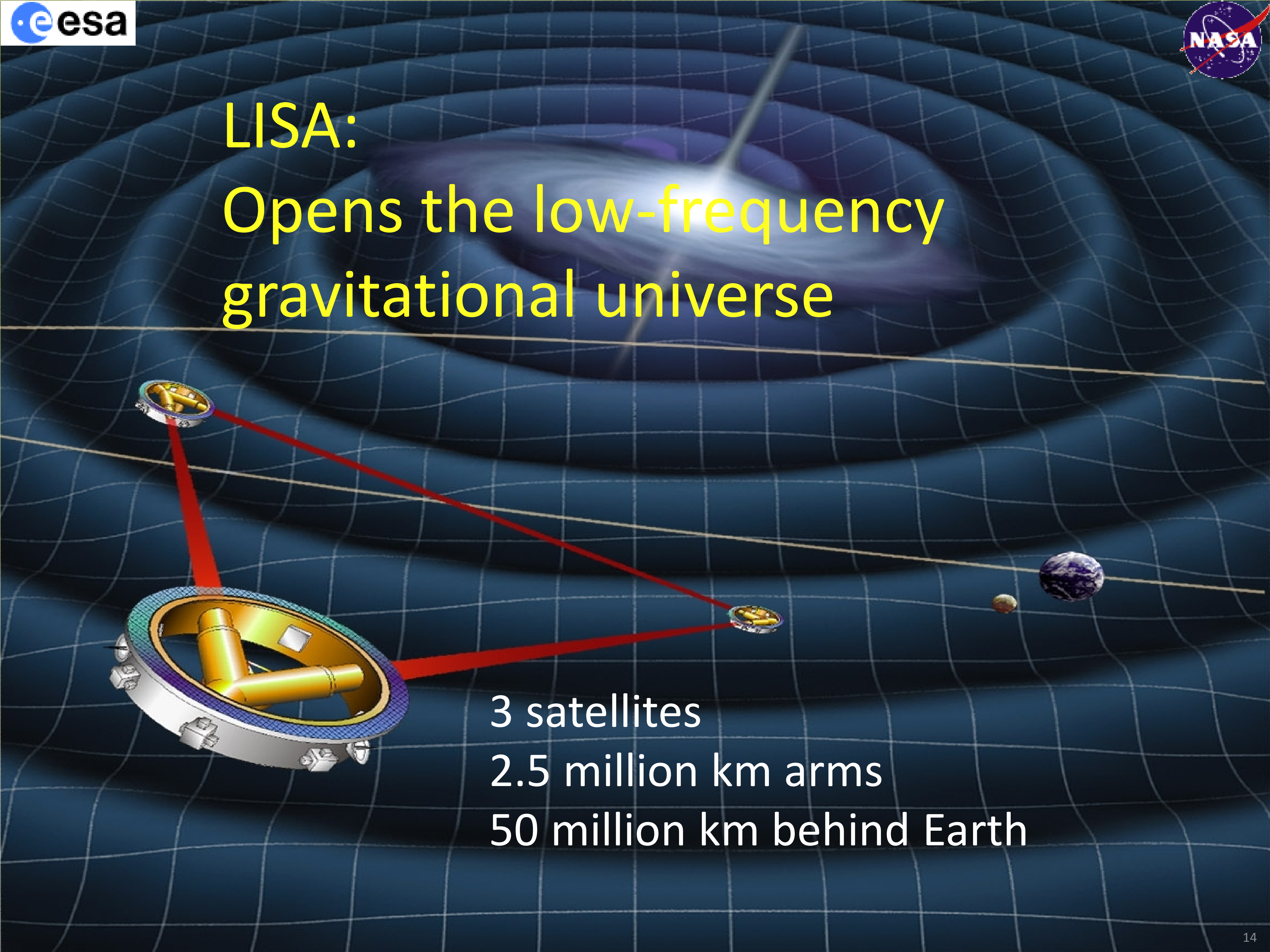


- Overall beam tube length ~ 30km
- Underground location
- Cryogenic
- Squeezing
- LF and HF Ifos

Sources of Gravitational Waves

- Ground-based detectors: Audioband

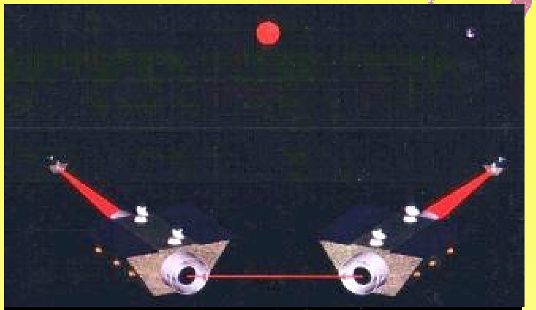


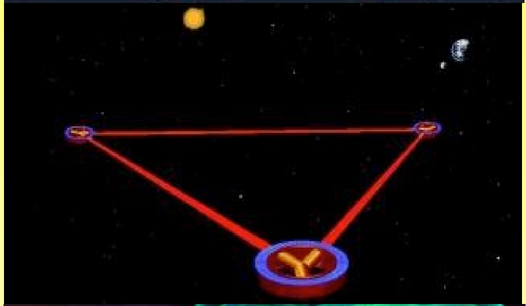

esa
NASA
LISA:
Opens the low-frequency
gravitational universe
3 satellites
2.5 million km arms
50 million km behind Earth

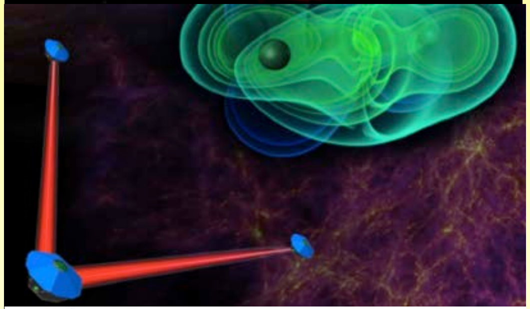
21 Years after the
First LISA Symposium at RAL 1996

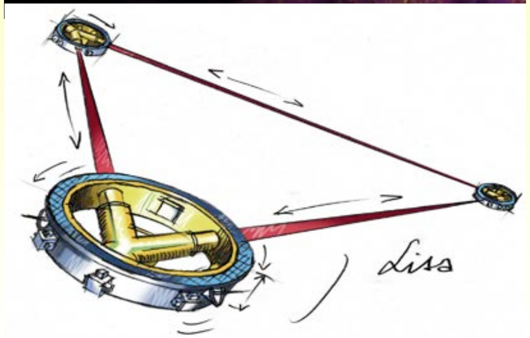
LISA: A Mature Concept

- M3 proposal for 4 S/C ESA/NASA collaborative mission in 1993
- LISA selected as ESA Cornerstone in 1995
- 3 S/C ESA/NASA LISA appears in 1997
- Joint ESA-NASA Mission Formulation study 2005-2011
- Reformulation 2012-13 as ESA-led eLISA (evolving LISA)
- Now back to 3-arm LISA with NASA





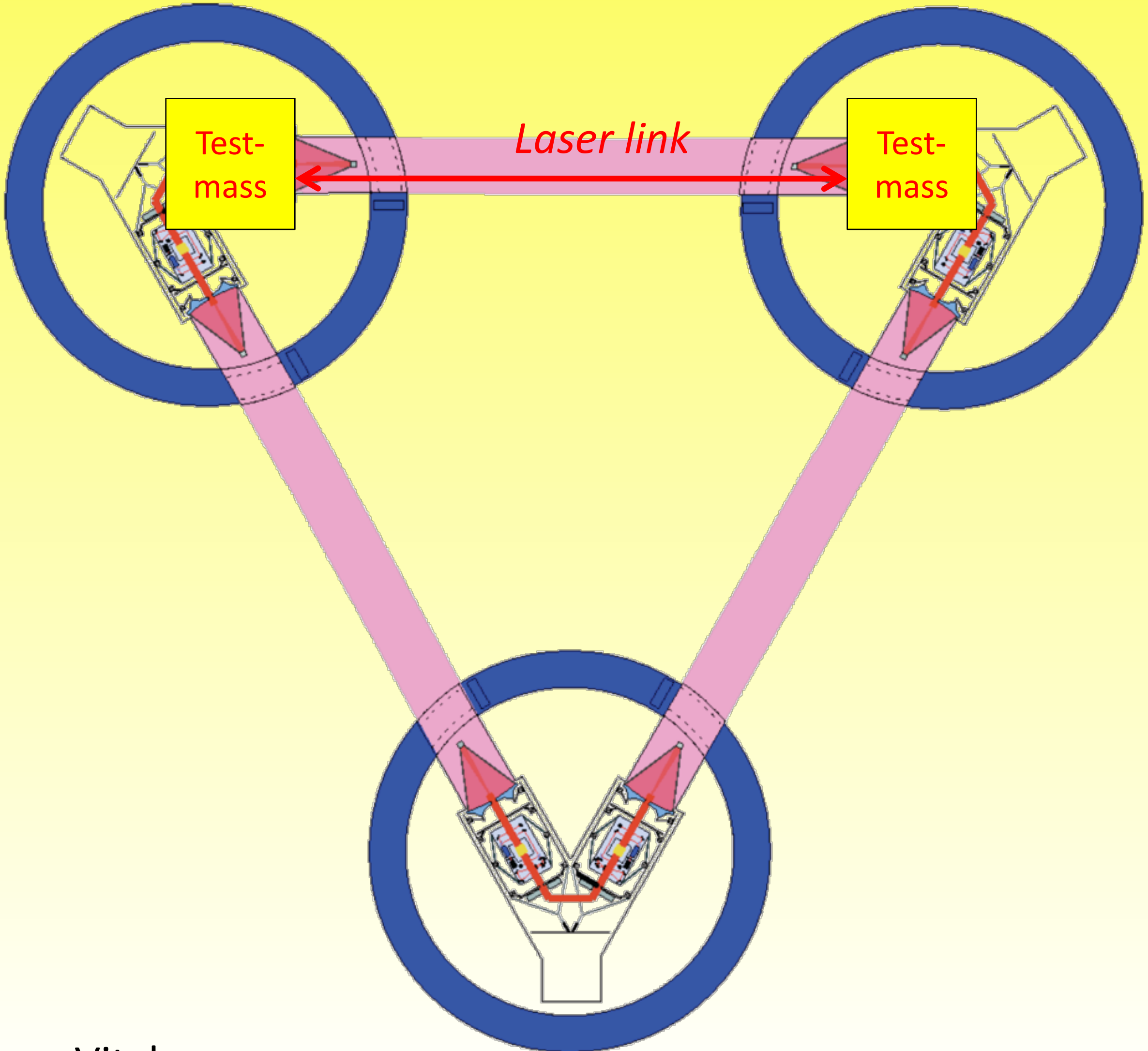




LISA Pathfinder

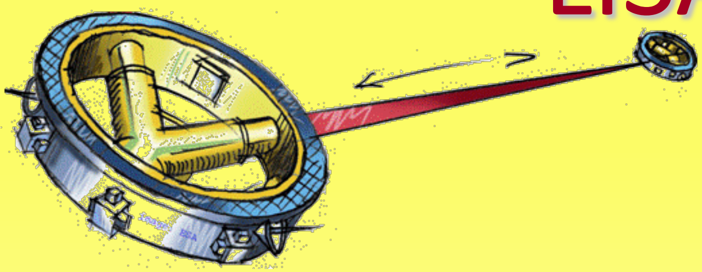
- Testing LISA technology in space!

One LISA Arm:
Few Million kms – two test masses



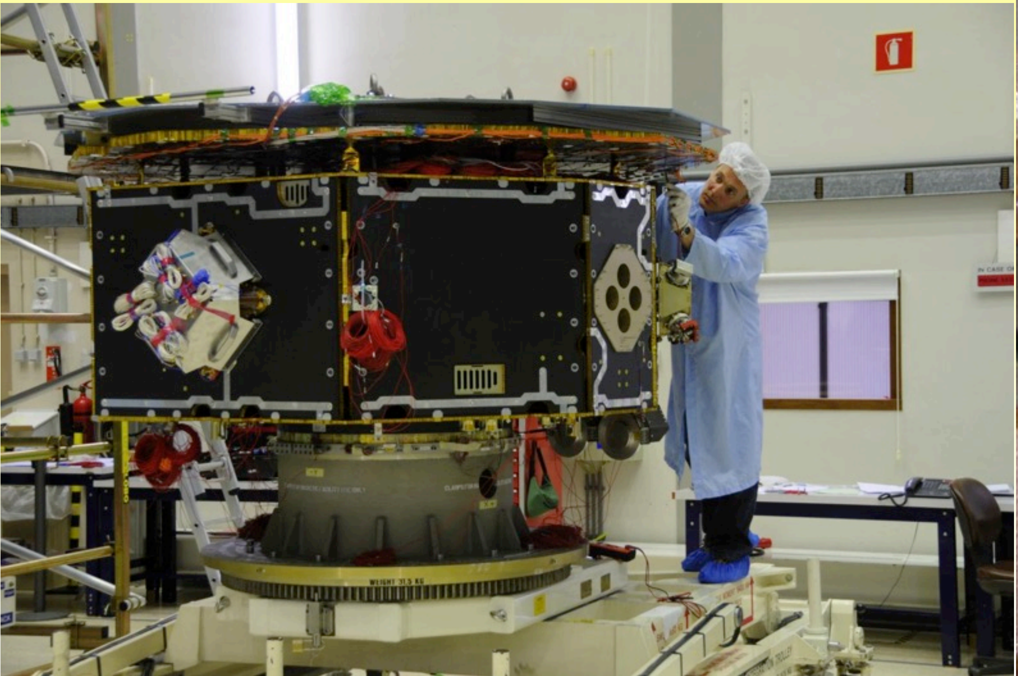
Courtesy: Stefano Vitale

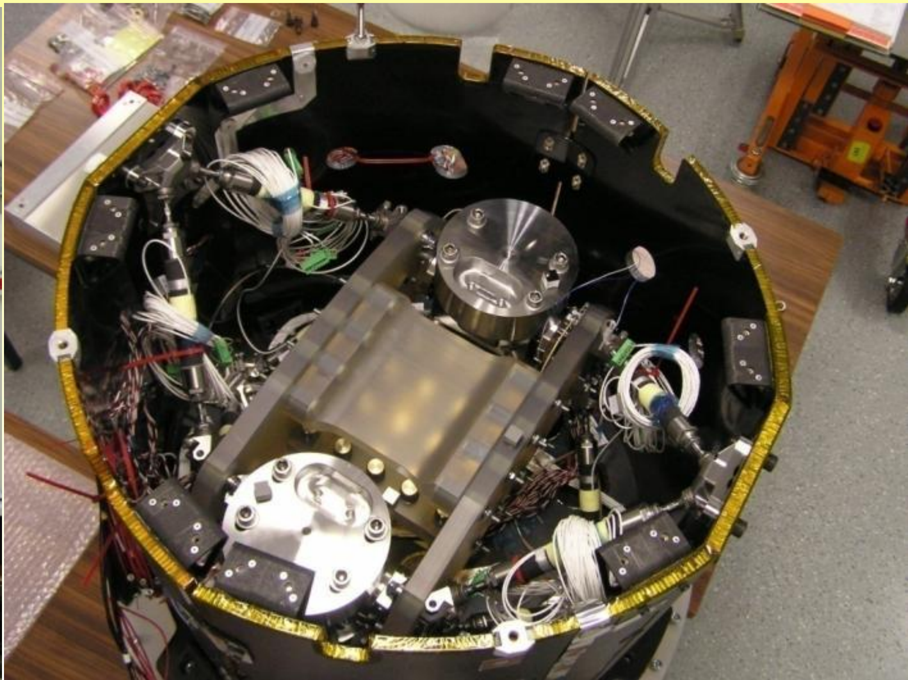
LISA Pathfinder



- Take one LISA arm
- Squeeze it into ONE satellite

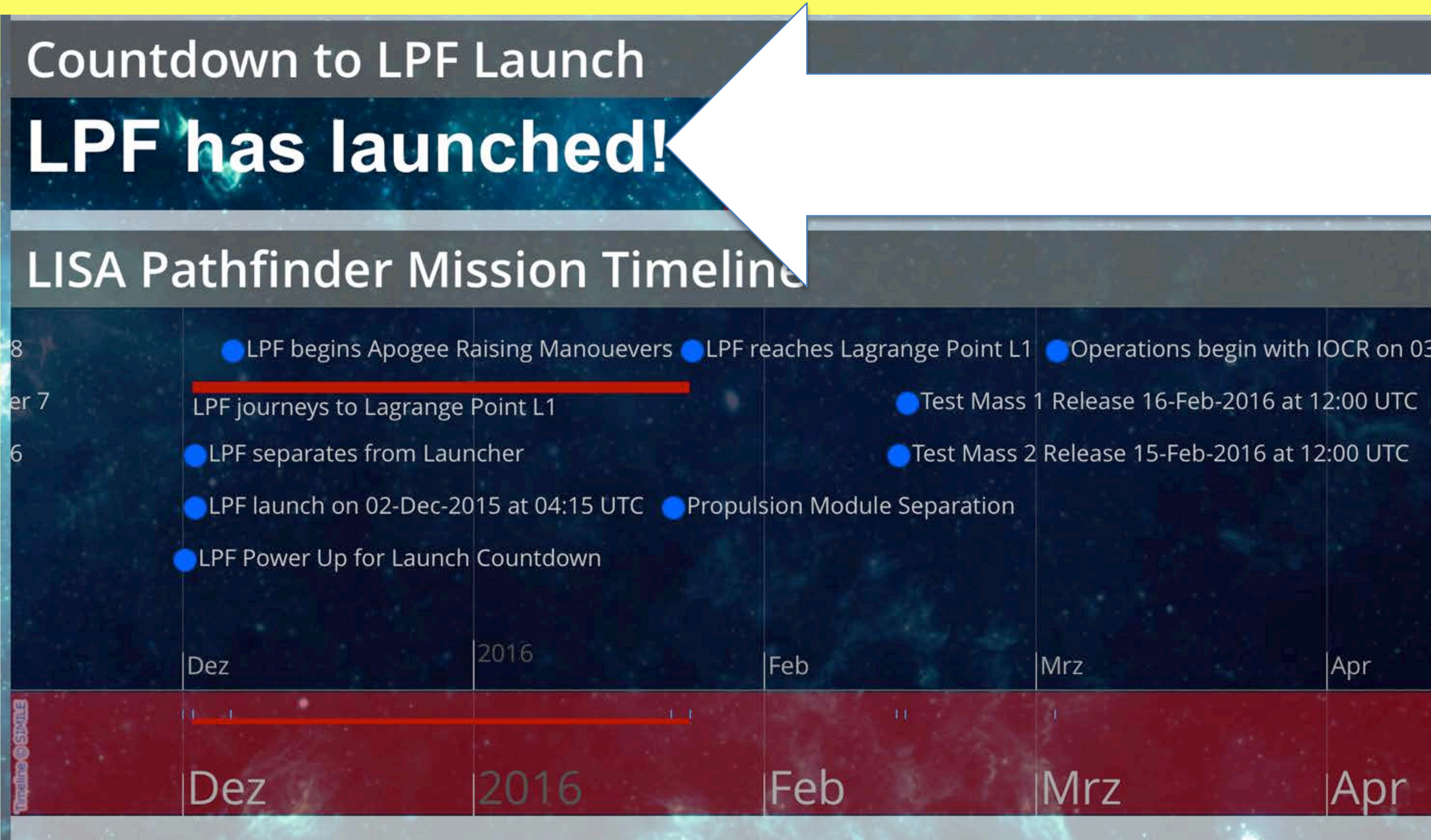
Courtesy: Stefano Vitale





September 2015: Spacecraft is completed!

100 Years since GR Publication: Dec. 2, 2015

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Articles published week ending 10 JUNE 2016

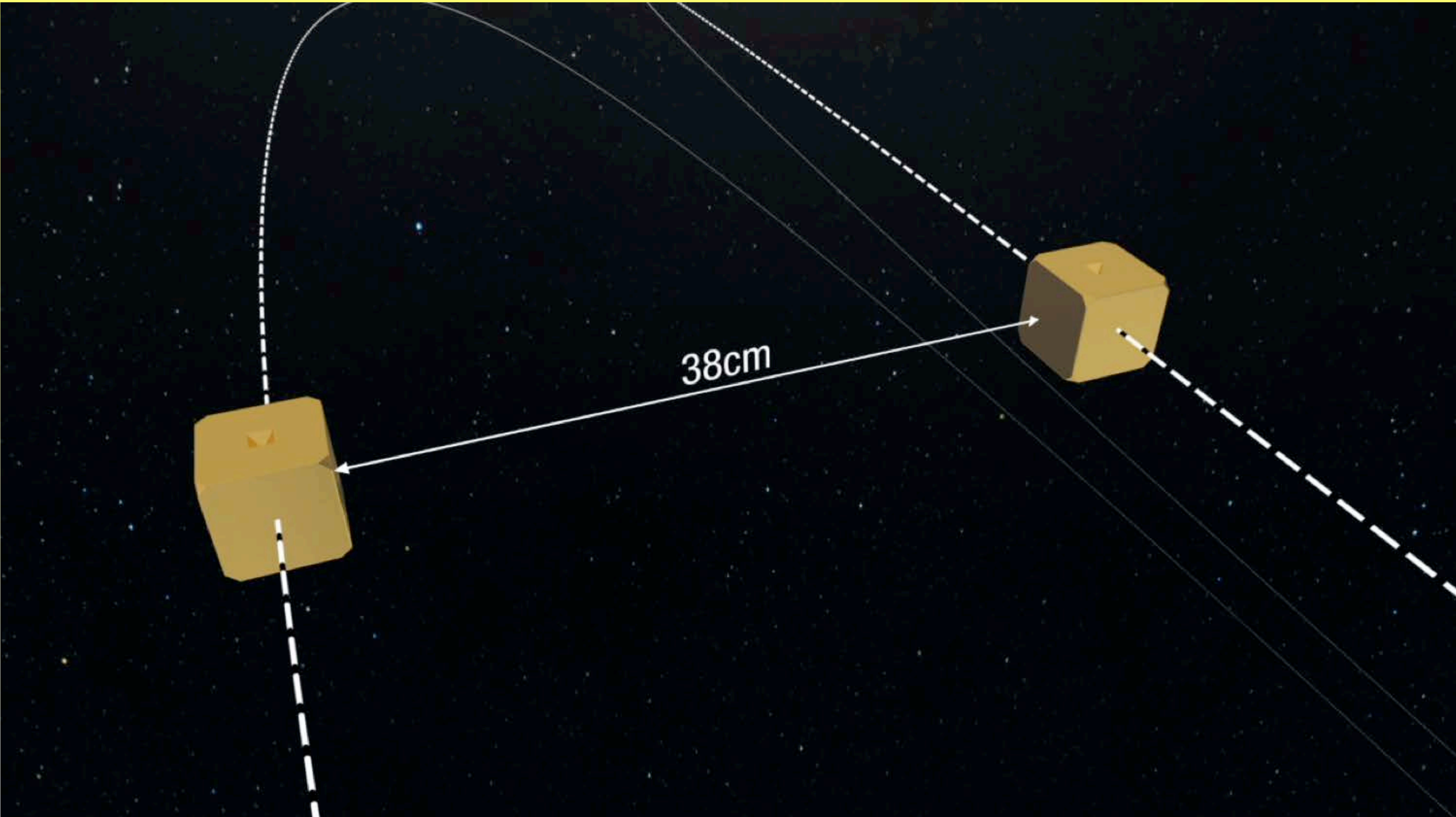
Published by
American Physical Society

APS physics

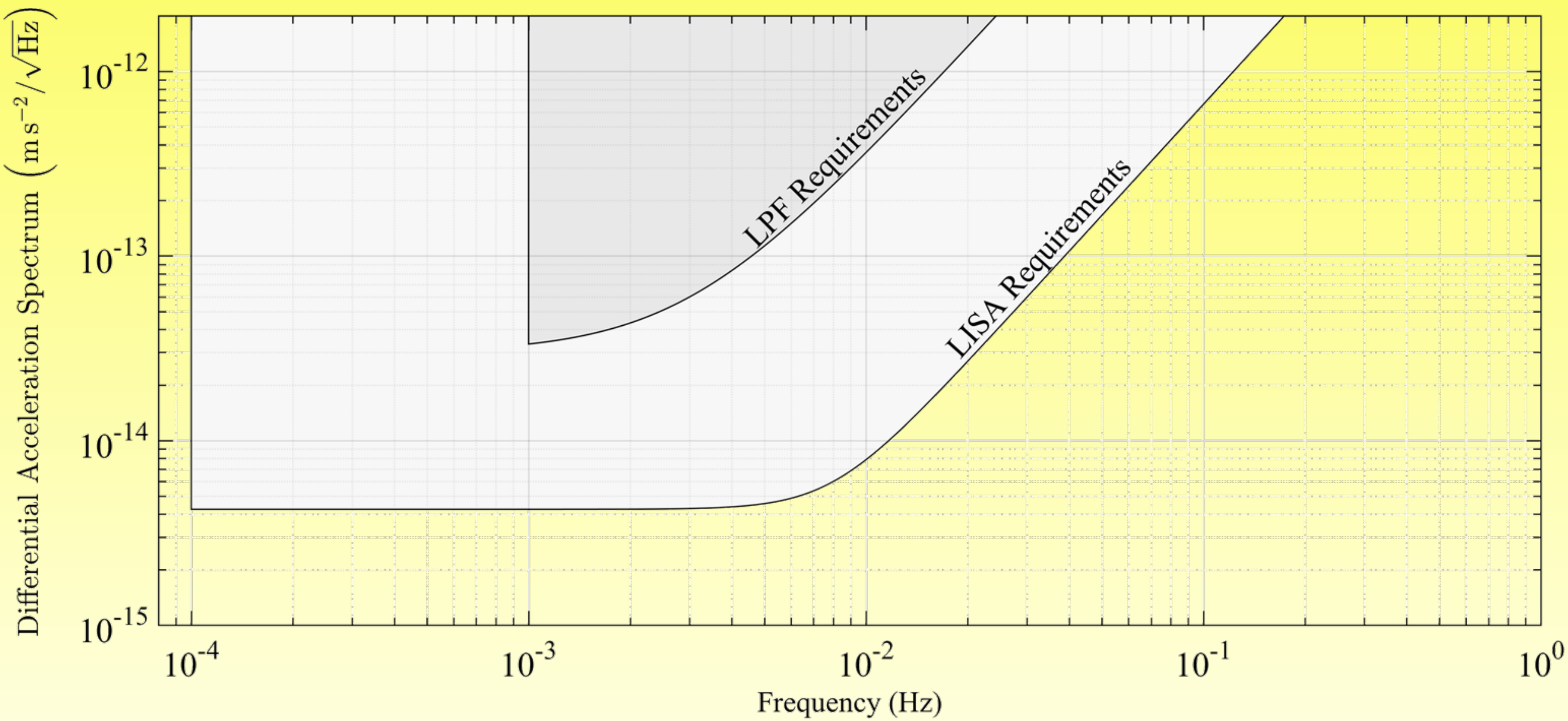
Volume 116, Number 23

The Stillest Place in the Universe!

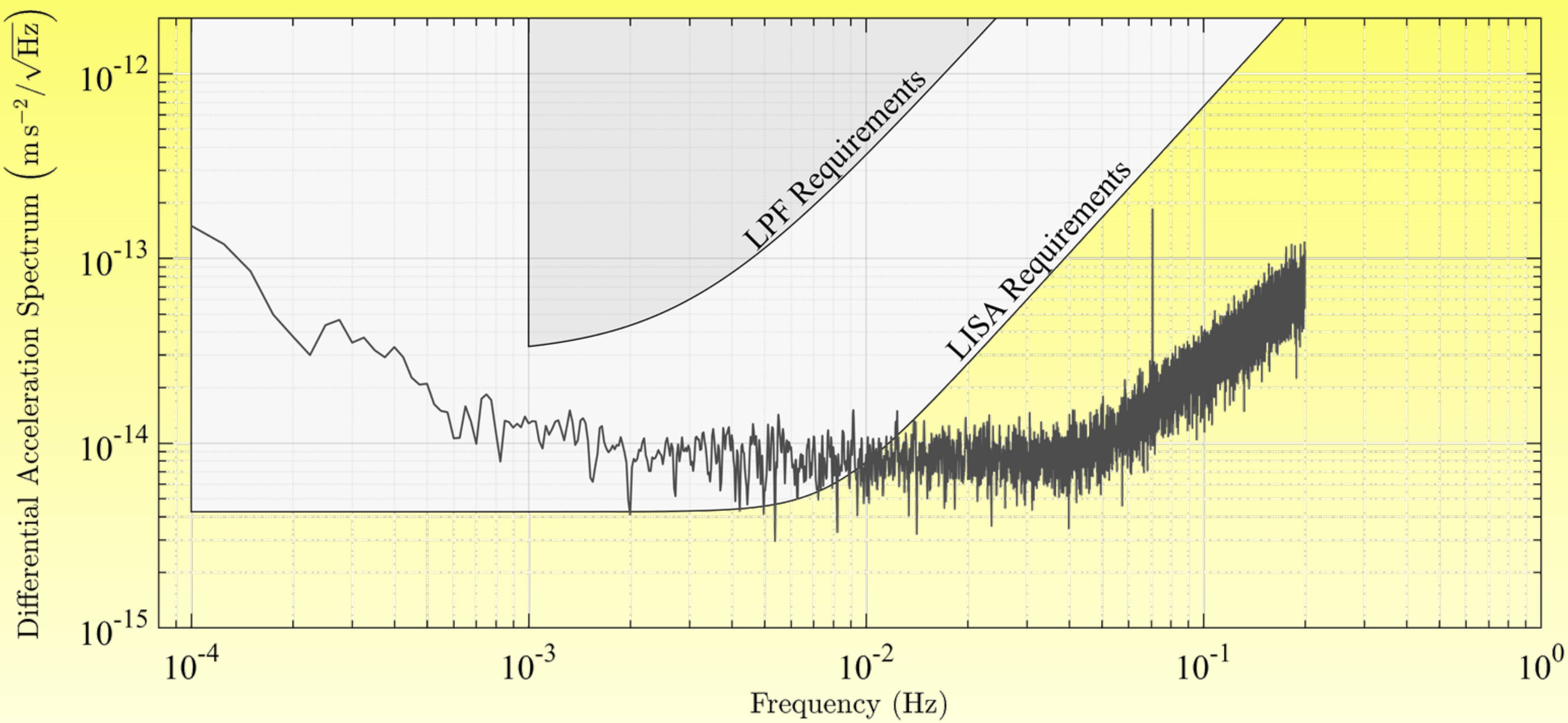
- More sensitive than the weight of a virus!



LISA and LPF Requirements



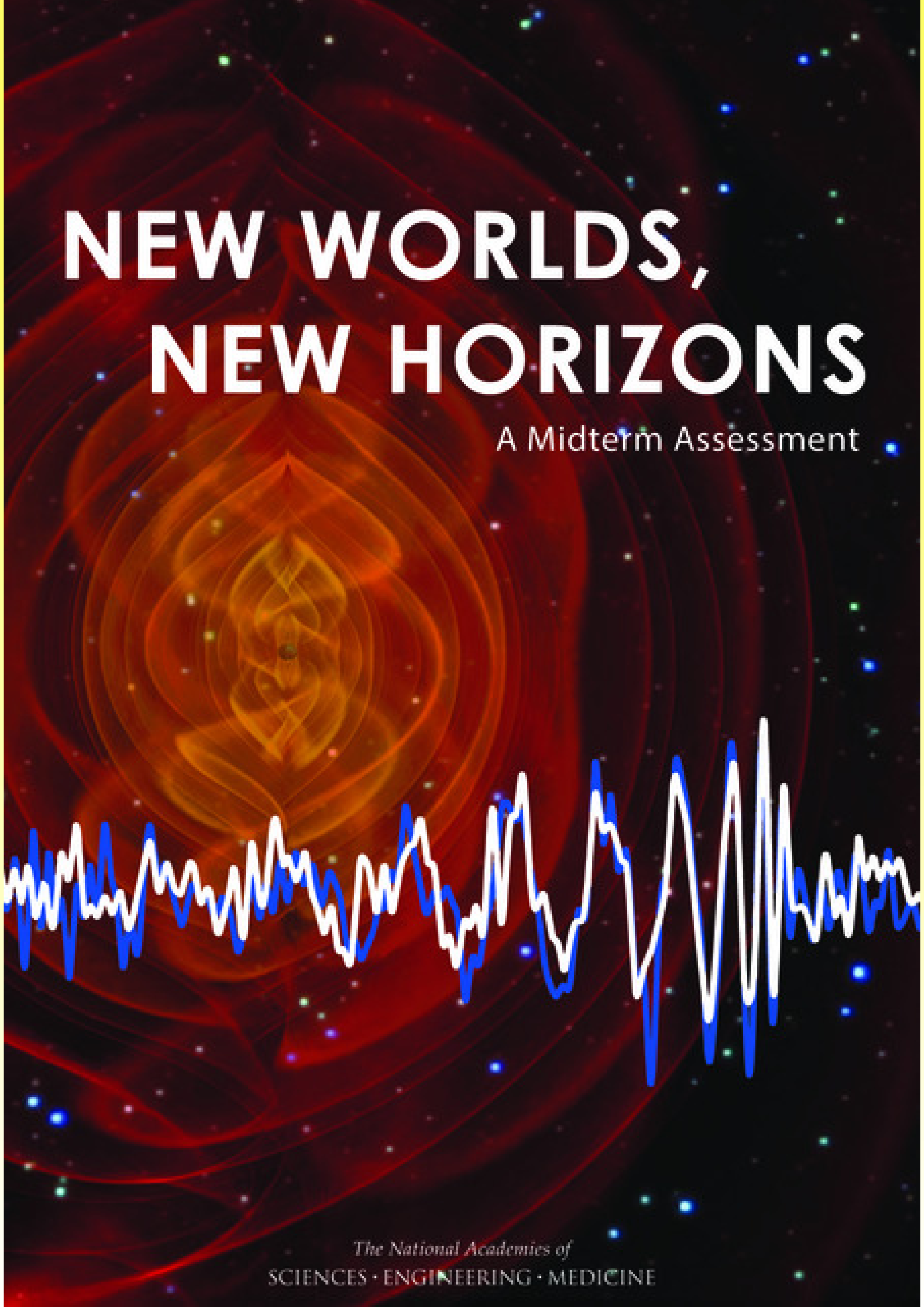
First Day of Operations: March 1, 2016



ESA L2 and L3 Missions

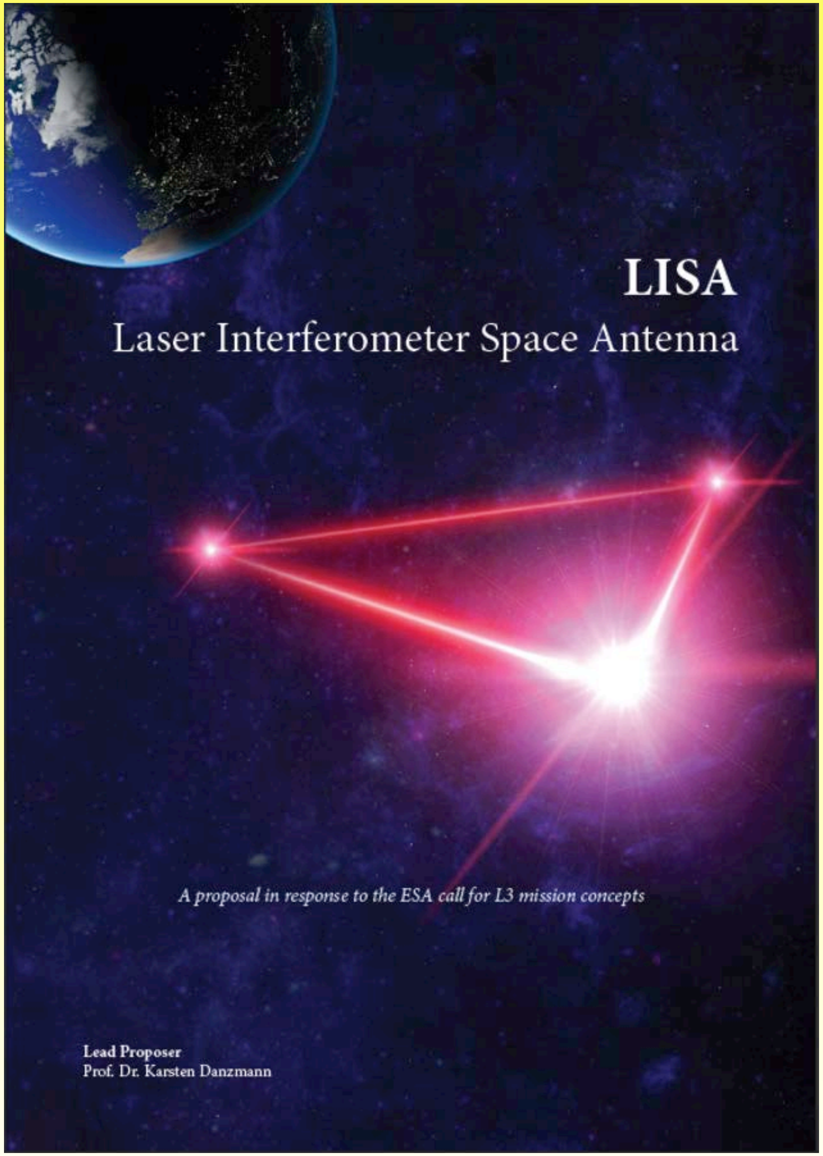
- Call for Mission Concepts fall 2016

NASA is back in LISA!



LISA Mission Concept Document

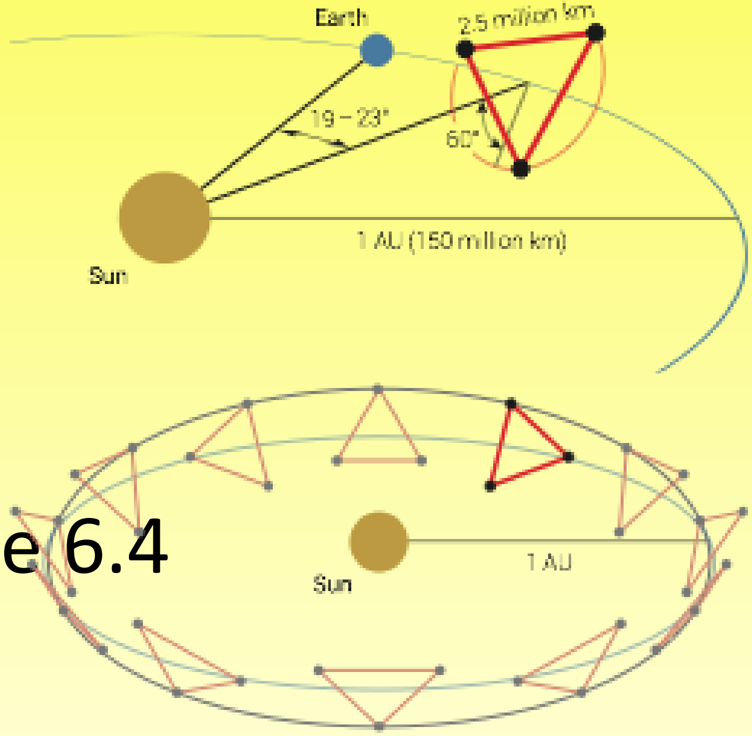
- Submitted on January 13th, 2017
- The LISA Consortium: 12 EU Member States plus the US !



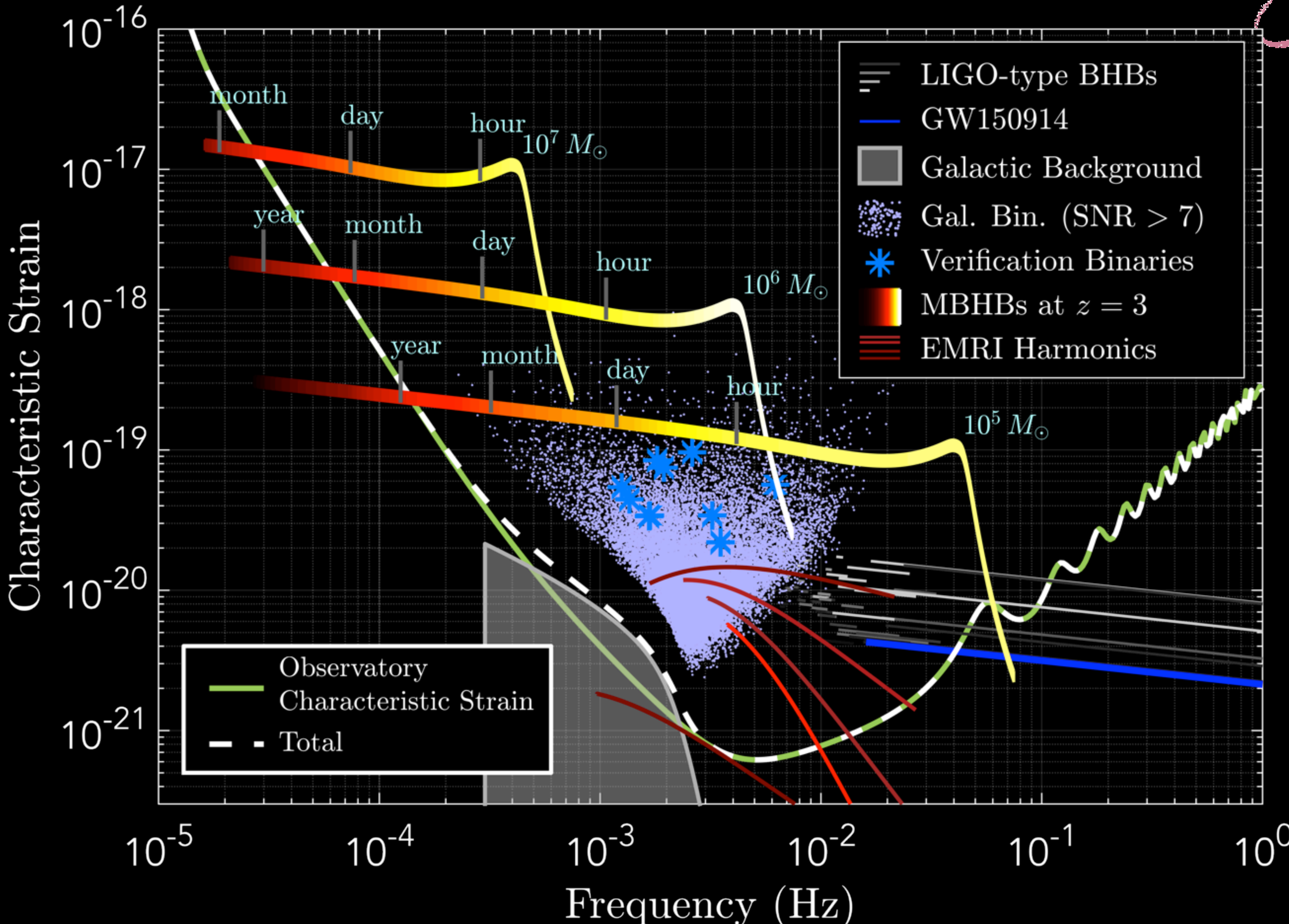
https://www.lisamission.org/proposal/LISA.pdf

Mission Profile and Orbit

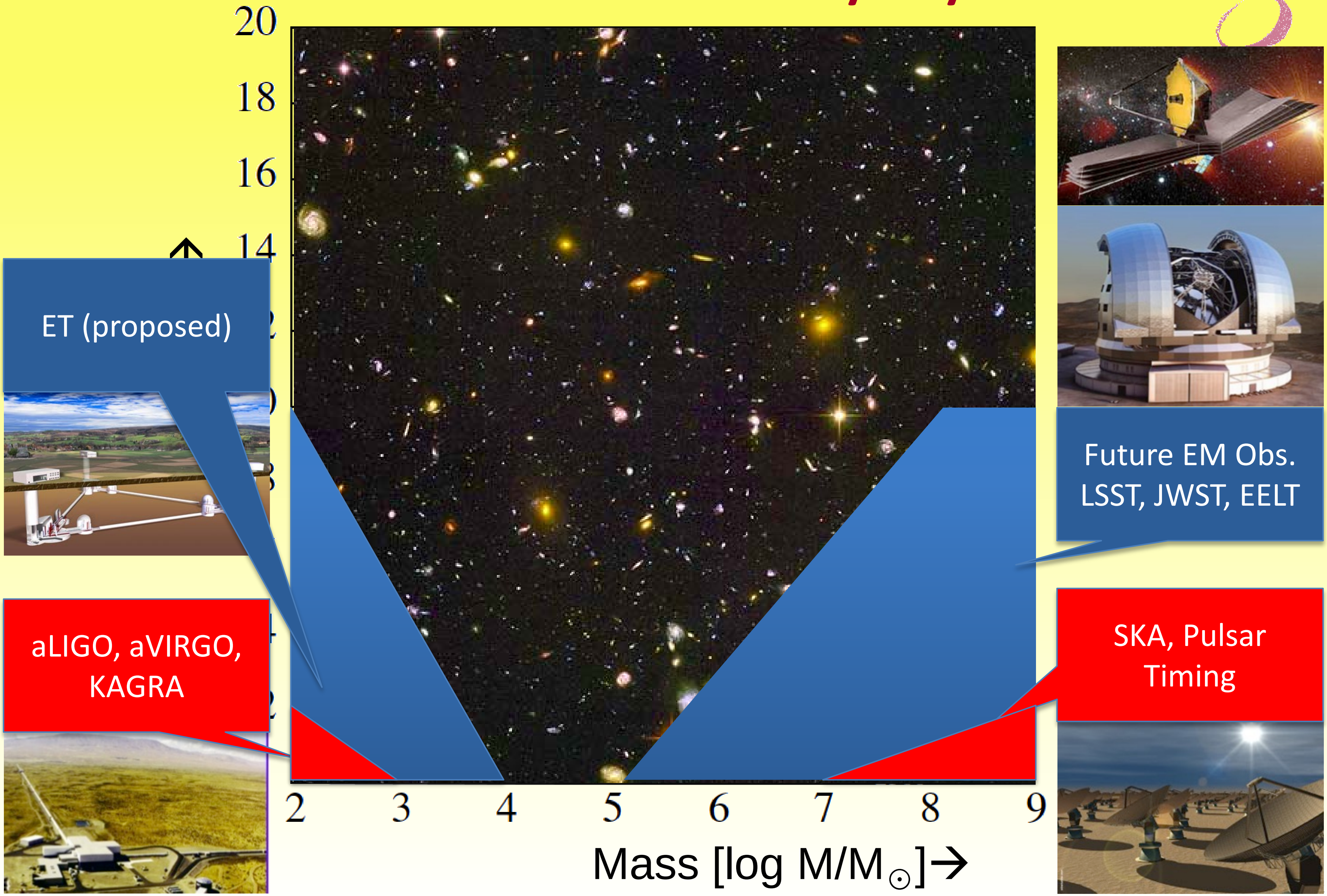
- Three arms of 2.5 Million km
- 2W lasers
- 30 cm telescopes
- Breathing angles ± 1 deg
- Doppler shifts ± 5 MHz
- Launch on dedicated Ariane 6.4
  - Transfer time ~400 days
  - Direct escape V_∞ = 260 m/s
  - Propulsion module and S/C composite



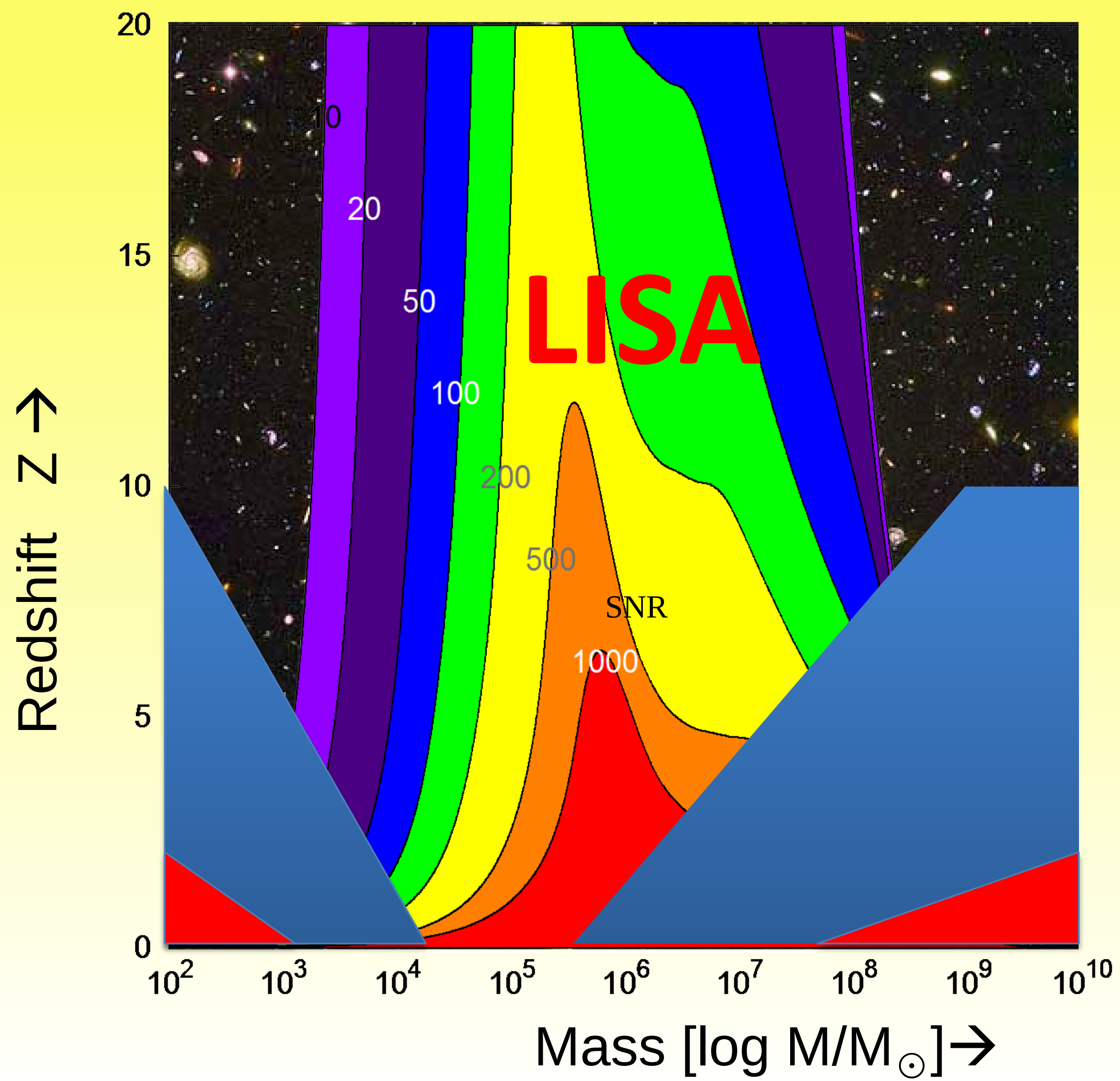
LISA Sources



Black Hole Astronomy by 2030

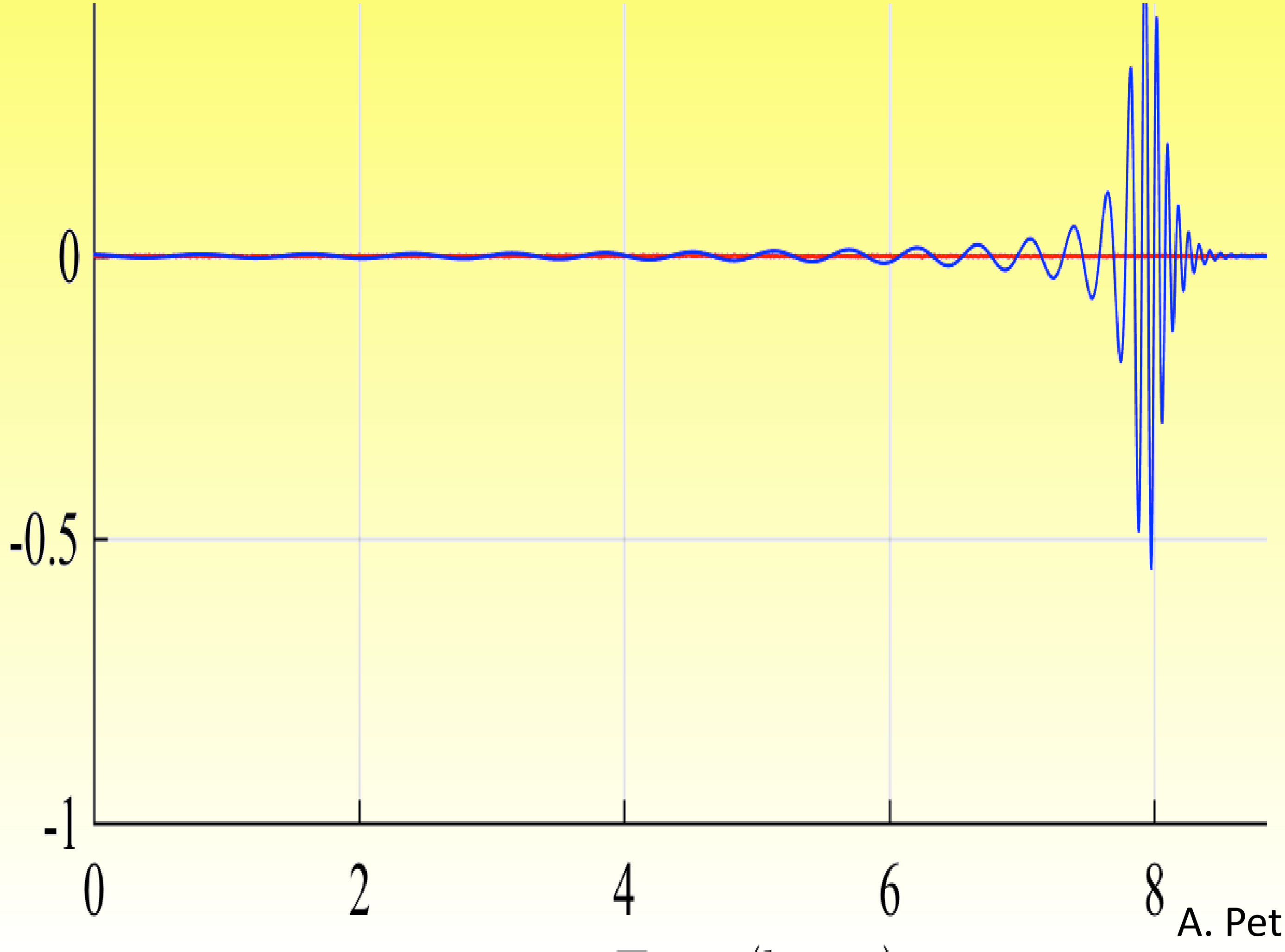


Black Hole Astronomy by 2030



Black Hole Mergers far above Noise

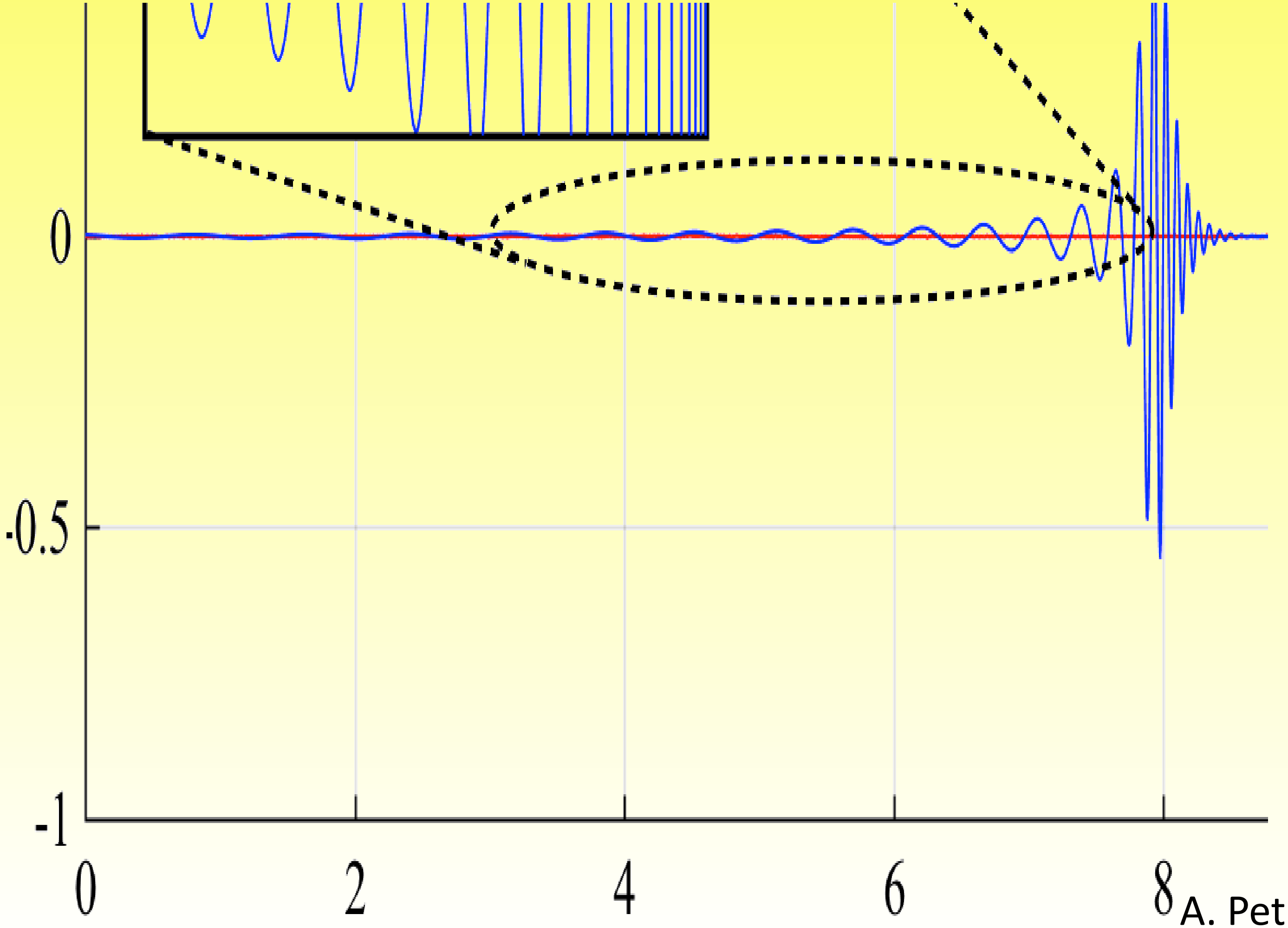
- $10^5\,M_\odot$ BH binary merger at z=5
- In Red: Pathfinder instrumental noise



A. Petiteau 2016

Black Hole Merger far above Noise

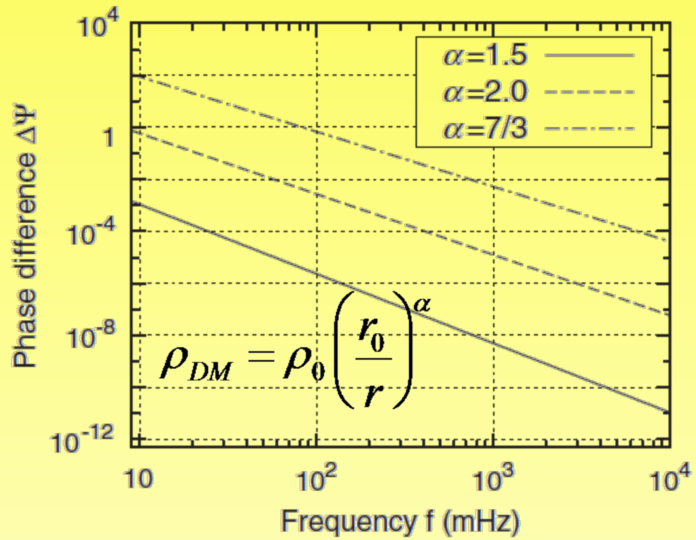
- $10^5\ M_{\odot}$ BH binary merger at z=5
- In Red: Pathfinder instrumental noise



A. Petiteau 2016

Dark Matter Probe

- Dark Matter spike around BH changes inspiral GW phase
- Sensitive even to Dark Matter interacting only gravitationally



PRL **110,** 221101 (2013) PHYSICAL REVIEW LETTERS week ending 31 MAY 2013

New Probe of Dark-Matter Properties: Gravitational Waves from an Intermediate-Mass Black Hole Embedded in a Dark-Matter Minispike

Kazunari Eda,* Yousuke Itoh, and Sachiko Kuroyanagi

Research center for the early universe, School of Science, University of Tokyo, Tokyo 113-0033, Japan

Joseph Silk

Institut d' Astrophysique, UMR 7095, CNRS, Université Pierre et Marie Curie Paris VI, 98 bis Boulevard Arago, Paris 75014, France

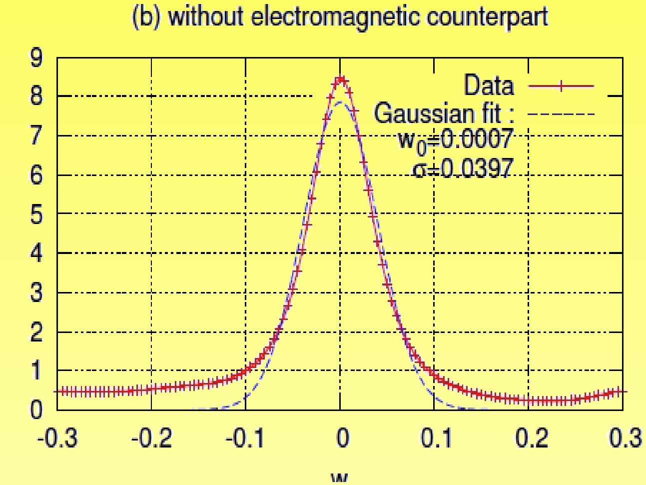
Cosmology with Standard Sirens

- With luminosity distances, LISA gives accurate and independent measurements of H_0 and w.

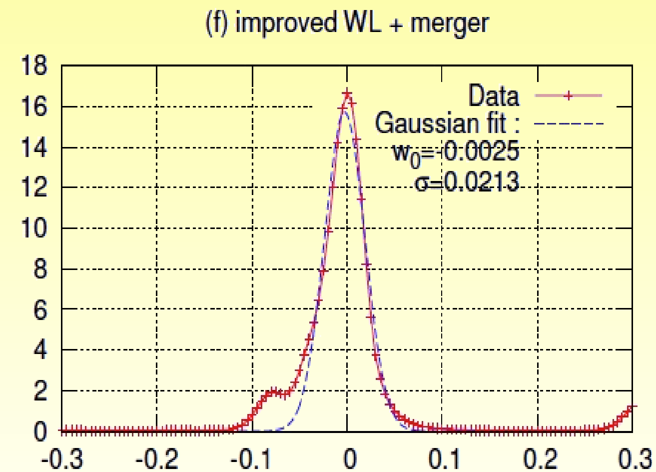
- **Using EMRIs, *without* identifications, LISA can determine H_0 to ±0.4% = ±0.3 km s^{-1} Mpc^{-1} after just 20 EMRI detections: ~3 months LISA data. (MacLeod & Hogan, PRD, 2008; SDSS) Today (WMAP) ±1.2 km s^{-1} Mpc^{-1}.**

- **Using massive mergers out to z = 3, again with *no* identifications, LISA can (in 3 years) determine dark energy equation of state parameter w to ±2-4%. (Petiteau et al, ApJ, 2011; Millennium). Compare EUCLID ±2%.**

No identifications



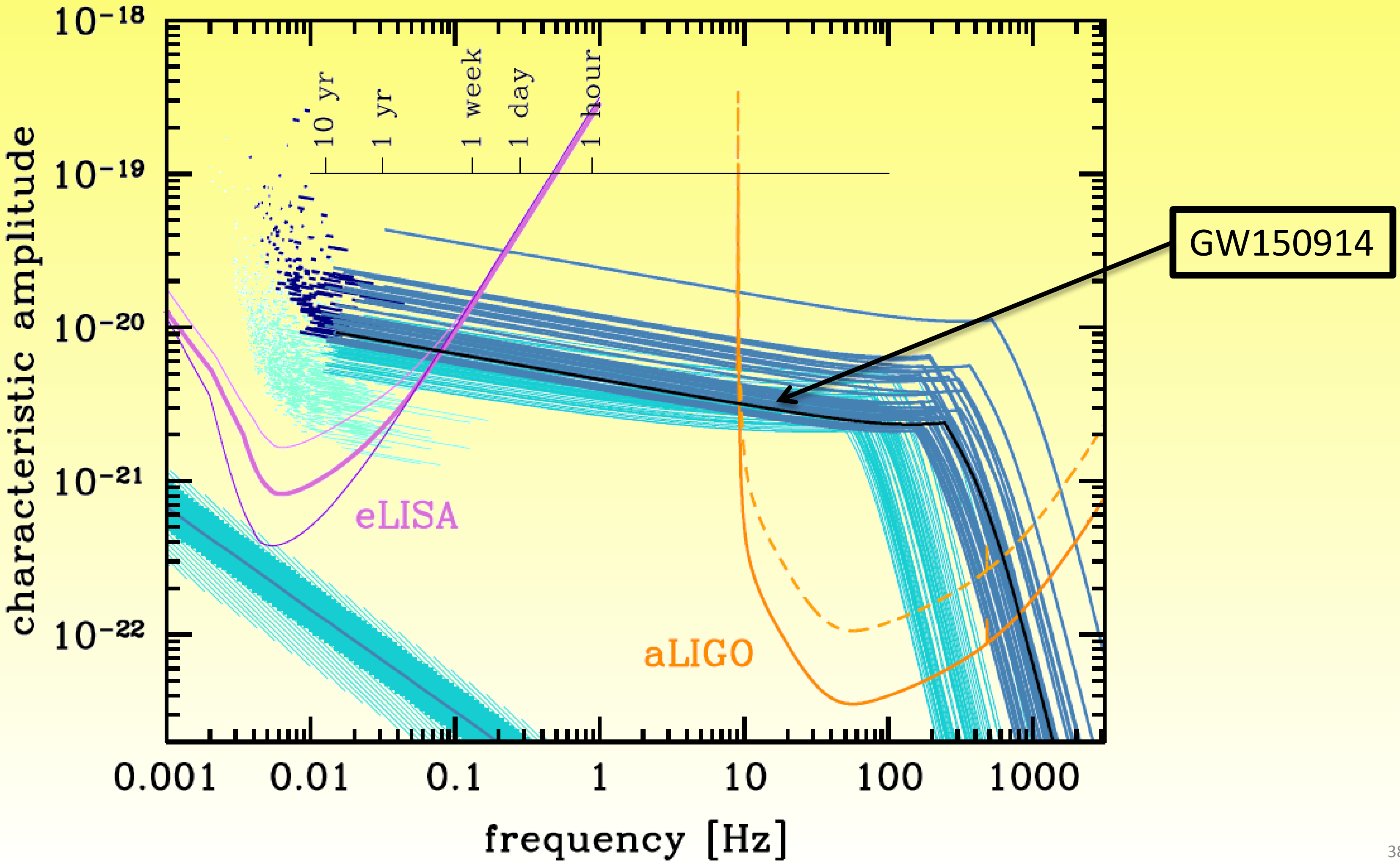
With identifications



Dark Energy equation-of-state parameter w

LISA: LIGO Event Predicted 10 Years in Advance!

- Accurate to seconds and within 0.1 square-degree!



ESA L2 and L3 Missions

- Call for Mission Concepts fall 2016
- Decision on Implementation 2020
- Launch of L2 in 2028
- Launch of L3 in 2034
- LISA shall be ready for an early launch!

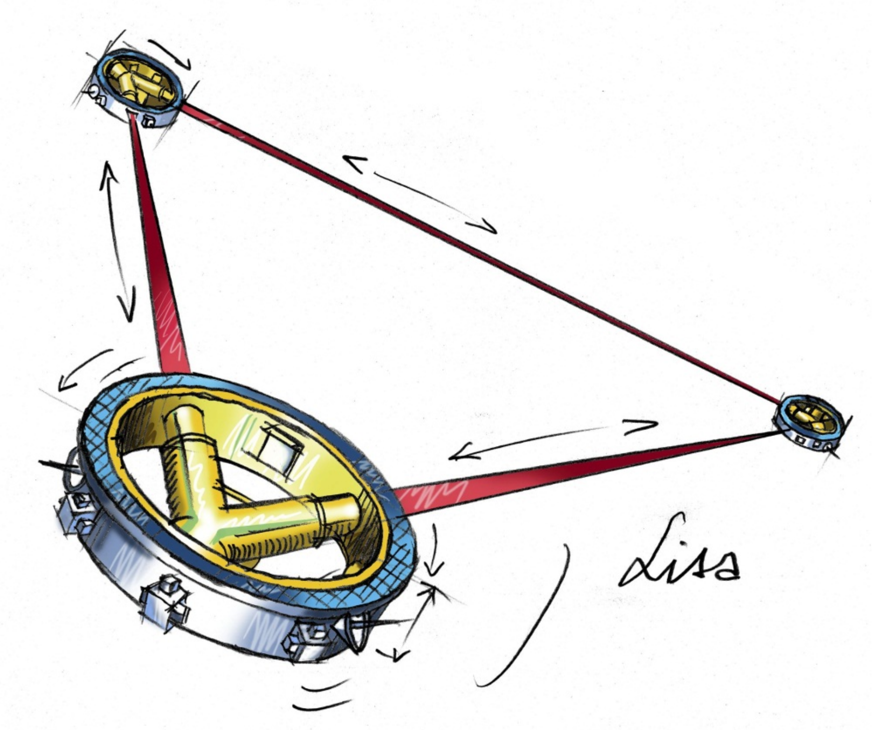
The End

LISA

Systems

Internal Final Presentation
ESTEC, 05-05-2017

Prepared by the CDF* Team



(*) ESTEC Concurrent Design Facility

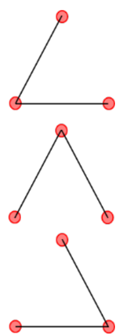
MISSION BACKGROUND

- LISA Mission concept has been around for a long time:
  - First ideas and studies date from 1974
  - First LISA-like proposal: LAGOS 1981
  - Mission studies: 1993 (ESA M3), 1996 (NASA)
  - Evolved into joint LISA study later -> until 2010 (ADS in Europe)
  - EU LISA CDF study in 2011
  - EuLISA/NGO for ESA L1 selection in 2012
  - LISA proposal in 2017

- Most of the architectures proposed in the past are based on a constellation of three spacecraft using laser interferometry in an Earth trailing orbit
- This has been the starting point for the CDF study, taking as reference the proposal of 2017

MISSION GOALS

- The goal of the mission is to detect and observe Gravitational Waves (GW)
- Laser Interferometry used to detect minute distance variations between free flying Test Masses (TM)
- Spacecraft required to “shield” the TM from external perturbations (SRP, drag free control), internal perturbations to be minimised (EMC, mass balance, thermal,…)
- Three arms required to determine origin and polarization (redundancy)



- Measurement broken into three legs:

- Expected variations are a few picometers, 1 pm = 10^{-12} m, sub atomic!

Science acquisition architecture fixed

MISSION REQUIREMENTS (KO)

Mission Constraints	
Req. ID	**Statement**
CONS-010	Mission costs: the ESA CaC less than 1050 M€ (2014 e.c.)
CONS-020	The mission shall be launched before 2034 TBC
CONS-030	TRL 6 shall be achieved by all elements before mission adoption (2024 TBC)
CONS-040	The mission shall be compatible with a launch on Ariane 6.4 from Kourou
CONS-050	Back up launcher shall be identified (not restricted to European launchers)

Payload Requirements	
Req. ID	**Statement**
PAY-010	The payload shall be identical in all three spacecraft
PAY-020	The payload shall consist of: Telescope Optical Bench Gravitational Reference Sensor Phase meter Diagnostics Package Data Procesing Unit Laser system
PAY-030	The total mass of the payload shall be lower than **360 kg**, including margins
PAY-040	The total power consumption of the payload shall be lower than **370 W**, including margins, during science operations
PAY-050	The payload data generation per SC rate shall be lower than **9503 bits/s**
PAY-060	The overal dimensions of the payload shall be under 2150, 1500, 900 mm
PAY-070	The payload shall be thermally isolated from the SVM

Mission Requirements	
Req. ID	**Statement**
MIS-010	The mission shall consist of three identical spacecraft
MIS-020	The mission shall perform laser interferometry in three independent interferometric combinations (3 arms)
MIS-030	The mission shall be designed for an in orbit lifetime of 6.5 years
MIS-040	The consumables shall be sized for a science phase of 10 years
MIS-050	The frequency band of the observatory shall be f =[0.1mHz, 0.1Hz], with a goal of f=[0.02mHz, 1Hz]
MIS-060	The total effective displacement noise $S_{\mathrm{IFO}}^{1/2}$ in a one-way single link test mass to test mass measurement shall be $S_{\mathrm{IFO}}^{1/2} \leq 10 \cdot 10^{-12} \frac{\mathrm{m}}{\sqrt{\mathrm{Hz}}} \cdot \sqrt{1 + \left(\frac{2\ \mathrm{mHz}}{f}\right)^4}$ in f =[0.1mHz, 0.1Hz], with a goal of f=[0.02mHz, 1Hz].
MIS-070	The total effective displacement noise $S_a^{1/2}$ in a one-way single link test mass to test mass measurement shall be $S_a^{1/2} \leq 3 \cdot 10^{-15} \frac{\mathrm{m\,s}^{-2}}{\sqrt{\mathrm{Hz}}} \cdot \sqrt{1 + \left(\frac{0.4\ \mathrm{mHz}}{f}\right)^2} \cdot \sqrt{1 + \left(\frac{f}{8\ \mathrm{mHz}}\right)^4}$ in f =[0.1mHz, 0.1Hz], with a goal of f=[0.02mHz, 1Hz].
MIS-080	The mission shall allow the collection of science data with an availability of at least TBD during nominal science phase
MIS-090	The missions shall allow to re-plan scheduled interruptions in case of a predicted merger event by moving such interruptions by, as a minimum, 2 TBC days.

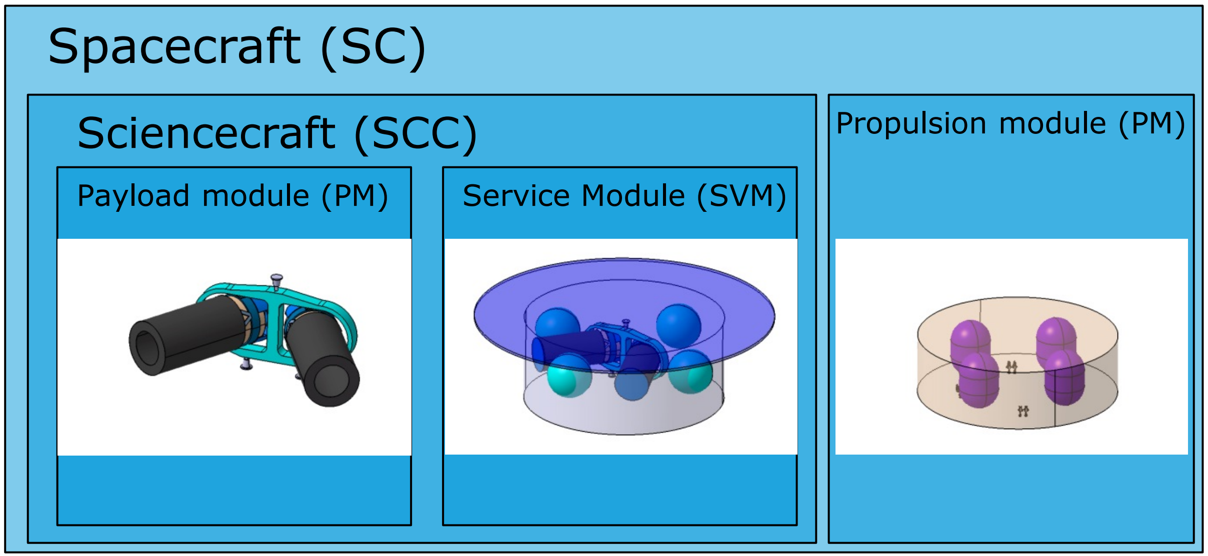
MISSION DRIVERS

- Science acquisition scheme drives the orbit selection and distance to Earth -> comms
- Demanding payload:
  - Thermal stability -> all systems on, internal configuration, sun shield
  - Thermal ranges -> heating power
  - Mechanical stability -> mechanisms to be avoided, no reaction wheels
  - High mass (474 kg w/o system margin)
  - High power (~600W w/o system margin)
  - High data rate (51 kbps for transmission, 800 kbps for storage, full constellation) -> comms
  - Large volume for main assembly -> driving sun shield and internal configuration
  - Integration of payload elements within the service module

MISSION DRIVERS

- Science data availability -> minimize interruptions, constellation acquisition
- Lifetime for the mission is 6.5 years, i.e. design shall be compatible with that duration and equipment qualified for that, but science extension of 6 years shall be considered for consumables:
  - Limited impact on solar panel
  - Significant impact on cold gas mass -> overall mass, configuration
- 3 spacecraft in a single launch -> either cylindrical (with or without propulsion stage) or trapezoidal configuration with dedicated spacecraft dispenser
- Stable thermal environment -> clean configuration wrt sun while in science mode, no elements shadowing solar array, isolation of solar array from sciencecraft

Mission Architecture



- Spacecraft dispenser

LAUNCHERS

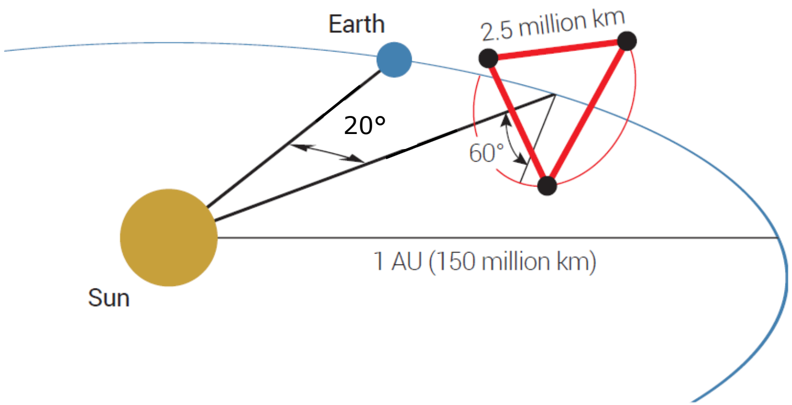
Possible launchers	Mass at launch	Design Load Factors		Frequency req.		Max fairing diameter (m)	Compliance	Cost
		Longitudinal (g)	Lateral (g)	Longitudinal (Hz)	Lateral (Hz)			
Ariane 6.4	7000 kg	-6/+2.5	±2	≥ 20 Hz	≥ 6 Hz	Ø = 4.572	Baseline	-
Atlas V551	6080 kg	-2/+6	±2	≥ 15 Hz	≥ 8 Hz	Ø = 4.572	Marginally compliant	US$ 135-185 Million
Falcon heavy	12365kg	-2/+6	±2	≥ 25 Hz	≥ 10 Hz	Ø = 4.6	Compliant	90M $
Proton M	Bellow 6475 kg**	-5.2/+3.8	±2	≥ 25 Hz	≥ 8.5 Hz	Ø = 4.35	Marginally compliant	US$ 90-100 Million
Delta IV Heavy	10140 kg	-2/+6	±2	≥ 30 Hz	≥ 8 Hz	Ø =4.572	Compliant	US$ 150-400 Million
Vulcan	10140 kg*	Not available	Not available	Not available	Not available	Ø = 4.572	Compliant	Not available
New Glenn	Not available	Not available	Not available	Not available	Not available	Not available	TBC	Not available

*Vulcan expected to achieve the current capability of the Delta IV Heavy (10140 kg) - Upgrade of Atlas V and Delta IV

**Complex Earth escape operations

ORBIT

- Three SC required in free flight forming an equilateral triangle, no actuation during science mode (except drag free control)
- Low perturbations environment required to achieve performances and limit the constellation deformation and fuel
- No need to keep rigid geometry, though range rate (Doppler) and breathing angle (optics/mechanisms) shall be limited
- Long mission duration, minimum of 4 years of science operations
- High data volume generated, remain in the vicinity of the Earth

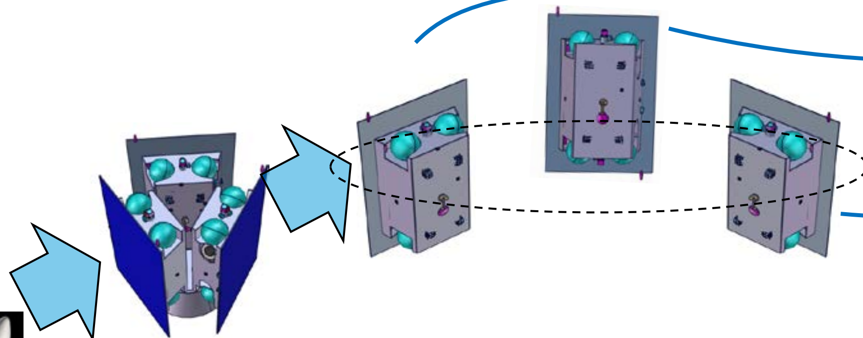


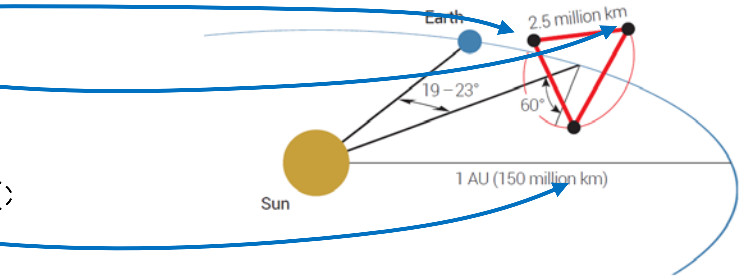
Orbit parameters	
Initial displacement angle (IDA)	20 deg
Distance to earth	50-65 million km
Arm length of constellation	2.5 million km
Inclination of constellation wrt ecliptic	60 deg
Corner angles	60 deg
Round trip time for comms	433 s
Earth azimuth and elevation during science	Az=360 deg; El=-9.35±3 deg
Arm length variation	±35000 km
Arm length variation rate	<10 m/s
Breathing angle	±0.9 deg
Breathing angle rate	5 nrad/s

ENVIRONMENT

- LISA will operate in a special but comparably well known environment
- Radiation
  - TID and TNID levels are moderate and similar to GEO missions
    - About 30% higher for extended mission
    - TNID hard to shield → identify and analyze sensitive items asap
  - Short-term SEE quasi identical for GEO and nom./ext. scenario (peak flux)
  - Long-term SEE similar to GEO missions but higher fluence for ext. scenario (up to ~90%)
- Micrometeoroids (prelim.)
  - Attitude disturbances
    - Considerable simplifications (e.g. no additional momentum by ejecta - up to 20 times larger momenta, IADC-2008-03) → needs further investigation
    - Significant number of “loss of laser pointing” (1urad)
  - Penetration risk
    - High risk for structure (100%) and CP tanks (52/76%) →further investigation
    - Need to shield tanks anticipated

MISSION PHASES





Launch in stacked configuration
Direct injection into escape trajectory

Separation of the stack right after launch

Separate trajectory for each S/C to final orbit

MISSION PHASES

Phase	Duration	Activities/Comments	System Mode
Pre-Launch (PLAU)	Up to 2 years	• After acceptance review until fairing enclosure • Shall support purging of payload • S/C shall be compatible with shelf lifetime of 2 years	-
Launch and Early Operations Phase (LEOP)	~2 days	• After fairing enclosure until insertion into transfer trajectory • Compatible with standard ESA LEOP ground station network • Initial check out of the system	Launch Mode
Near Earth Commissioning Phase (NECP)	~TBD weeks	• NECP shall start immediately after LEOP • Until completion of initial commissioning of S/C, payload elements TBC	Transfer Mode / Thruster Firing Mode
Transfer Phase (TP)	1.5 years	• Should start together with NECP (latest after NECP completion) • Until S/C's have been inserted into constellation configuration (not acquired) • Navigation and orbital manoeuvres to achieve final orbit	Transfer Mode / Thruster Firing Mode

MISSION PHASES

Phase	Duration	Activities/Comments	System Mode
System Commissioning Phase (SCP)	9 months	• Shall start immediately after TP • Shall last no longer than 9(TBC) months • Instruments commissioning and first acquisition of constellation	Science Mode
Nominal Science Phase (NSP)	4 years	• Shall start immediately after SCP and shall be 4 years • Instrument data collection • Shall end with transition to ESP/DCP	Science Mode
Extended Science Phase (ESP, optional)	6 years	• NSP could be extended up to 6 extra years in order to increase the scientific return of the mission	Science Mode
Decommissioning Phase (DSP)	~TBD weeks	• Design and operations of mission shall comply with rules and procedures put forth in ECSS-U-AS-10C with exact measures towards compliance to be agreed by Agency • Consumables for Decommissioning shall be calculated for worst case required delta-V, identified in MAG, with margins according to Margin Philosophy • DCP shall be completed within two months after end of operational lifetime	Transfer Mode

SYSTEM MODES

Launch Mode	• Lift off to separation, all equipment OFF, except essential ones • Detumbling and sun acquisition right after separation
Transfer Mode	• Service Module ON, Payload OFF (thermally conditioned) • Communication TM/TC
Thruster Firing Mode	• Same as TM with transfer thrusters ON (CP or EP)
Science Mode	• All systems ON, 100% duty cycle • Communications TM/TC and science data • Payload sub modes (acquisition sizing)
Safe mode	• Safe mode, minimum set of equipment ON • Communications TM/TC through LGA • Sub modes TBD

DEFINITION OF SYSTEM OPTIONS

- Science data acquisition scheme fixed
- Baseline orbit selected (small modifications possible)
- Payload kept identical for all the options

- Three main options have been defined:
  - CP option, making use of a bipropellant chemical propulsion module for transfer and cold gas system for science, highest heritage from LPF
  - EP option, making use of integrated Electric Propulsion for the transfer, but still making use of cold gas system for science, heritage for science operations
  - EP+ option, making use of integrated Electric Propulsion for the transfer and Micro Electric Propulsion for science (miniRIT / FEEPs / Colloids), most optimised option in terms of mass and volume
- Several subsystems identical in all options

CP OPTION

- Classic configuration from past studies, sciencecraft + propulsion module
- Maximises heritage from LPF:
  - Chemical propulsion for transfer – full compatibility with the mission
  - Maximum reuse of payload (GRS, DFACS, micro propulsion system)
- Cylindrical configuration adopted (better symmetry), stack of 6 mission elements, 6 separations required
- PM structure supporting sciencecraft during launch (load path), discarded after transfer
- Payload volume and cold gas tanks drive the sciencecraft configuration, required sun shield of 4m diam (larger than required solar array) -> large amount of propellant
- Simpler sciencecraft, lower number of propulsion systems
- PM shields part of the service module elements (HGA, STR, SAS)
- Previous studies made use of electric micro propulsion instead of cold gas

EP OPTION

- Switches to EP technology for the transfer (EMC TBC)
- No dedicated propulsion stage used, no clear gain (low mass penalty of EP system), simplifies the separation sequence
- Trapezoidal configuration investigated, cylindrical configuration could be possible but would require supporting structures for launch, increasing again the number of separations
- Requires a dedicated payload dispenser for the launcher (SWARM like)
- Configuration driven by main payload assembly and cold gas tanks, size of the required sun shield similar to required solar array area
- Transfer phase becomes the sizing case for power (EP ON)
- Use of cold gas maintains heritage wrt LPF during science operations, but limits the lifetime extension capability (not much more than 10 years, depending on margin philosophy)

EP+ OPTION

- Evolution of previous option with a swap of cold gas micro propulsion for electric micropropulsion:
  - miniRIT (used as sizing case)
  - FEEPs
  - Colloids
- Relies on micro propulsion with low readiness level (final selection after detailed dedicated technology assessment)
- Offers a more compact and easier configuration (smaller number of propellant tanks, smaller size, shared by all thrusters), though larger solar array is required
- Requires dedicated thrusters for AOCS during transfer due to lower maximum thrust of EP micro propulsion thrusters
- Offers larger margin for increase of propellant load and wrt launcher capabilities (though not enough to fit in A6.2)

SUBSYSTEM SUMMARY

- Structures based on sandwich panels with reinforcements (longerons) for all the options. Dedicated secondary structure for payload accommodation
- Configuration driven by instrument main assembly, propellant tanks, solar array (EP and EP+) and sun shield (CP) requirements.
- Communications based on X band system (160W RF), LGAs for LEOP and safe modes, mechanical steering HGA (35cm) for transfer and science .Comms routed through spacecraft, one antenna rotation every two weeks (low gravity field imbalance due to rotation). PAA maintained as an option to be further investigated (final report)
- Data handling based on integrated unit (OBC, RTUs and MMU) connected to all different payload elements for TM/TC and time reference distribution/synchronisation (1553/CAN and SpW buses). OBC taking part of payload functionalities (at least 3X LPF computing capability). MMU sized for 1 month (256Gb)

SUBSYSTEM SUMMARY

- Power based on fixed solar array with current efficiencies. Sized for transfer in EP and EP+ options (2.3 and 2.5kW respectively) and science for CP (1.6 kW). Battery for launch support, orbital manoeuvres, safe mode (2.5kWhr, 60% DoD)
- Challenging thermal design based on active control, heaters and MLI. High heating power required during transfer
- Mechanisms, separation mechanisms based on clambands for CP option and hold points for EP and EP+. 2 dof pointing mechanism required for HGA. 2 dof thruster pointing mechanism for EP and EP+. In payload:
    - Telescope Pointing / In Field Pointing / both
    - Point ahead mechanism
    - Telescope cover
    - GRS mechanisms

SUBSYSTEM SUMMARY

- AOCS/DFACS large heritage from LPF. Payload as main sensor for science mode, improved star trackers for acquisition, gyros and sun sensors for transfer and safe modes. Dedicated sensor in telescope for constellation acquisition. 12 + 12 cold gas thruster for CP and EP, 6+6 electric micro thrusters for EP+. Need for dedicated thrusters for AOCS during transfer in EP+. Dedicated Xe cold gas thrusters for detumbling in EP and EP+.
- 4+4 22N thrusters for transfer supported by 4+4 10N thrusters RCS for CP propulsion module
- 4+4 50mN Xe cold gas thrusters for detumble in EP and EP+ options
- Electric propulsion based on PPS1350 for transfer (1+1).
- miniRIT used as sizing case for EP+. 6+6 100uN and 4+4 1000uN for AOCS during transfer (EP ON)
- Selection of micro electric propulsion at a later stage in the program

SYSTEM BUDGETS CP

PM Mass Budget		Margin	Mass [kg]
Attitude, Orbit, Guidance, Navigation Control		5.00	13.64
Chemical Propulsion		6.53	145.80
Mechanisms		20.00	40.80
Structures		20.00	331.61
Thermal Control		17.97	21.83
Harness	5%		27.63
Dry Mass w/o System Margin			**581.36**
System Margin	20%		116.27
Dry Mass incl. System Margin			**697.63**
CPROP Transfer Fuel Mass			397.72
CPROP Transfer Fuel Margin	2%		7.95
CPROP Transfer Oxidizer Mass			649.40
CPROP Transfer Oxidizer Margin	2%		13.12
CPROP Transfer Pressurant Mass			3.40
CPROP Transfer Pressurant Margin	2%		0.07
CPROP AOCS Propellant Mass			36.23
CPROP AOCS Propellant Margin	2%		0.72
Total Propellant Mass			**1115.46**
Total Wet Mass			**1813.09**

PLM Mass Budget		Margin (%)	Mass [kg]
Instruments		14.32	473.95
Dry Mass w/o System Margin			**473.95**

SVM Mass Budget		Margin (%)	Mass [kg]
Attitude, Orbit, Guidance, Navigation Control		5.00	24.55
Communications		5.00	25.75
Chemical Propulsion		18.10	168.99
Data-Handling		20.00	16.44
Electric Propulsion		0.00	0.00
Mechanisms		20.00	16.80
Power		5.78	108.51
Structures		20.00	142.62
Thermal Control		17.97	21.83
Harness	5%		26.27
Dry Mass w/o System Margin			**551.75**

S/C Mass Budget		Mass [kg]
Dry Mass PLM		**473.95**
Dry Mass SVM		**551.75**
System Margin	20%	205.14
Dry Mass incl. System Margin		**1230.83**
CPROP Cold Gas Mass		199.82
CPROP Cold Gas Margin	0%	0.00
Total Wet Mass		**1430.65**

Stack Mass Budget	Mass [kg]
ScienceCraft 1 Dry	1230.83
Propulsion Module 1 Dry	697.63
Total Spacecraft dry mass excl. Adapters	**1928.46**
Propulsion Sciencecraft 1	199.82
Propulsion Propulsion Module 1	1115.46
Total Spacecraft wet mass excl. Adapters	**3243.74**
Spacecraft target mass	2253.33
Below target mass	-990.41
# Of satellites	3.00
Launch Adapter	240
Total Stack Mass incl. Adapters	**9911.23**
Target Wet Mass incl. Adapter	7000.00
Below Target Mass by	-2971.23

CP RECOVERY

- Several options were investigated to recover the CP option.
  - Not having any margin on the DFACS cold gas propellant
  - Having only 4 years of operation, lower propellant need for science and for transfer
  - Using EP propulsion for the DFACS

Mass difference	Delta Mass/SC	Delta mass total
10 Years margin	-959	-2878
10 Years no margin	-721	-2163
4 Years	-148	-443
MiniRIT	-149	-446
MiniRIT 4 year	121	362

SYSTEM BUDGETS EP

PLM Mass Budget	Margin (%)	Mass [kg]
Instruments	14.32	473.95
Dry Mass w/o System Margin		**473.95**

SVM Mass Budget		Margin (%)	Mass [kg]
Attitude, Orbit, Guidance, Navigation Control		5.00	24.55
Communications		5.00	23.65
Chemical Propulsion		17.93	190.20
Data-Handling		20.00	16.44
Electric Propulsion		7.07	80.68
Mechanisms		12.64	45.06
Power		5.86	122.99
Structures		20.00	208.42
Thermal Control		17.97	21.83
Harness	5%		36.69
Dry Mass w/o System Margin			**770.49**

S/C Mass Budget		Mass [kg]
Dry Mass PLM		**473.95**
Dry Mass SVM		**770.49**
System Margin	20%	248.89
Dry Mass incl. System Margin		**1493.33**
EPROP Transfer Propellant Mass		145.00
EPROP Fuel Margin	2%	2.90
CPROP Cold gas fuel Mass		234.95
CPROP Fuel Margin	2%	4.70
Total Propellant		**387.55**
Total Wet Mass		**1880.88**

Stack Mass Budget	Mass [kg]
ScienceCraft 1 Dry	1493.33
Propulsion Sciencecraft 1	387.55
Total Spacecraft mass excl. Adapters	**1880.88**
# Of satellites	3.00
Launch Adapter	1000
Total Stack Mass incl. Adapters	**6642.63**
Target Wet Mass incl. Adapter	7000.00
Below Target Mass by	357.37

System budgets EP+

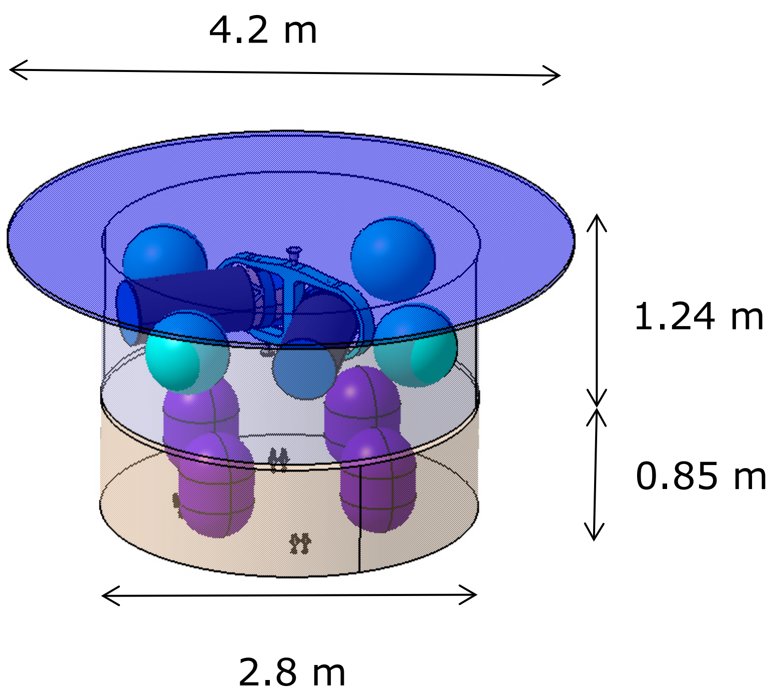
PLM Mass Budget		Margin (%)	Mass [kg]
Instruments		14.32	473.95
Dry Mass w/o System Margin			**473.95**

SVM Mass Budget		Margin (%)	Mass [kg]
Attitude, Orbit, Guidance, Navigation Control		5.00	24.55
Communications		5.00	23.65
Chemical Propulsion		17.93	4.41
Data-Handling		20.00	16.44
Electric Propulsion		7.07	170.69
Mechanisms		12.64	45.06
Power		5.86	132.76
Structures		20.00	208.42
Thermal Control		17.97	21.83
Harness	5%		32.39
Dry Mass w/o System Margin			**680.18**

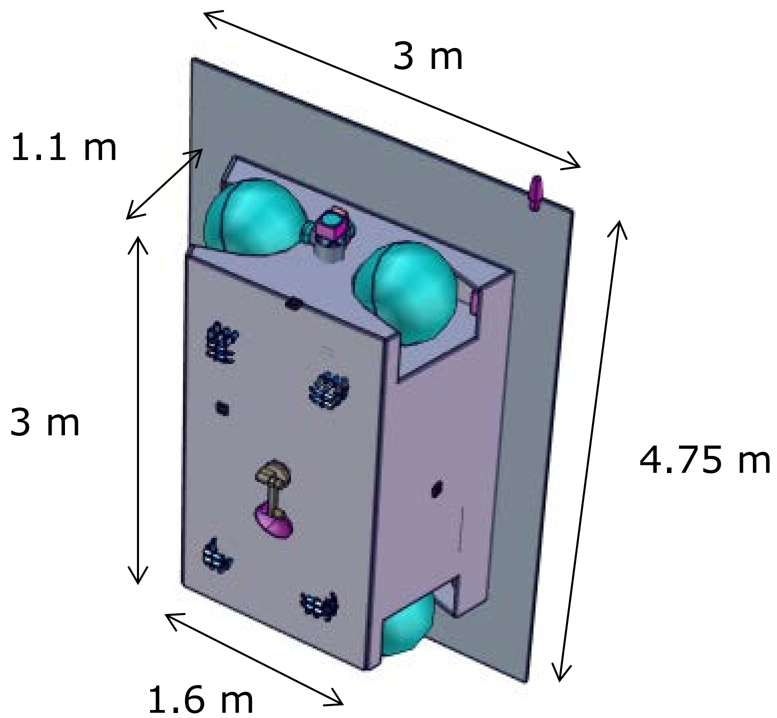
S/C Mass Budget		Mass [kg]
Dry Mass PLM		**473.95**
Dry Mass SVM		**680.18**
System Margin	20%	230.83
Dry Mass incl. System Margin		**1384.96**
EPROP Transfer Propellant Mass		114.50
EPROP Fuel Margin	2%	2.29
CPROP Cold gas fuel Mass		19.70
CPROP Fuel Margin	2%	0.39
Total Propellant		**136.88**
Total Wet Mass		**1521.84**

Stack Mass Budget	Mass [kg]
ScienceCraft 1 Dry	1884.96
Propulsion Sciencecraft 1	136.88
Total Spacecraft mass excl. Adapters	**1521.84**
# Of satellites	3.00
Launch Adapter	1000
Total Stack Mass incl. Adapters	**5565.52**
Target Wet Mass incl. Adapter	7000.00
Below Target Mass by	1434.48

CP option



EP and EP+ option



OPTIONS COMPARISON

		CP	EP	EP+
Total Mass Budget		**Mass [kg]**	**Mass [kg]**	**Mass [kg]**
Attitude, Orbit, Guidance, Navigation Control		38.2	24.5	24.5
Communications		25.7	23.6	23.6
Chemical Propulsion		314.8	190.2	4.4
Data-Handling		16.4	16.4	16.4
Electric Propulsion		0.0	80.7	170.7
Mechanisms		57.6	45.1	45.1
Power		108.5	123.0	132.8
Structures		474.2	208.4	208.4
Thermal Control		21.8	21.8	21.8
Harness	5%	54.0	36.7	32.4
Dry Mass w/o System Margin		**1133.1**	**770.5**	**680.2**
Instruments		473.9	473.9	473.9
Total Mass Budget w/o System Margin		**1607.1**	**1244.4**	**1154.1**
Total Mass Budget with System Margin		**1928.5**	**1493.3**	**1385.0**
Transfer propellant		1104.3	147.9	116.8
AOCS/DFACS propellant		199.8	239.6	20.1
TOTAL		**3232.6**	**1880.9**	**1521.8**

- CP option not feasible with A6, EP offers 1300kg allocation for payload dispenser plus launch margin while EP+ offers 2400kg

PROPULSION COMPARISON

	CP	EP	EP+
Chemical Propulsion	314.8	190.2	4.4
Electric Propulsion	0.0	80.7	170.6
Propulsion dry	**315**	**271**	**175**
Cprop mass	1115	0	0
Eprop mass	0	148	117
Microprop mass	200	240	20
Total	**3244**	**1881**	**1522**

- Chemical propulsion is by far the most inefficient for the transfer (even if power penalty is taken into account), not recommended
- EP propulsion preferred for transfer (EMC pending)
- In principle all micro propulsion technologies fulfil the requirements though EP micro propulsion has a lower technology readiness
- EP micro propulsion has a dry mass that is half of the cold gas system
- EP micro propulsion reduces the required propellant mass for science by a factor 10

IMPACT OF EXTENDED OPERATIONS

- Equipment qualified for nominal lifetime 6.25yr
- System sized for extended operations (orbit, propellant, power, comms, memory)

- Orbit design, constrain to 65mKM after 10 years, slight improvement could be achieved if designed for 4 years – shorter distance to Earth- low impact
- Propellant mass for science (and volume), scales linearly with time – large impact for cold gas option, reaching limit but feasible
- Transfer propellant, decrease for smaller mass (science propellant, structures), though not enough to rescue CP option
- Power, low impact, EP options sized for transfer, CP for science, but size driven by sunshield, growth capability for solar panel
- Assessment of survivability of mission elements pending

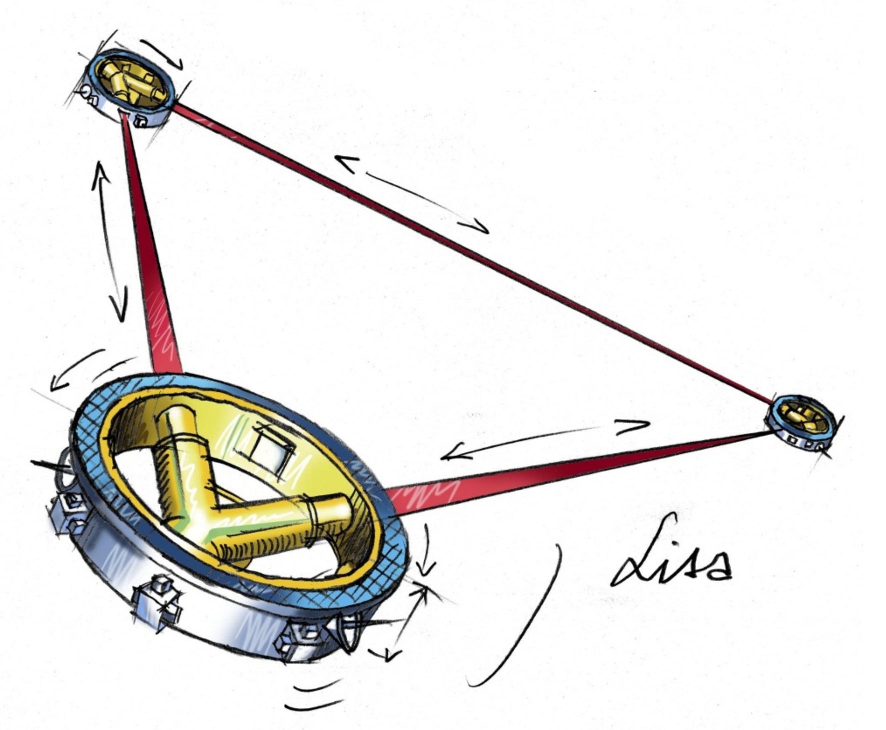
LISA

Payload

Internal Final Presentation
ESTEC, 05-05-2017

Prepared by the CDF* Team

(*) ESTEC Concurrent Design Facility



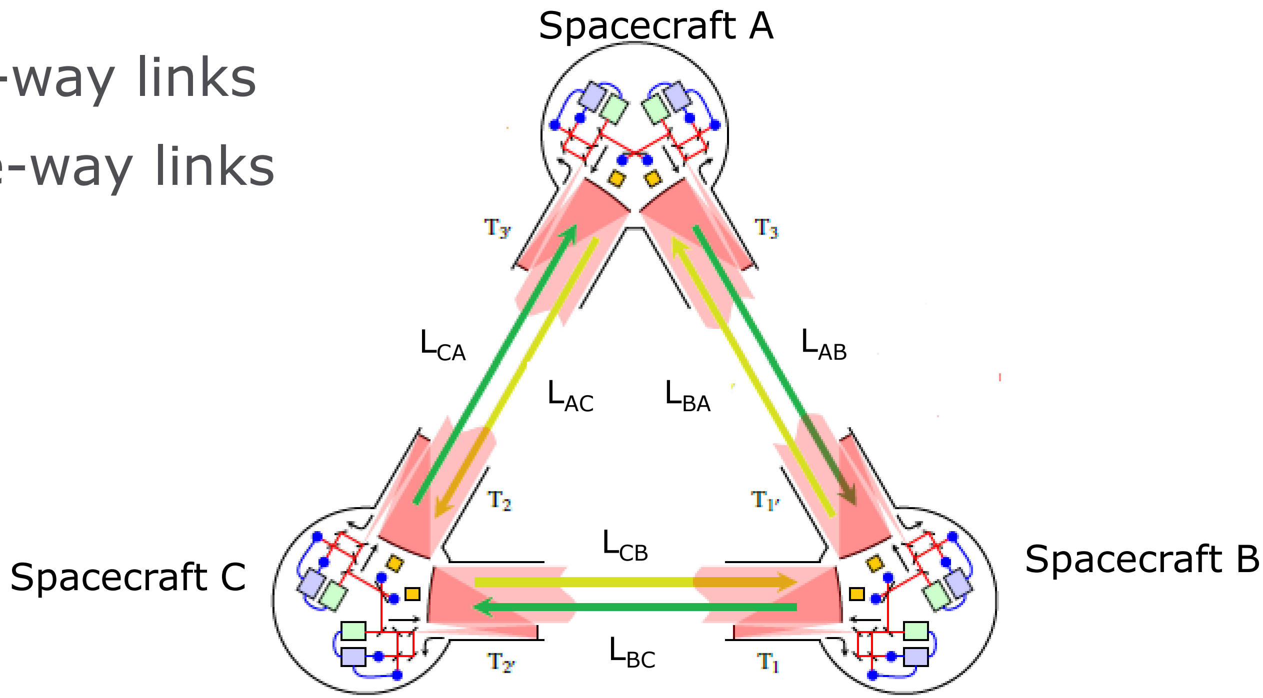
Outline

- Architecture
- Redundancy
- Interfaces
- Trade-offs
- PL budgets

Payload Architecture

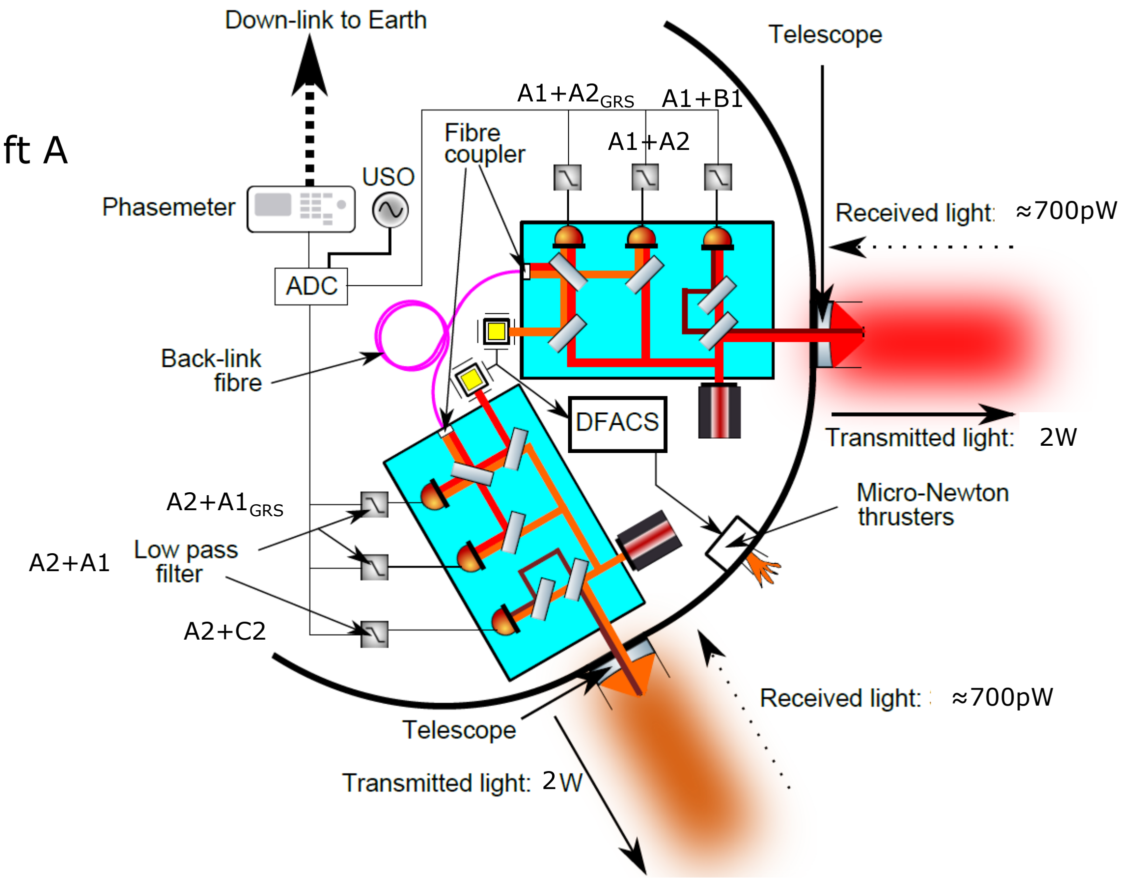
Mission configuration

- 3 identical satellites
- 3 arms
- System: 6 one-way links
- Satellite: 4 one-way links

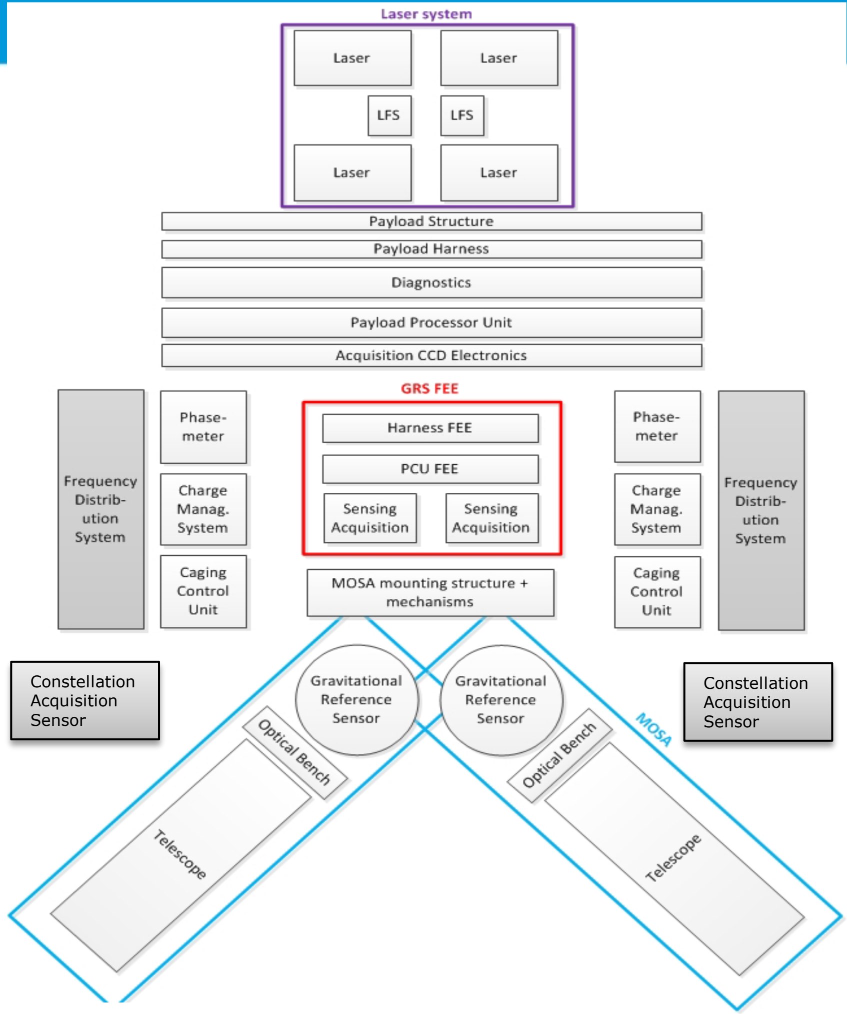


Measurement scheme

Spacecraft A



Payload sub-systems: breakdown



Redundancy

Payload sub-systems: Redundancy

- Proposed redundancy baseline:

  1. If loss of equipment = complete loss of mission
     => **full** redundancy proposed

  2. If loss of equipment = loss of polarisation & performance degradation
     (loss of 1 arm, or no loss)
     => **partial** redundancy or no redundancy

  3. Other cases TBD

Payload sub-systems: Redundancy

- Current Redundancy assumptions:
  - Laser: full redundancy, 2 for each link, i.e. 4 on each S/C
  - Optical bench (active elements):
    - Full redundancy for science interferometers,
    - Strategy TBD for local interferometers
  - Frequency Reference: 1 per link, redundancy insured at S/C level
  - GRS, Charge Management System, Caging control units : 1 per arm, no redundancy
  - GRS FEE: redundant, modification to LPF design to ensure GRS independent switching
  - Phasemeter: strategy TBD.
  - Constellation acquisition sensor: strategy TBD (redundancy implies optical flux loss).
  - Electronics : full redundancy with cross-strapping
- MTTF and risk register analysis will be performed during Phase 0

INTERFACES (preliminary definition)

Payload sub-systems Interfaces: way forward

- Interface definition on-going
- N2-Chart will be generated
    - Example:

Telescope	Far laser signal	
Laser signal to far satellite	Optical Bench	
		Gravitational Reference Sensor

TRADE OFFS

Trade-offs timeline and status

- Science trade-offs (e.g. arm length, telescope size, constellation acquisition, arm locking…) postponed to Phase A.
- Architecture trade-offs (e.g. model philosophy, pre-integration of Payload, testing, redundancy…) postponed to Phase 0.
- Telescope trade-offs (see dedicated slide, currently 4 designs in play)
- Pointing strategy trade-offs (see dedicated slide, IFP vs. OATM)
- Laser trade-offs (e.g. seed laser architecture, modulation for telecoms architecture, separate electronics…) postponed to Phase 0/A.

Trade-offs timeline and status

- Laser Frequency Stabilisation (e.g. position within payload, type of frequency reference confirmation, specification confirmation) postponed to Phase 0
- OB (see dedicated slide, currently 1 bench per arm, 2 OB per S/C)
- GRS (see dedicated slide) modifications on venting and UV source
- CAS (e.g. operating temperature, cooling strategy, matrix type…) postponed to Phase 0
- Phasemeter (e.g. # of channels, bandwidth, integrated frequency distribution system…) postponed to Phase 0.

Trade off 1: telescope

- Two industrial architectures used as baseline.
- Telescope includes removable cover (sun illumination avoidance). Mechanism cf. session 12
- Results to be consolidated
- Alternatives taken into account:
  - ESA design (two off axis 3 mirror stages, all conic, cf. Session 12)
  - NASA design (cf. Session 7: J. Livas)
- Telescope size 300mm (TBC)
- Pointing architecture trade-off
  - *IFP in-field pointing*
    - *Purely re[illegible]ti[illegible]e*
    - *Dio[illegible]s ?*
  - *OAT[illegible]ll [illegible]OSA mo[illegible]n*
- Thermal constraints and interfaces
- Defocusing effects and corrections?
- Telescope materials
- Interface to optical bench, Constellation Acquisition Sensor, …

POSTPONED Phase A

Trade-off 1: Telescope pointing

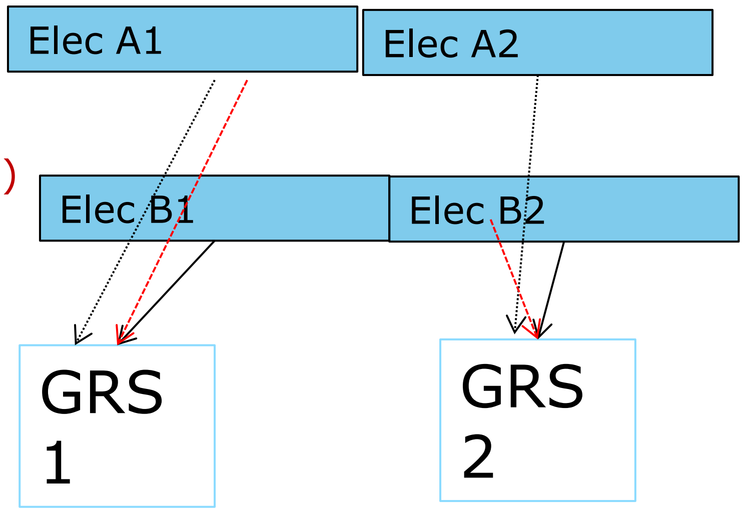
In-field pointing	MOSA pointing
Small mass moved Testing on telescope level Acquisition …	Small range of movement Simpler telescope Path-length stability easier Testing/Integration …
Complexity Heavy Telescope More mechanisms Mechanisms in optical path Thermal impact of mechanics Testing/Integration Alignment …	Large mass moved Mechanism (pivot/actuator) Harness? Gravitational balancing MLI shedding …

Trade off 2: OB

- 1 OB per arm is baseline
  - *Mass, Volume, Cost, architecture, testing…*
- Shape **-> Phase 0**
- Mounting
  - *GRS + telescope + OB*
  - *Mechanical, thermal and optical stability*
- Interfaces (free space vs. fibre connections, electronic vs. optical signals…)
  - *With lasers, other OB (e.g. fibre vs. free space back-link), diagnostic…*
- # of photodetectors (per bench) 60 to 70 per S/C TBC **-> Phase 0**
- Heat dissipation from detectors 7.5W per S/C + 25% margin
- Acquisition sensor (APD matrix, QPD,…) passively cooled, thermally isolated from bench
- Photodetectors thermally isolated from bench
- *Thermal stability requirement: 10µK/√Hz @0.1mHz TBC*
- Optical path length compensation (TBC)

Trade off 3: GRS

- Two per S/C
- 2 redundant Front End Electronics sets in 1 housing
  - *Commutation from A set to B set for both GRS on LPF, never implemented*
  - *Possibility of partial commutation to be evaluated* **-> Phase 0**
- Thermal stability requirement 0.1mK/√Hz @0.1mHz (TBC)
- Transfer conditions
  - *Caging mechanism doesn't require power*
  - *Venting separation from GPRM to be implemented*
- Stand-by power consumption (heating only, if venting)
- GRS read-out: x (longitudinal) optical, rest capacitive
- Potential Modifications:
  - *UV discharge (UV source)*
  - *Venting ITF*



Payload budgets

Dimensions

Dimensions [mm]								
Item	*Amount*	*Volume margin*	*Volume*	*Length mm*	*Width mm*	*Depth mm*	*Diameter mm*	*Reference*
Telescope	2	10	138235692	764			480	LISA ASD DD 3001
Optical Bench	2	50	37699112	229			458	
Gravitational Reference Sensor	2	10	11453855	442			182	LISA PF
GRS Front-End Electronics (SAU)	2	10	**35507560**	458	331	197		RUAG
GRS Front-End Electronics (PCU)	1	10	**5359200**	178	204	143		RUAG
Phasemeter	2	50	**8437500**	278	167	167		
Laser (4, 2 =OFF)	4	20	**7500000**	245	245	98		
Laser Control Unit (4, 2=OFF)	2	20	**3600000**	204	153	102		
Laser Frequency Stabilisation	2	10	**10725000**	243	292	126		
UV Light Unit	2	20	**3600000**	204	153	102		
Caging Control Unit	2	20	**12960000**	166	280	259		
Payload Processing Unit & Diagn	1	20	**18000000**	314	262	210		
Acquisition CCD Electronics	1	20	**2246400**	158	206	29		
Payload Harness	1							
Payload Structure	1							
Multi-layer Insulation	1							
GRAND TOTAL			295324319					

Mass

Mass budget (laser 2, telescope 1)					
Item	*Amount*	*kg/unit*	*margin*	*total*	*reference*
Telescope	2	26.98	20	64.74	Guesstimate based on discussion with Isabel
Big mirror	1	3.76	0	3.76	
Medium mirror	2	0.42	0	0.83	
Small mirror	4	0.03	0	0.14	
Mechanism in-field pointing	1	4.60	0	4.60	
Mechanism telescope motion	1	5.40	0	5.40	Mission formulation
Structure (rods)	2	0.13	0	0.26	
Cover + Mechanism	1	2.00	0	2.00	
Baffle	1	9.98	0	9.98	
Optical Bench	2	15.00	20	36	NGO design
Gravitational Reference Sensor	2	17.80	5	37.38	LISA PF flight value
GRS Front-End Electronics	2	32.75	20	78.6	LISA GRS FEE RUAG
FEE SAU	2	14.35	0	28.7	LISA GRS FEE RUAG
FEE PCU	1	1.43	0	1.43	LISA GRS FEE RUAG
FEE Harness	1	2.62	0	2.62	LISA GRS FEE RUAG
Phasemeter	2	10.00	20	24	LISA MF + 25% because more channels
Frequency Distribution System	2	?	?		
Laser	4	16.79	20	80.592	
Seed Laser	1	1.50	0	1.5	TESAT
Amplifier	1	8.90	0	8.9	LUSO
Modulator	1	1.19	0	1.19	LISA PF flight value
Laser Harness	1	0.20	0	0.2	
Laser Control Unit	1	5.00	0	5	Guesstimate Oliver
Laser Frequency Stabilisation	2	12.00	20	28.8	Ball Aerospace Cavity (NPL)
Optics	1	10.00	0	10	
Electronics	1	2.00	0	2	
Charge Management System	2	4.50	10	9.9	Airbus system doc LISA PF, pg 80
Caging Control Unit	2	3.36	10	7.392	LISA PF flight value
Diagnostics	1	3.00	10	3.30	Estimate based on Airbus system doc LISA PF, p
Magnetometers	11	0.16	0	1.7325	
Radiation Monitors	1	1.27	0	1.27	
Heaters		0.00	0	0	
(Payload PU included) Electronic	1	0.00	0	0	in PPU
Payload Processing Unit	1	8.00	20	9.6	Proposal
Acquisition CCD Electronics	1	10.00	20	12	Guesstimate w Oliver
Payload Harness	1	37.20		37.2	Proposal
Payload Structure	1	44.44		44.44	Proposal
optical assembly mechanics	1	22.44	0	22.44	
GRS+OB+TEL structure	2	8.20	0	16.4	
Thermal Control	2	0.88	0	1.76	
Optical Assembly Thermal Contr	1	3.84	0	3.84	
GRAND TOTAL				473.95	

Power

			Mode						
Item	*Amount*	*Margin*	*Science*	*Acquisition*	*Accelerometer*	*Fast Discharge*	*Duty cycle*	*Peak power [W]*	*Dissipated power [%]*
Telescope	**2**	**20**	**0.52**	**5.52**	**0.52**	**0.52**	**0.09**		**99.00**
Big mirror	3		0.00	0.00	0.00	0.00			
Small mirror	4		0.00	0.00	0.00	0.00			
Mechanism in-field pointing	1		0.00	2.50	0.00	0.00			99.00
Mechanism telescope motion	1		0.00	2.50	0.00	0.00			99.00
Structure (rods)	2		0.00	0.00	0.00	0.00			
Cover + Mechanism	2		0.00	0.00	0.00	0.00		10.00	99.00
Baffle	1		0.00	0.00	0.00	0.00			0.00
Optical Truss SED	4		0.13	0.13	0.13	0.13			99.00
Optical Bench	**2**	**20**	**3.75**	**5.45**	**3.75**	**3.75**	**0.69**		**99**
Photodiodes + pre-amplifiers	30		0.13	0.13	0.13	0.13			
Fibre Injector Switch	2		0.00	0.00	0.00	0.00			
Re-focussing mechanism	1		0.00	0.00	0.00	0.00			
Acquisition sensor	1		0.00	1.70	0.00	0.00			
Baseplate/Mirrors	1		0.00	0.00	0.00	0.00			
Point Ahead Angle Mechanism	1		0.00	0.00	0.00	0.00			
Gravitational Reference Sensor	**2**	**5**	**0.00**	**0.00**	**0.00**	**0.00**			**0**
Caging Release			0.00	0.00	0.00	0.00		2.50	
GRS Front-End Electronics	**2**	**10**	**56.00**	**84.00**	**84.00**	**56.00**	**0.67**		**99**
Phasemeter	**2**	**20**	**50.00**	**50.00**	**50.00**	**50.00**			**99**
Laser (4, 2=OFF)	**2**	**20**	**100.00**	**100.00**	**100.00**	**100.00**			**98**
Laser Frequency Stabilisation (2, 1=OFF)	**1**	**50**	**6.00**	**6.00**	**6.00**	**6.00**			**99**
Charge Management System	**2**	**10**	**5.00**	**0.00**	**0.00**	**10.00**			**99**
Caging Control Unit	**2**	**10**	**0.00**	**0.00**	**0.00**	**0.00**		**18.00**	**99**
Diagnostics	**1**	**10**	**15.80**	**0.00**	**0.00**	**0.00**			**99.00**
Magnetometers	11		0.80	0.00	0.00	0.00			
Radiation Monitors	1		6.00	0.00	0.00	0.00			
Heaters	1		1.00	0.00	0.00	0.00		5.00	
(Payload PU included) Electronics	1		0.00	0.00	0.00	0.00		0.00	
Payload Processing Unit	**1**	**20**	**30.00**	**30.00**	**30.00**	**30.00**			**99**
Acquisition CCD Electronics	**1**	**20**	**0**	**6.70**	**0.00**	**0.00**			**99**
GRAND TOTAL			**566.83**	**624.17**	**600.05**	**560.45**			

Data rates for downlink sizing

Source		Measurement	Channel Count	Sample Rate [Hz]	Bits per Channel	Rate [bits/s]
		Payload				
IFO Longitudinal		Science IFO	2	3.3	32	211.2
		Test Mass IFO	2	3.3	32	211.2
		Test mass y IFO	2	3.3	32	211.2
		Reference IFO	2	3.3	32	211.2
		Clock Sidebands	4	3.3	32	422.4
Freq reference		error point	1	3.3	32	105.6
		feedback	2	3.3	32	211.2
		clock sidebands monitoring (local pilot tone beat)	1	3.3	32	105.6
IFO Angular		SC η,φ	4	3.3	32	422.4
		TM η,φ	4	3.3	32	422.4
		TM θ (from y IFO)	2	3.3	32	211.2
Ancillary		Time Semaphores	2	3.3	32	211.2
Optical Monitoring		PAAM Longitudinal	2	3.3	32	211.2
		PAAM Angular	4	3.3	32	422.4
		Optical Truss	6	3.3	32	633.6
GRS Cap. Sens.		TM x,y,z	6	3.3	24	475.2
		TM θ,η,φ	6	3.3	24	475.2
DFACS		breathing errorpoint	1	3.3	32	105.6
		breathing actuator	1	3.3	32	105.6
		TM applied torques	12	3.3	24	950.4
		TM applied forces	12	3.3	24	950.4
		SC applied torques	3	3.3	24	237.6
		SC applied forces	3	3.3	24	237.6
Science Diagnostics	Themometers	EH	16	0.1	32	51
		OB	20	0.1	32	64
		Telescope	10	0.1	32	32
		interface	10	0.1	32	32
	Magnetometers	TM	12	0.1	32	38
	radiation monitor		1			30
	FIOS output powers (Inloop and Out of Loop)		4	3.3	32	422
	pressure sensor		4	0.1	32	13
	body mic	CGAS tanks	6	3.3	32	634
		breathing mechanism	4	3.3	32	422
	RIN monitoring	2 lasers, 2 frequencies, 2 quadratures	8	3.3	32	845
				0.0		0
				0.0		0
Payload HK						1000
Total Payload						***11345***
		Platform				
Housekeeping [Based on LPF]						4000
Total Platform						***4000***
		Totals				
Raw Rate per SC						15345
Packetisation Overhead [10%]						1535
Packaged Rate per SC						**16880**
Packaged Rate for Constellation						**50639**

Total Payload		***11345***
Platform		
Housekeeping [Based on LPF]		4000
Total Platform		***4000***
Totals		
Raw Rate per SC		15345
Packetisation Overhead [10%]		1535
Packaged Rate per SC		**16880**
Packaged Rate for Constellation		**50639**

8 hours of GS contact/day

152 kbit/s

Data Rates for Mass Memory sizing

Source		Measurement	Channel Count	Sample Rate [Hz]	Bits per Channel	Rate [bits/s]
Payload						
IFO Longitudinal		Science IFO	2	20.0	32	1280.0
		Test Mass IFO	2	20.0	32	1280.0
		Test mass y IFO	2	20.0	32	1280.0
		Reference IFO	2	20.0	32	1280.0
		Clock Sidebands	4	20.0	32	2560.0
		raw phases (sci, balanced, hot redundant)	112	20.0	32	71680.0
		raw phases TMx (sci, balanced, hot redundant)	48	20.0	32	30720.0
		raw phases TMy (sci, balanced, hot redundant)	48	20.0	32	30720.0
		Reference IFO (balanced)	6	20.0	32	3840.0
				20.0	32	0.0
				20.0	32	0.0
Freq reference		error point	1	20.0	32	640.0
		feedback	2	20.0	32	1280.0
		clock sidebands monitoring (local pilot tone beat)	1	20.0	32	640.0
IFO Angular		SC η,φ	4	20.0	32	2560.0
		TM η,φ	4	20.0	32	2560.0
		TM θ (from y IFO)	2	20.0	32	1280.0
Ancillary		Time Semaphores	2	20.0	32	1280.0
Optical Monitoring		PAAM Longitudinal	2	20.0	32	1280.0
		PAAM Angular	4	20.0	32	2560.0
		Optical Truss	6	20.0	32	3840.0
GRS Cap. Sens.		TM x,y,z	6	20.0	24	2880.0
		TM θ,η,φ	6	20.0	24	2880.0
DFACS		breathing errorpoint	1	20.0	32	640.0
		breathing actuator	1	20.0	32	640.0
		TM applied torques	12	20.0	24	5760.0
		TM applied forces	12	20.0	24	5760.0
		SC applied torques	3	20.0	24	1440.0
		SC applied forces	3	20.0	24	1440.0
		control error	15	20.0	24	7200.0
		control guidance	15	20.0	24	7200.0
		sensor inputs	22	20.0	24	10560.0
		Commanded voltages (ac, dc)	48	20.0	24	23040.0
		thruster commands	12	20.0	24	5760.0
Science Diagnostics	Themometers	EH	0	0.1	32	0
		OB	0	0.1	32	0
		Telescope	0	0.1	32	0
		interface	0	0.1	32	0
	Magnetometers	TM	0	0.1	32	0
	radiation monitor		0			30
	TM beam power (OOL)		0	20.0	32	0
	pressure sensor		0	0.1	32	0
	body mic	CGAS tanks	6	20.0	32	3840
		breathing mechanism	4	20.0	32	2560
				0.0		0
				0.0		0
				0.0		0
Payload HK						0
Total Payload						***244190***
Platform						
Housekeeping [Based on LPF]						0
Total Platform						***0***
Totals						
Raw Rate per SC						244190
Packetisation Overhead [10%]						24419
Packaged Rate per SC						**268609**
Packaged Rate for Constellation						**805827**

Total Payload	***244190***
Platform	
Housekeeping [Based on LPF]	0
Total Platform	***0***
Totals	
Raw Rate per SC	244190
Packetisation Overhead [10%]	24419
Packaged Rate per SC	**268609**
Packaged Rate for Constellation	**805827**

Thermal Requirements

	Min OPT	Max OPT	Min NOPT	Max NOPT	Thermal Stability [K / root Hz @ 0.1 mHz]
Telescope	-100	30	-100	+50	TBD
Optical Bench	10	30	0	40	10^-5
Gravitational Reference Sensor	0	30	-10	40	10^-4
GRS Front-End Electronics	0	40	-20	50	TBD
Phasemeter	10	30	0	40	10^-3
Frequency Distribution System	10	30	0	40	TBD
Laser (4, 2=OFF)	23	29	-10	30	10^-3
Laser Frequency Stabilisation (2, 1=OFF)	10	30	0	40	10^-4
Charge Management System	10	30	0 -> -10	40->50	-
Caging Control Unit	10	30	-10	50	-
Diagnostics	10	30	0 ->-10	40->50	-
Payload Processing Unit	10	30	0 ->-10	40->50	-
Acquisition CCD Electronics	10	30	0 ->-10	40->50	-

Summary

- Payload design and breakdown TBC
- Critical payload items identified
- Necessary trade-offs identified
- Payload budgets consolidated
- Interface identification on-going
  - Interface constraints definition to be performed

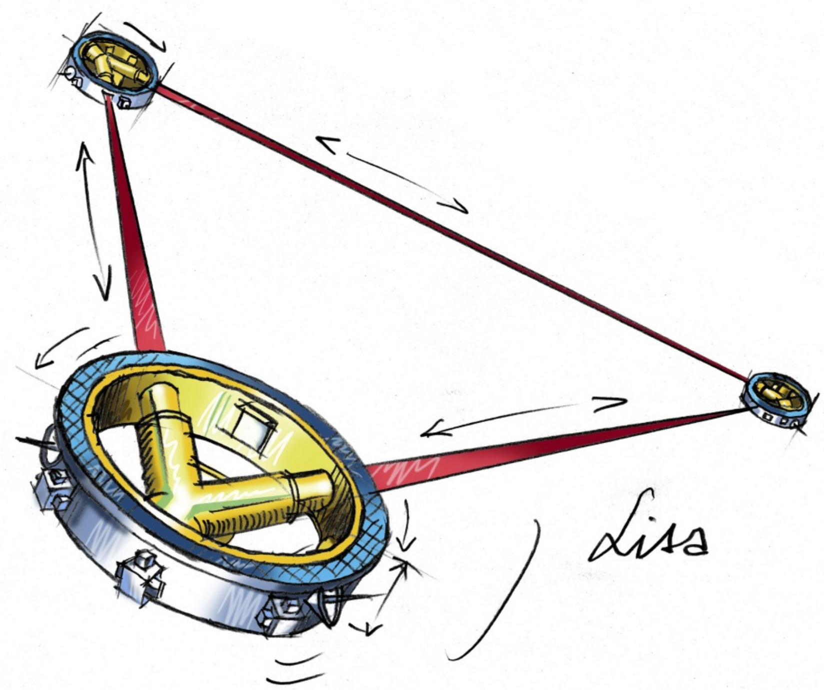
LISA

Mission Analysis

Internal Final Presentation
ESTEC, 05-05-2017

Prepared by the CDF* Team

(*) ESTEC Concurrent Design Facility



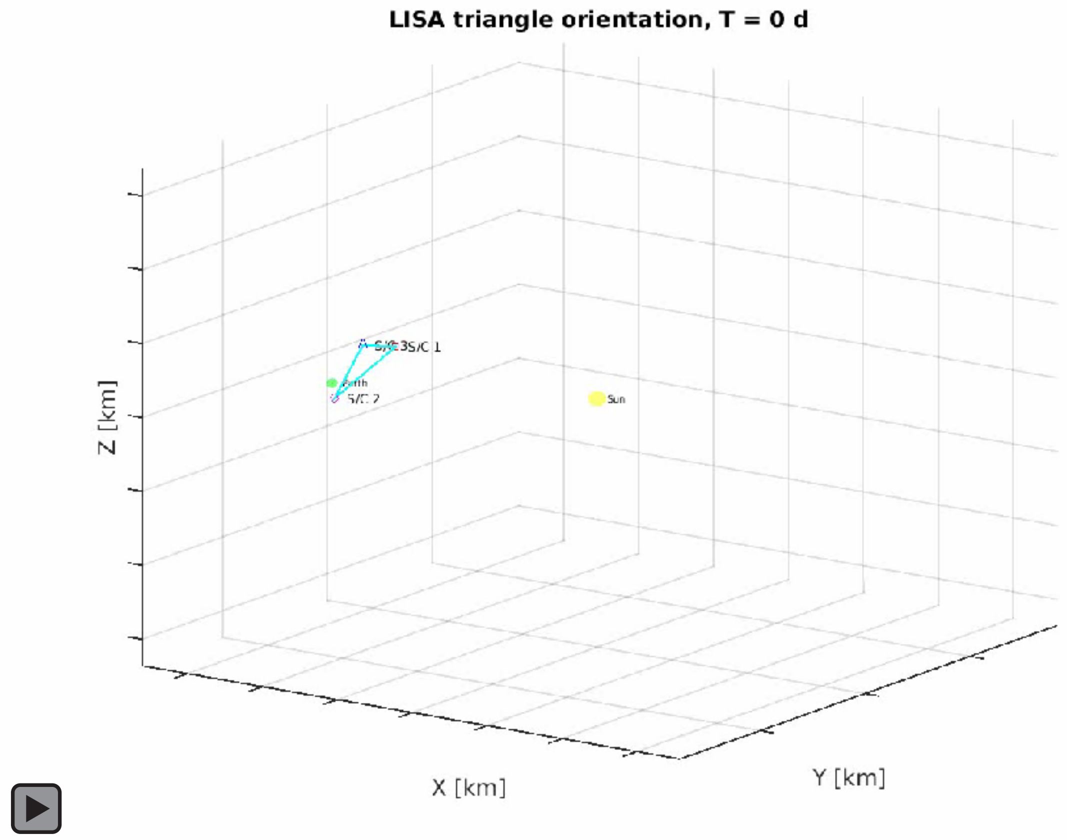
Assumptions and Requirements

- Arm length 2.5 million km
- Corner angle variation less than +/- 1 deg
- Arm length rate less than +/- 20 m/s
- Operational phase duration 10 years
- Maximum range 65 million km (taken as maximum distance from Earth centre to centre of triangle)
  - Trade-off of initial displacement angle to satisfy duration+distance+ requirements
- Joint launch of all three spacecraft with Ariane 64 in 2034
- Trade-off of CP vs. SEP
  - For SEP: Thrust 90mN, Isp 1660 s
  - Transfer duration no longer than ~1.5 years

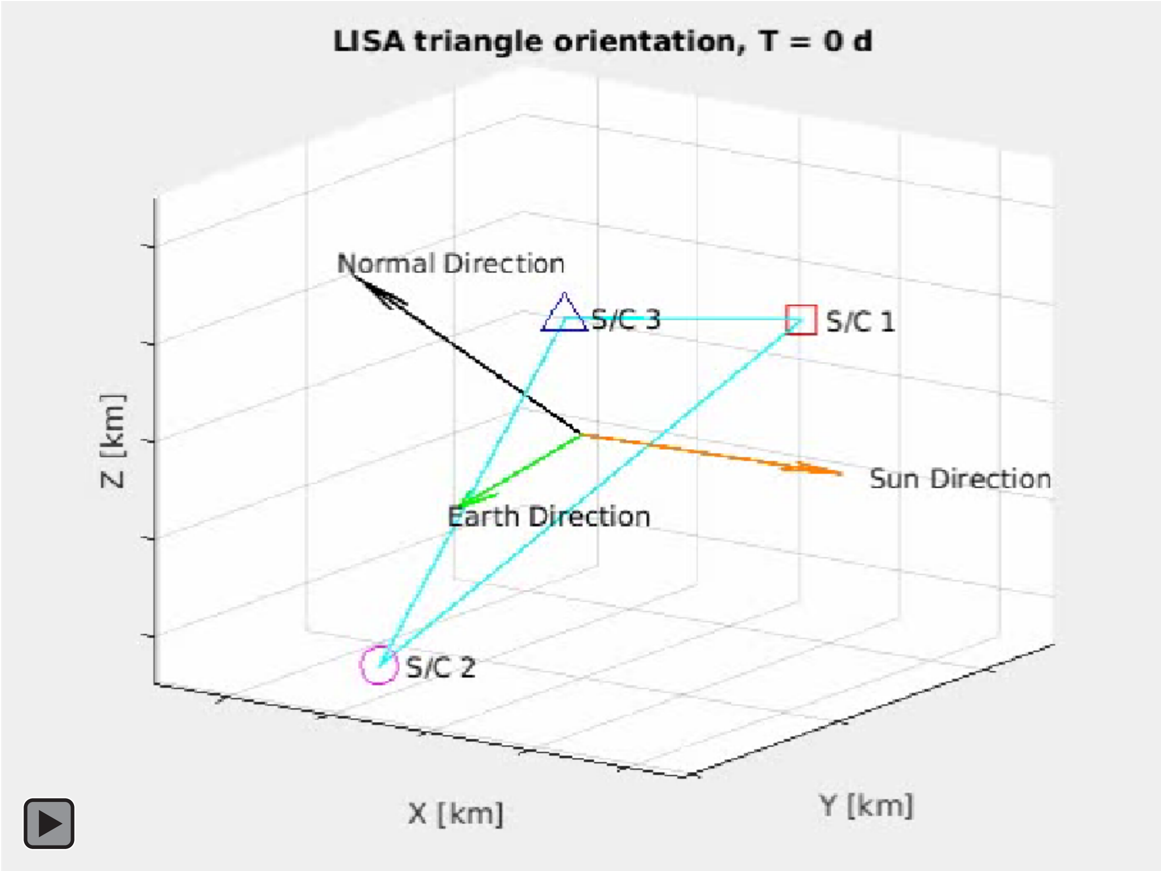
Operational Orbit

- Both heading and trailing orbits are feasible
  - Transfer durations and properties of both types are similar
  - In the study, only the class of trailing orbits was regarded in detail
- First task: Determination of required initial displacement angle

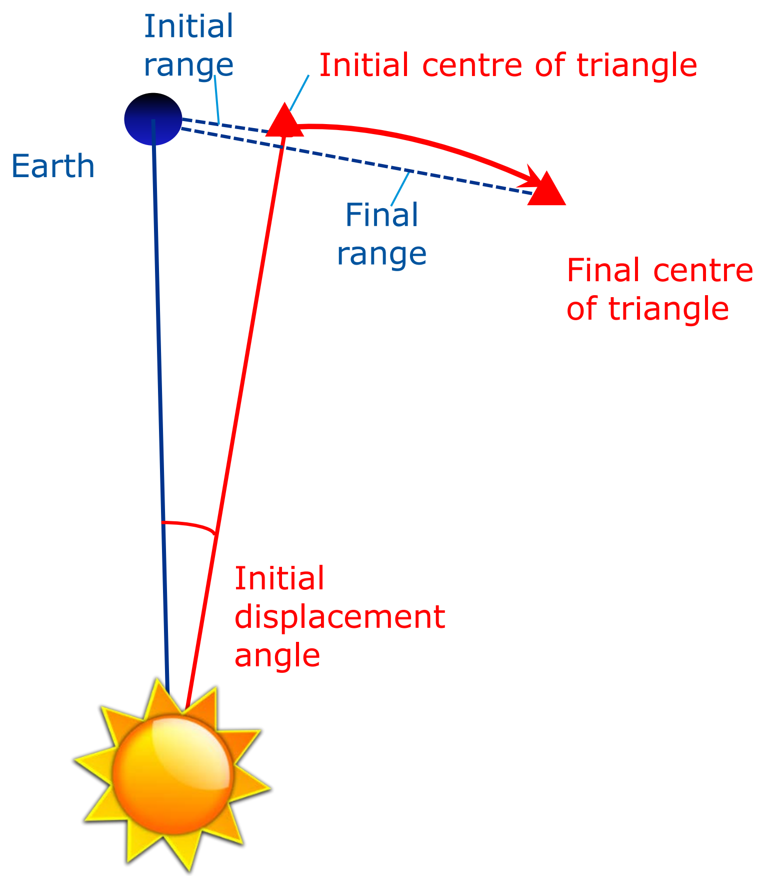
Triangular Configuration in Space



Triangular Configuration Geometry

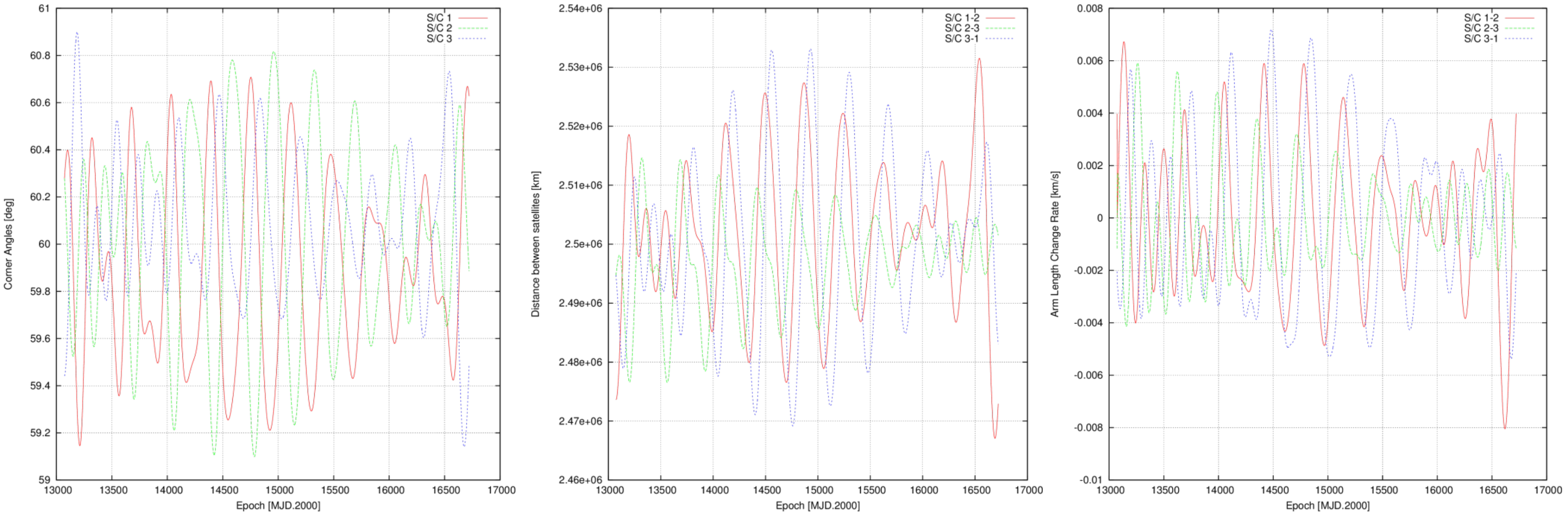


Initial Displacement Angle



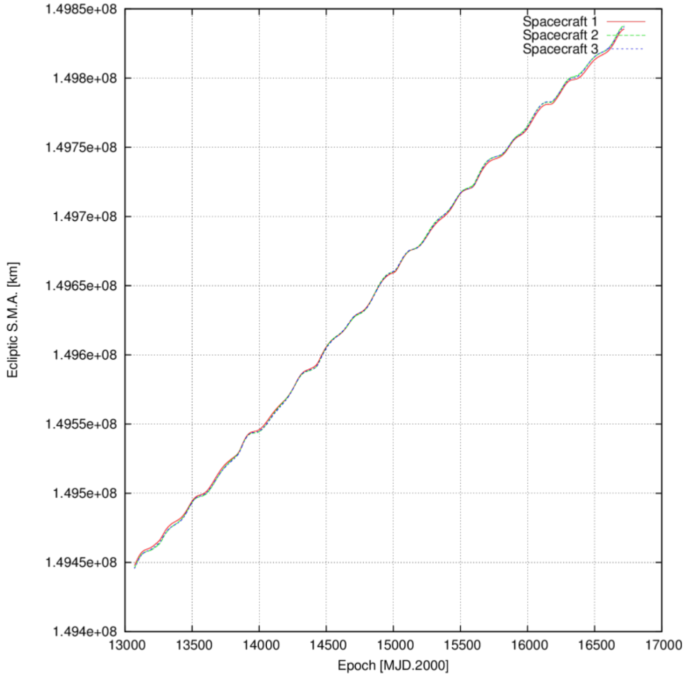
- The trailing orbit is perturbed by the gravitational pull of the Earth
  - The spacecraft orbit gains energy and the spacecraft positions will drift outward and back (in the GSE frame)
  - New parameter: The initial displacement angle (IDA). The smaller the IDA:
    - The faster and cheaper the transfer
    - The faster the drift rate and the shorter the time to a 65 million km range
    - The larger the perturbations of the configuration
  - Numerical analysis led to choice of -20 deg for the given set of requirements
- For heading orbits, the situation is analogous
  - There the IDA value is +20deg

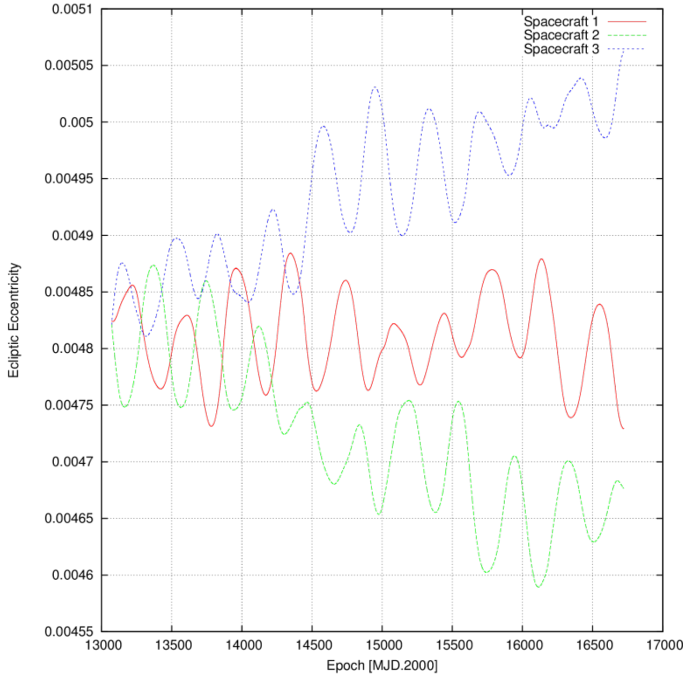
Operational Orbit, IDA -20 deg



- Fully numerical analysis, without stationkeeping
- Corner angle varies in range of 60 +/- 0.9 deg
- Arm length within 35,000 km of nominal value of 2.5 million km
- Arm length rate of change does not exceed 8 m/s, mostly much lower

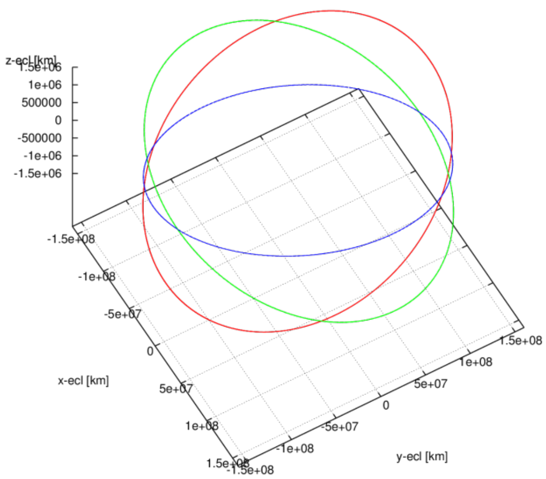
Operational Orbit: Orbital elements

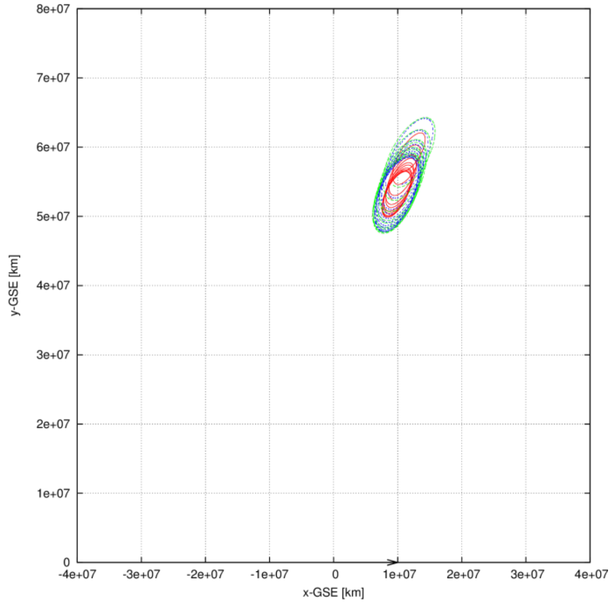


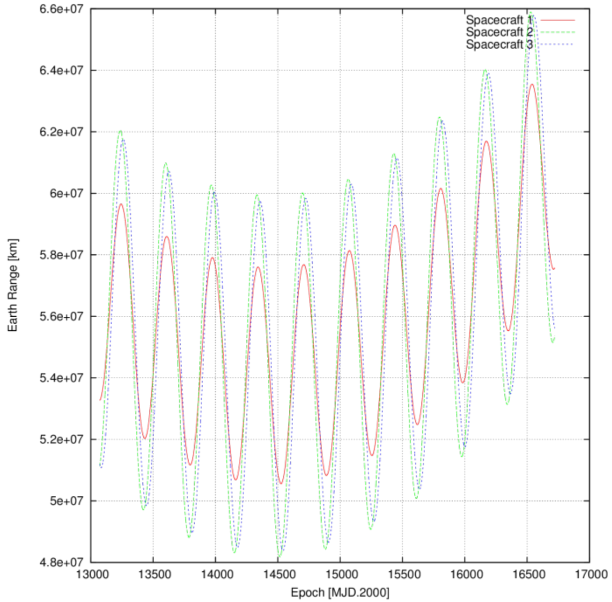


- Semi-major axis increases constantly but not steadily due to perturbation by Earth
- Eccentricities of three orbits diverge (slightly)
- Initial s.m.a. is biased to value < Earth orbit to limit drift rate
  - This is a major driver for the delta-v cost of the transfer

Operational orbit: Evolution







- Fully numerical analysis, without stationkeeping
- Corner angle varies in range of 60 +/- 0.9 deg
- Arm length rate of change does not exceed 8 m/s, mostly much lower
- Effect of s.m.a. biasing clearly visible in Earth range evolution

Launch and Transfer

- All spacecraft are launched together with one Ariane 64
- Launch orbit is similar to GTO:
  - Argument of perigee ~180 deg
  - Inclination ~7
  - Perigee altitude ~250 km
  - One upper stage firing
  - Drop zones and time line similar to standard GTO launch
  - Same launcher program for every date of the year
  - Target C3 slightly above 0 km^2/s^2
- All spacecraft follow essentially the same trajectory initially, apart from small delta-v imparted by deployment mechanism and upper stage manoeuvres
  - Trajectories separate significantly when spacecraft perform first large CP manoeuvre or start their SEP system, many days after launch

CP transfer cost over year (IDA -20 deg)

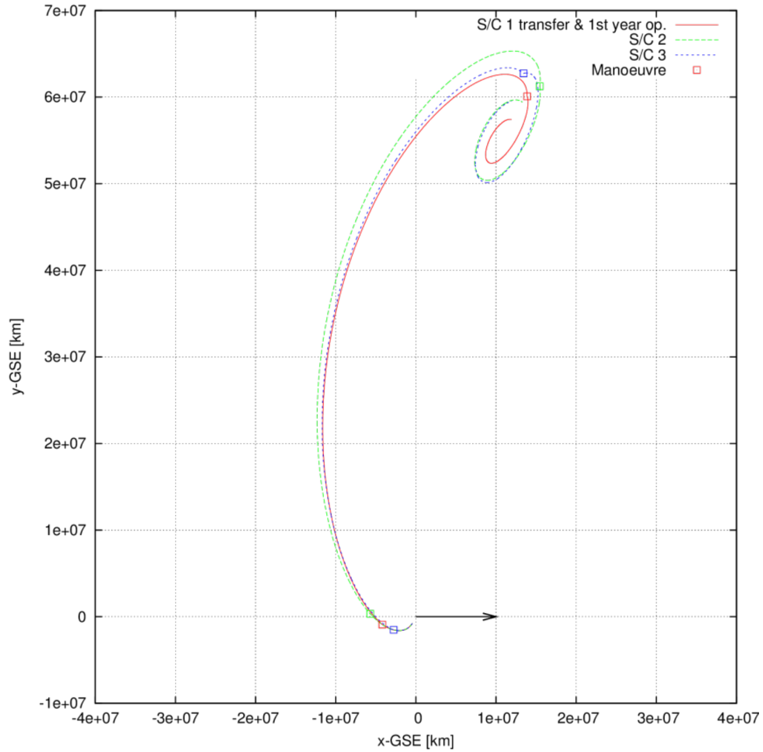
Launch date	2034/1/21	2034/2/21	2034/3/21	2034/4/21	2034/5/22	2034/6/22	2034/7/22	2024/8/22	2034/9/22	2034/10/22	2034/11/21	2034/12/21
W/C delta-v [m/s]	929	921	923	1010	1102	1023	904	839	945	912	903	902
W/C transfer duration [d]	383	362	360	348	344	357	377	344	372	392	392	385

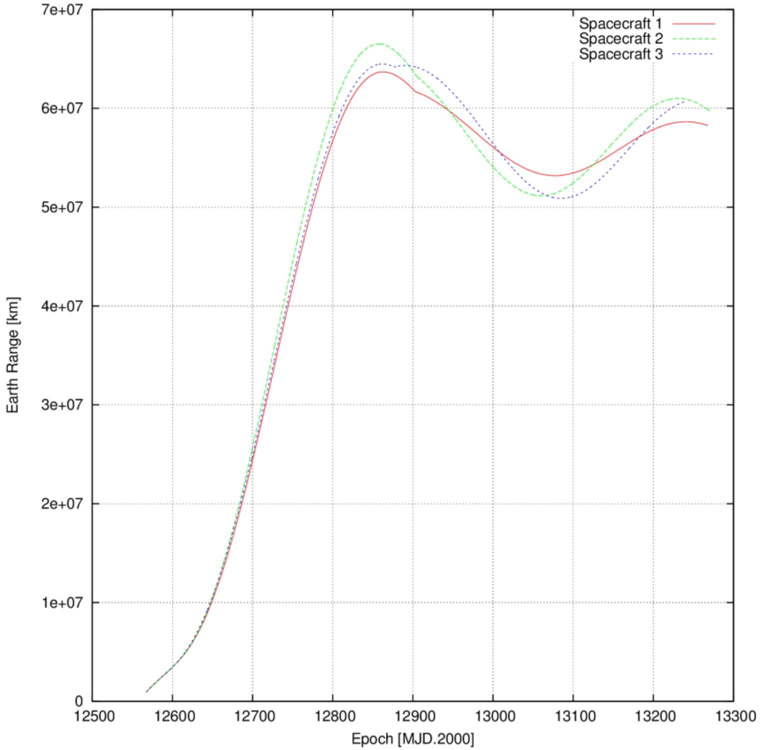
- IDA -20 deg
- 2.5 million km arm length
- 10 year science mission duration
- ~1 year transfer duration

  (no margins added)

Reference case

CP Reference Case: Launch in May





CP Reference Case: Manoeuvres

	Spacecraft Number: 1	Spacecraft Number: 2	Spacecraft Number: 3
Manoeuvre 1 Epoch:	2034/07/12-00:30:15	2034/07/24-17:05:53	2034/06/28-18:35:16
Time from start [d]:	50.52101	63.21242	37.27450
Delta v [km/s]:	0.19909	0.42086	0.22499
Right ascension [deg]:	-173.67268	-150.35512	-147.01842
Declination [deg]:	-17.08385	-58.28138	11.68375
SAA [deg]:	93.68995	103.87730	119.94583
EAA [deg]:	78.65840	95.71637	95.30908
Manoeuvre 2 Epoch:	2035/04/30-04:28:54	2035/05/02-07:54:01	2035/04/03-19:19:07
Time from start [d]:	342.68673	344.82918	316.30494
Delta v [km/s]:	0.78485	0.68134	0.80104
Right ascension [deg]:	110.11332	99.02466	116.66864
Declination [deg]:	-12.47398	3.24965	18.77466
SAA [deg]:	97.07192	82.94629	126.52132
EAA [deg]:	145.97122	152.88295	156.09227
Total delta v [km/s]:	0.98395	1.10220	1.02603

- Impulsive manoeuvres as following from optimization process
  - Operational considerations such as 98% + 2% split or adequate time spacing not yet considered (out of scope for CDF)
  - SAA and EAA angles as well as thrust direction in EME frame are given
  - Trajectory files uploaded to Miscellaneous folder

SEP Transfers

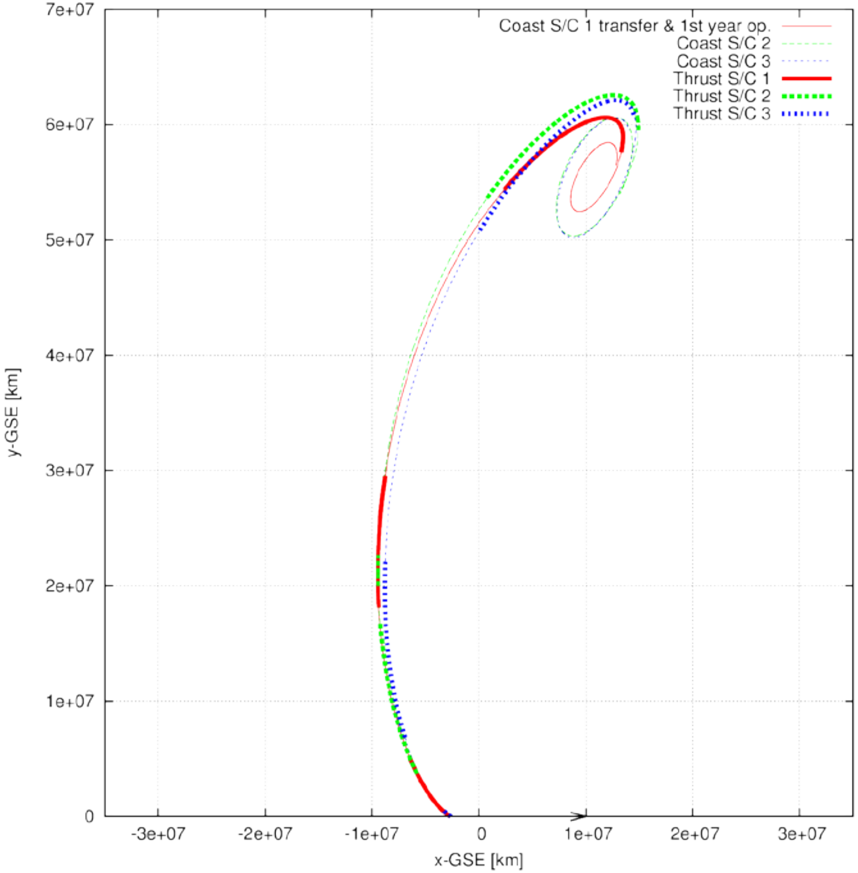
- For SEP transfers, SAA is constrained to 90 +/- 40 deg during thrust arcs, imposing additional constraints on trajectory design
- Three thrust arcs have been assumed. After respective third thrust arc, each spacecraft will have reached its operational orbit
- Thrust/mass ratio can be of concern. If too low, transfer duration of ~1.5 years cannot be achieved

SEP transfers, 1900 kg wet mass

S/C	1	2	3
Delta-v [m/s]	1164	988	1091
Thrust-on-time [d]	275	234	257
Arrival mass*) [kg]	1768	1788	1777
Transfer duration [d]	445	423	428
Min SAA [deg]	52	61	63
Max SAA [deg]	112	123	124

*) No margins, navigation or attitude control taken into account

SEP Transfer, Thrust / Coast Arcs



SEP transfers, 1550 kg wet mass

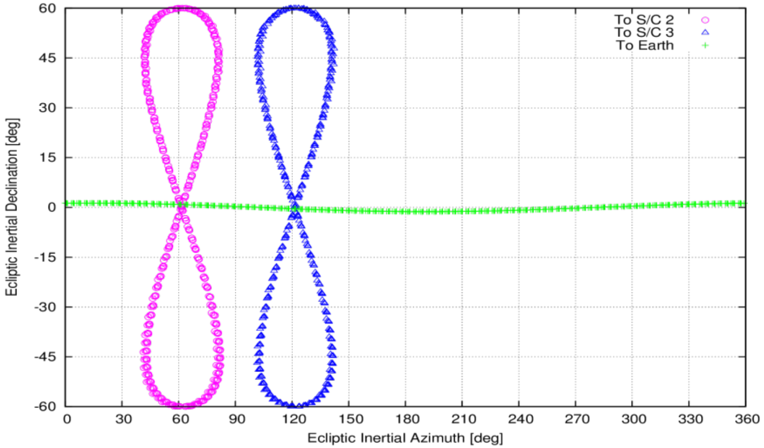
S/C	1	2	3
Delta-v [m/s]	1069	895	1000
Thrust-on-time [d]	206	174	193
Arrival mass*) [kg]	1451	1467	1457
Transfer duration [d]	411	406	412
Min SAA [deg]	76	60	54
Max SAA [deg]	130	130	125

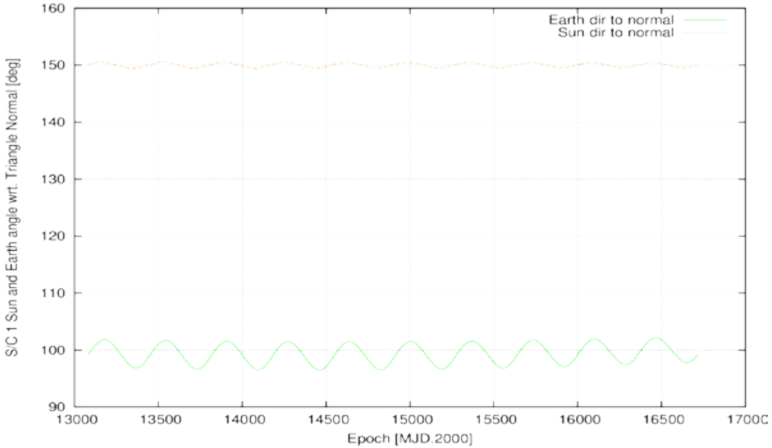
*) No margins, navigation or attitude control taken into account

Stationkeeping Assessment

- The effectiveness of stationkeeping on the operational orbit has been assessed
  - Stationkeeping frequency was restricted to one Hohmann transfer per spacecraft every two years
  - A Hohmann transfer requires two manoeuvres spaced by ~6 months, correcting semi-major axis and eccentricity
  - The manoeuvre date for each spacecraft will be different
  - Impulsive stationkeeping cost observed in numerical simulation: 9 m/s per spacecraft over the entire 10 year science phase
  - This strategy is found to reduce the corner angle variation to within 60 deg +/- 0.6 deg
  - The benefit should be traded against the drawback, which is mainly operational: Numerous interruptions of science operations for a TBD time

Inertial Orientation Results (Example)





- Inertial, ecliptic pointing direction to Earth changes
  - By 360 deg in azimuth
  - By +/-1.5 deg in declination
- Viewing directions to two other spacecraft vary by
  - +/- 60 deg in elevation
  - Only a limited value in azimuth

Other issues

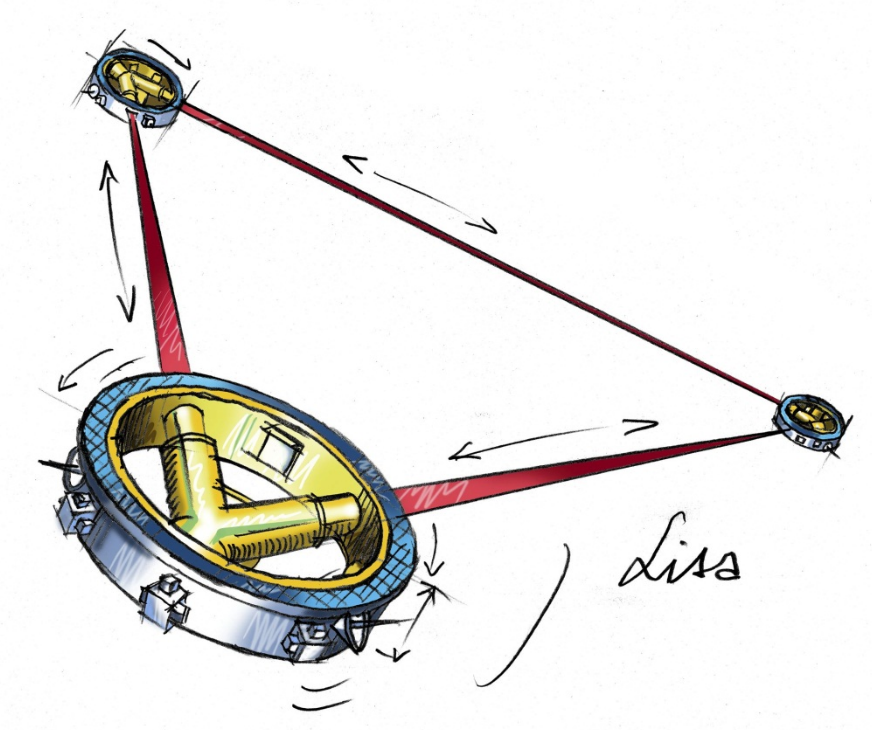
- Ground station comms:
  - All three spacecraft always approx. in same region of the sky, have to share same G/S
- Attitude in operational orbit:
  - Need to maintain mutual visibility completely defines attitude
  - Earth and Sun position define antenna orientability, power input
- Transfer attitude:
  - SEP more constraining than CP: thrust direction is defined during long thrust arcs
  - Power and Earth comms impose added constraints

LISA

Ground Segment and Operations

Internal Final Presentation
ESTEC, 05-05-2017

Prepared by the CDF* Team



(*) ESTEC Concurrent Design Facility

LEOP Mission Phase Assumptions

Launch and Early Operations Phase (LEOP) ≈ 3 days (TBC)

- Quasi continuous coverage
- Suitable combination of 15m stations (TBC) and 35m station.
- Coverage still to be analysed (NNO-2/MAL-X/TBD as Acquisition Aid).
- No time critical manoeuvres are foreseen.

Activities: constellation deployment, subsystem activations (on-board computer, power, thermal, TT&C, AOCS, propulsion system, EP HGA deployment) part of the separation sequence.

Recommendation: Very long LEOP should be avoided as far as possible.

NECP Mission Phase Assumptions

Near Earth Commissioning Phase (NECP) ≈10 wks

- End of LEOP until s/c Transfer Manoeuvre.
- S/C NEC will be performed individually per s/c (TBC):
  - Single station daily coverage for the S/C Commissioning, 6 days per week
  - Reduced coverage (2h/day p. S/C) for the "waiting" S/C, 6 days per week

Activities: initial checkout of all subsystems (except DFACS), tracking and orbit determination, attitude determination on-ground, calibration of the AOCS sensors and actuators, detailed power/electrical/data systems checkout, payload electronics checkout.

Transfer Phase Assumptions

Transfer Phase ≈1 yr

- 2h per week per s/c (simultaneous RRAR and memory dumps)
- Until 1 month before Insertion Manoeuvre.
- Formation Orbit Insertion Manoeuvre preparation close to final orbit arrival: extended coverage.

Activities: platform checkout, tracking and orbit determination, ranging/Doppler. Early venting based on LPF experience (TBC) and GRS warm-up: activities should be compatible with the communications windows.

SCP Mission Phase Assumptions

System Commissioning Phase (SCP) < 9 months

- S/C commissioning: single station daily coverage 10h.
- First Constellation Acquisition: double station coverage 10h during 7 days. OD accuracy higher with DDOR is TBD, a navigation analysis is need.
- DFACS testing: continuous coverage (quasi continuous coverage as possible with the 3 ESA deep space stations is assumed to be compatible TBC).
- ~ 2.5 months (4 weeks tracking and correction manoeuvres, 1 week propulsion module jettisoning, one week constellation acquisition, one week final drag free control testing. Per spacecraft and overlapping).
- ~ 3 months of Instrument Calibrations (TBC).

Activities: as per NECP, constellation acquisition, Drag-Free Control testing, platform and payload calibration, tracking and orbit determination, propulsion module deployment (CP).

NSP Mission Phase Assumptions

Nominal Science Phase (NSP) 4 years

- Daily coverage of 10h per day per spacecraft

Activities: Nominal science operations planning should be simple as there is a single operating mode wherein data is collected, recovery of anomalies, pointing of the TTC, orbit determination and control using tracking data (TBC), offline attitude determination and control based on the attitude sensors data in the s/c telemetry (TBC) and no need of commanded updates of control parameters in the on-board attitude control system. Orbit maintenance manœuvres (no baseline), instrument maintenance activities (TBC).

ESP & DCP Mission Phases Assumptions

Extended Science Phase (ESP, optional) 6 years

- As per NSP

Decommissioning Phase (DCP) ≈ 4 wks (TBC)

- DCP is TBD and G/S coverage should be adapted depending on the criticality.

Activities: (TBC) spacecraft passivation and its effect onto the disposal orbit and the space system fully decommissioned.

Data rates Assumptions

Per Spacecraft	Antenna	Ground [hours]	Bit Rate [kbit/s]	Data Volume [Mbit/day]
LEOP	LGAs	Quasi-continuous	128	(6h=>2765)
Transfer (10Mkm) 1 day/week	LGA	2	2*	16
	MGA	2	128*	922
	HGA	2	140	1008
Science (65Mkm) daily	LGA	10	0,05	2
	MGA	10	13	468
	HGA	10	128	4608

(Assumption: ±1500 bps HKTM generation rate per spacecraft)

* at 10Mkm

Transfer Assumptions (2)

- EP/HGA: 2hours/7days found reasonable (dump all stored HKTM).

If an anomaly occurs after the last communication it would not be noticed for a week, time to analyse the anomaly, recover the spacecraft and resume the transfer without big impacting the transfer duration.

- CP/MGA: 2hours/7days could also be applied (dump partially stored HKTM).

Interposing (EP/LGA & CP/MGA) passes will reduce the communication outage.

Only LGA communications during transfer should be imposed only if strictly necessary (e.g.: power constraints, limitations on #HGA pointing, etc.).

Other Assumptions

- Usage of 15m stations during LEOP can be considered. All deep space communications via 35m stations ESA DSA (Beacon mode with 15m station shall be discussed).
- DSA 35m beam width is ≈0.07deg << 2 deg (s/c inter distance): combining 2 or more s/c into a single DSA not possible.
- The NASA DSN usage (free of charge) as part of an overall cooperation agreement to be discussed.
- Compression using POCKET+ highly recommended.
- Single communication session per day during Science Phase to dump the data from the constellation.
- Safe Mode: dump of the Emergency Log and minimum telemetry that allows ground analyse and recovery.

CP, EP, EP+ Options

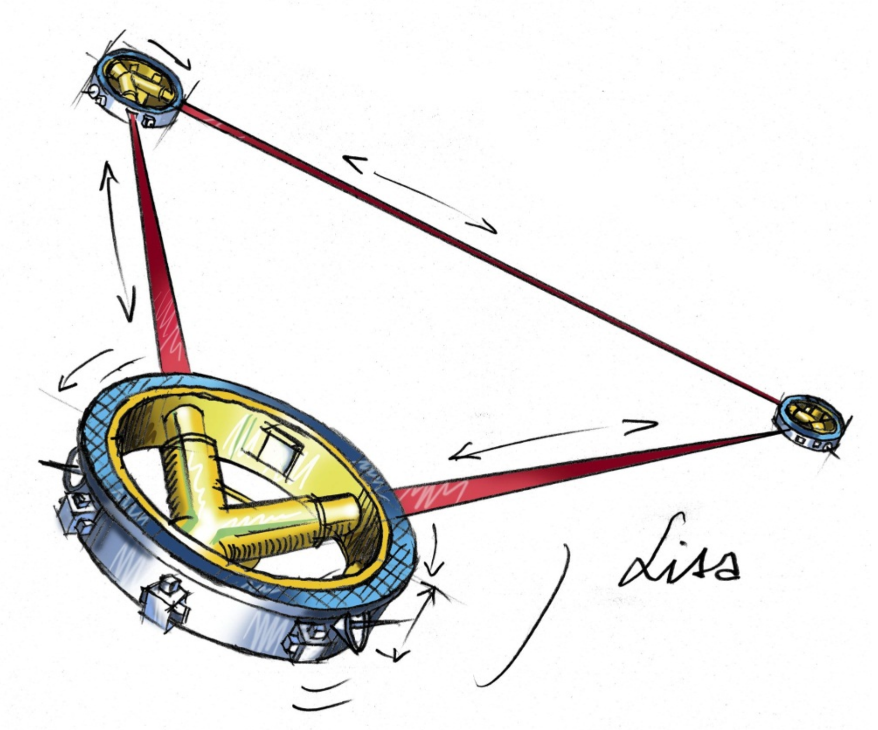
	CP	EP	EP+
Launch Type	Direct Escape	Direct Escape	Direct Escape
Constellation Deployment	☐	☐	☐
Tranfer Trajectory Manoeuvre	1+	1	1
GS Contact during Transfer	2h/wk (MGA)	2h/wk (HGA)	
Interruption of Transfer Man for Comm	N/A	☐	☐
Transfer duration	1 yr	±430 d	±430 d
Orbit Insertion Manoeuvre	☐	☐	☐
Propulsion Module Separation	☐	N/A	N/A
FD: OD, range, range-rate	☐	☐	☐
FCT: nominal/critical operations, MPS	☐	☐	☐
Other Operations Systems	☐	☐	☐

LISA

DFACS / AOCS

Internal Final Presentation
ESTEC, 05-05-2017

Prepared by the CDF* Team



(*) ESTEC Concurrent Design Facility

Outline

- Requirements and Design Drivers per Mission Phase
- Assumptions and Trade-Offs (per mission phase)
- Summary of Design per Option (CP, EP, EP+)
- Conclusions / Open points

Problem Drivers / Requirements

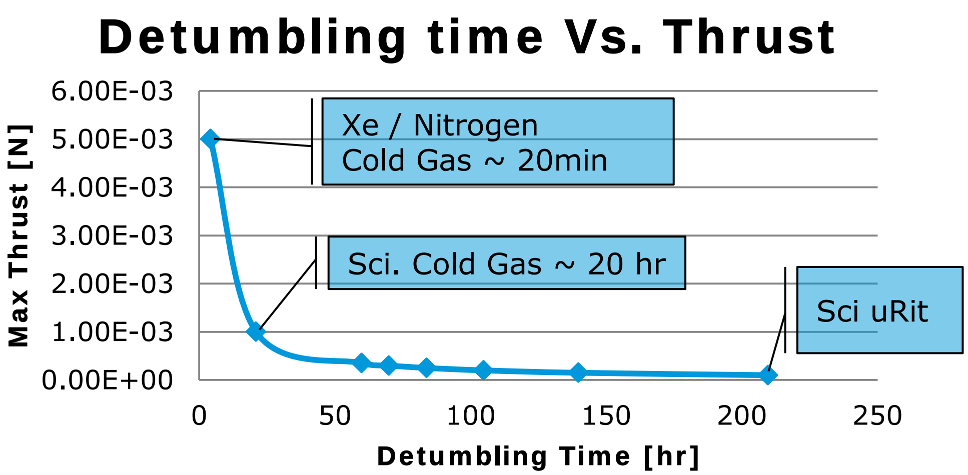
- Control rotational and/or translational degrees of freedom
- Four distinct phases identified with different competing requirements:

	De-tumbling	Cruise	Constellation Acquisition	Science
Main Drivers	Time (no sun guaranteed)	Main engine torque/momentum compensation	Point laser towards other S/C in constellation	Science Performance
Req.	High RCS thrust	- Adequate RCS thrust - High fuel consumption or high ISP	- Hi accuracy short term pointing stability - High accuracy absolute pointing (lower search area)	- LPF Heritage - uN thrusters required High fuel consumption or high ISP - Sensing inside payload *Performance not analyzed

Discussions and Trade-offs per Mission Phase

De-tumbling

- Separation options:
  - Spin stabilized (cylindrical conf.) @ 5deg/s: sun pointing, ~30deg nutation
  - No spin (rectangular conf.), 3deg/s tumbling: sun pointing not guaranteed
- De-tumbling time depends mainly on thruster max thrust

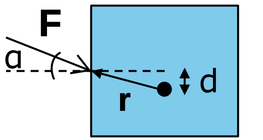


Hardware Required

CP	EP	EP+
De-tumbling RCS		
22N CP +200s ISP	5 mN N2 Cold Gas 40s ISP	5 mN Xe Cold Gas 25s ISP
1.2 kg Biprop	0.62kg Xe	0.9kg Xe
Sun sensor & GYP Required		

- Hi thruster thrust required to reduce de-tumbling time (power, battery sizing).

Cruise



$$\mathbf{T} = \mathbf{r} \times \mathbf{F}$$

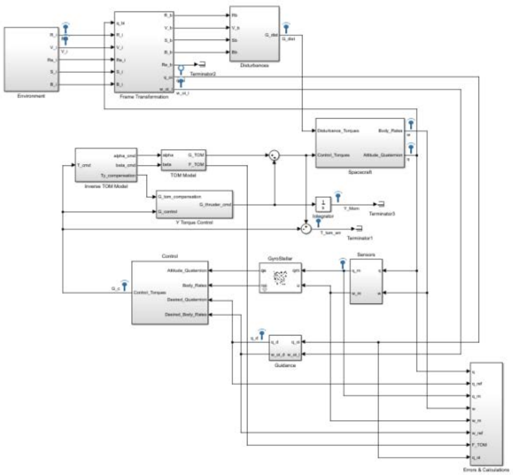
- Main engine misalignment w.r.t CoM produces residual torques
  - Nominal alignment (ground): +1 deg or worse
- Must be compensated to maintain attitude:
  - Thrusters: continuous thrusting (continuous fuel consumption)
  - Off-Pulsed Three/Four Main engine configuration (less fuel)
  - ~~Wheels: absorb momentum, then dump with thrusters (less fuel)~~
  - Main engine Thruster Orientation Mechanism (TOM / gimbal):
    - Minimize misalignment error (~0.1deg) + thruster attitude control
    - Control attitude directly (2-axis) + axial control (thrusters)

	CP w / 4ME	EP w / TOM	EP+ w / TOM
Req. RCS Thrust (50% margin)	2.4 N	670 uN	780 uN (thruster conf)
Fuel Consumption	32 kg Biprop	47kg N2	1.2kg Xe
Sensors	Star Tracker + Gyro		

Cruise Attitude Control w / TOM

- Promising: no fuel consumption to compensate transverse (X-Z) torques
  - Worse attitude control accuracy (Delta-V direction error)
  - Larger axial torques due to CoM offsets
- Simulink simulation created to analyze this in detail including:
  - Disturbances: Solar, gravity gradient (others are negligible)
  - Sensors: Star Tracker & Gyro in a Gyro-stellar estimator
  - TOM Model for X-Z torques (transverse) with:
    - Position of the tom w.r.t. geometric center: [0,-1.6,0]';
    - Max angular displacement: 15deg
    - Angular Resolution: 0.1deg (typical range 0.05 to 0.2 deg)
    - Max Angular Rate: 0.2deg/s (typical range 0.1 to 0.5 deg/s)
  - Inverse TOM model (small angle approx.) to command the TOM.
  - Thrusters (perfect model) for Y torques (axial to engine)
  - Quaternion based attitude control

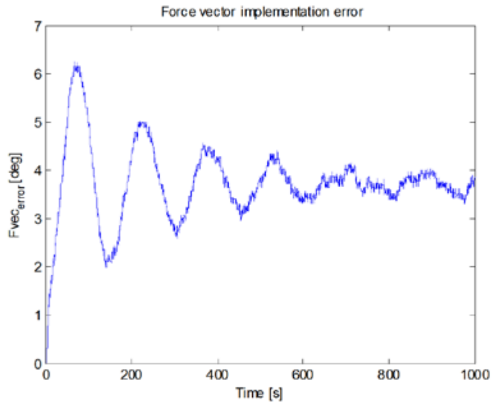
Simulation Model

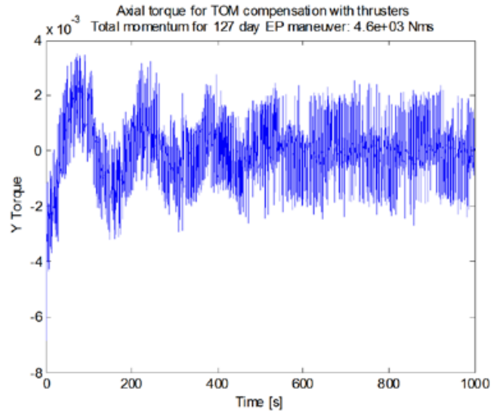


Cruise Attitude Control w / TOM - Results

- Attitude control can achieve ~4deg Delta-V direction error
- Residual Axial Torque ~4mNm (due to CoM offset):
  - Must compensate by thrusters @ 0.75m: 5mN required
- Total momentum ~4.6e3 Nms. Propellant required:
  - 17kg if ISP is 40s (cold gas, but need 5mN thrust)
  - 27kg if ISP is 25s (cold gas de-tumbling thrusters EP)
- => No significant saving with TOM attitude control
- Solution proposed: TOM for alignment compensation only

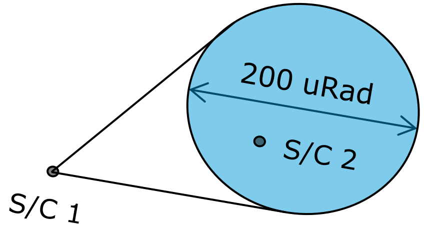
	CP w / 4ME	EP w / TOM	EP+ w / TOM
Req. RCS Thrust (w/ 50% margin)	2.4 N	670 uN	780 uN
Fuel Consumption	32 kg Biprop	47kg N2	1.2kg Xe





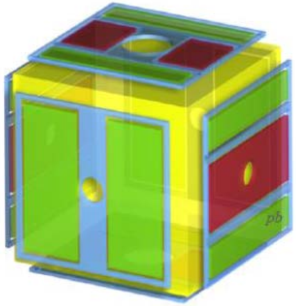
Constellation Acquisition

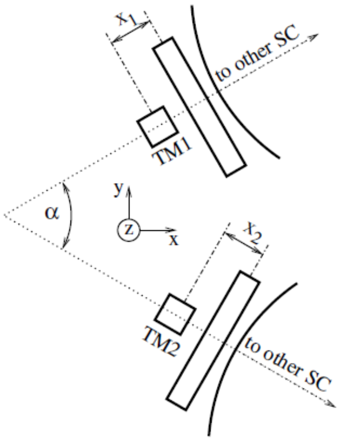
- Process to establish bidirectional laser link between spacecraft
- Must point Laser beam (~5uRad divergence): similar pointing perf. required
- Best achievable pointing errors when commanding by ground: ~175 uRad
  - Ground navigation error @ arm length: ~50 to 75 uRad
  - High Accuracy Star Tracker Errors: ~25 uRad
  - Star Tracker – Telescope alignment (ground): ~100 uRad
- => Scanning of laser beam is required
  - ~90 min for 175 uRad @ ½ beam/s (TBD w/sensor).
- => High accuracy short term attitude stability required: use TM "gyro" mode
  - LPF Suspension drift (disturbance estimation): ~10-100 uRad/h (SCI-ACC)
  - Not feasible to use "dead reckoning" attitude control from ground
- => Need for a detector inside payload with ~1-5 uRad resolution (per pixel)
  - Acquisition Sensor FOV larger than 200 uRad to guarantee other S/C inside FOV when starting acquisition.



Science Drag Free Attitude Control System

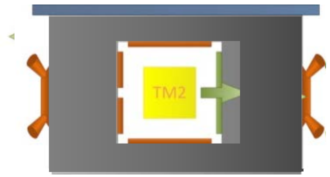
- Control 16 of 19 DoFs system (2x TMs, S/C & inter-telescope angle)
- All sensing inside payload:
  - TM Electrostatic pos/ang (6 axis)
  - TM Optical sensing (1 linear axis, 2 angles)
  - Spacecraft-to-spacecraft angles (2x 2-axis DWS)
- Actuators:
  - MPS (uN thrusters) on S/C
  - Electrostatic system for TM control (in payload)
  - Inter-telescope angle (alpha) actuator in payload

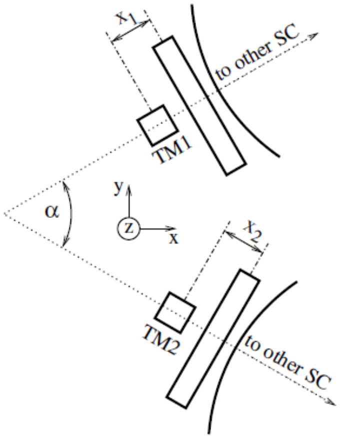




Science Drag Free Attitude Control System

- During science, the Drag Free Attitude Control System (DFACS)
  - Controls S/C position w.r.t TM in inter-s/c LOS directions (possibly also Z axis to average of TM1 & 2 Z axis)
  - S/C Attitude controlled using DWS (laser) angular measurements w.r.t other S/C (2x 2-angles)
  - Inter-telescope angle (alpha) to maintain inter-s/c laser pointing
  - All other TM DoFs (attitude + position) controlled electrostatically to the S/C
- For reference, LPF controls 15/19 DOF, no telescopes (no alpha)
- DFACS only uses payload sensors & actuators + MPS
- Science transition modes required to maintain control of all axis
- Performance analysis out of the CDF scope



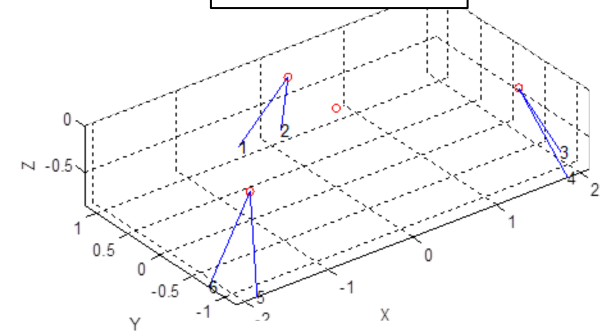


Science Sizing Drivers

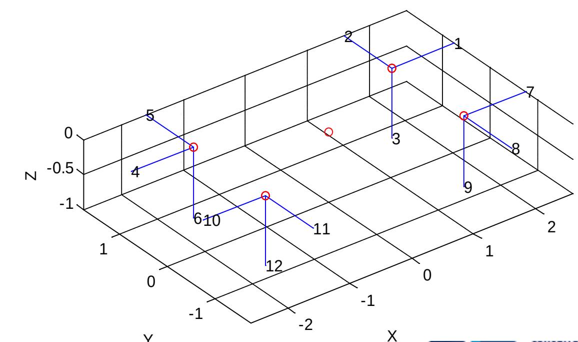
- DFACS Needs to compensate for:
  - Solar Radiation Pressure Force (30% margin)
    - ➢ Dependent on sun-shield area / thruster geometry
  - Solar Radiation Pressure Torque (50% margin)
    - Due to Center of Pressure – CoM misalignment
    - Due to 30deg sun off-pointing (constellation geometry)
    - ➢ Compensation efficiency given by thruster arm
  - Antenna repointing (100% margin)
    - ➢ Dependent on antenna inertia
  - Test Mass DC (S/C self gravity) forces
    - For drag-free axis (100% margin)

	CP w / CGAS	EP w / CGAS	EP+ w / EP
Propellant Required (with / no margins)	194 kg / 141 kg	232 kg / 167 kg	12 kg / 9 kg

LPF MPS Configuration



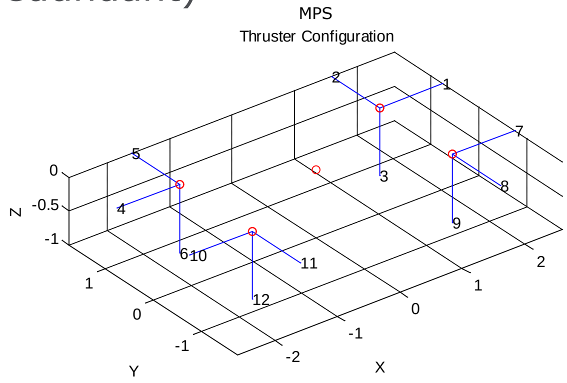
Axis	Force	Torque
X-Y	0.34	0.87
Z	0.87	0.34



Summary of Design Options

CP Option

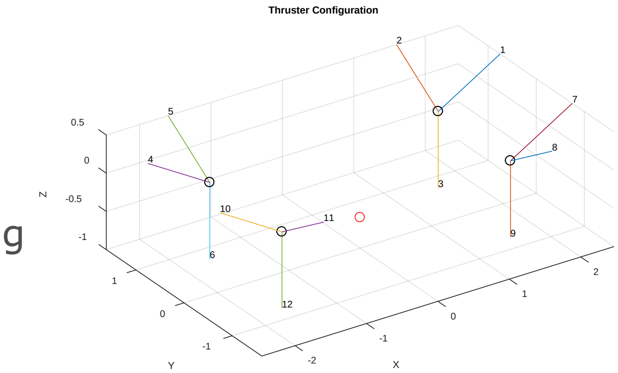
- Sensors (in addition to payload):
  - 4x Star Trackers (2 in sci-craft, 2 in Prop Module), cold redundant
  - 9x Sun Sensors (6 in sci-craft, 3 in Prop Module) triple majority voting
  - 2x Gyros in Science-craft
- Actuators (in addition to payload):
  - 24 500uN+ Cold Gas Thrusters in sci-craft (12 cold redundant)
  - 8 RCS Thrusters in prop module (4 cold redundant)
- Propellant:
  - 200 kg / 152 kg (no margin) Cold Gas for DFACS
  - 36 Kg of RCS bipropellant for LEOP/Transfer



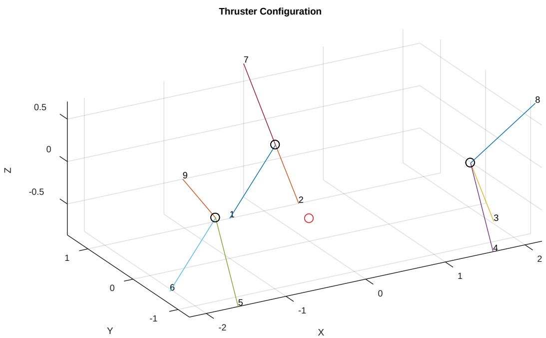
Min Torque Authority: 750 uNm

EP Option

- Sensors (in addition to payload):
  - 2x Star Trackers (in sci-craft), cold redundant
  - 6x Sun Sensors (in sci-craft) triple majority voting
  - 2x Gyros in Science-craft
- Actuators (in addition to payload):
  - 24 500uN+ Cold Gas Thrusters in sci-craft (12 cold redundant)
  - 8 Xenon Cold Gas for de-tumbling (4 cold redundant)
  - Main EP Thruster with TOM to manage misalignment with CoG
- Propellant:
  - 232 kg Cold Gas for DFACS (Science) / 167kg (10 years no margins)
  - 68 Kg of Cold Gas for de-tumbling & transfer (including EP maneuvers)
  - Total: 299 kg N2 (10 years full margins) / 235 kg (no science margins)



EP+ Option



- Sensors (in addition to payload):
  - 2x Star Trackers (in sci-craft), cold redundant
  - 6x Sun Sensors (in sci-craft), triple majority voting
  - 2x Gyros in Science-craft
- Actuators (in addition to payload):
  - 18 100uN uRIT Thrusters in sci-craft (9 cold redundant)
  - 8 Xenon Cold Gas for de-tumbling (4 cold redundant)
  - Main EP Thruster with (TOM) to manage misalignment with CoG
  - 8 High-Thrust (1mN) nRIT thrusters for attitude control during EP burns
- Propellant:
  - ~ 4 kg Xenon for DFACS (Science)
  - ~ 9 Kg of Xenon for Transfer
  - ~ 2 Kg of Xenon for de-tumbling

~ 15 kg Xe

Conclusions & Open Points

- Feasible options found for the mission:
  - Most promising: EP+ (full electric… almost)
  - Science Heritage option: EP (using LPF Cold Gas system)
- Thruster configuration should be optimized (especially for EP option)
  - Canting vs. number of thrusters & bias efficiency (min thrust)
    - Compute efficiency maps for different configurations
- Number of different thrusters for EP+ option should be optimized
- Analyze Science (DFACS) Performance for all options (EP or Cold gas thrusters)
  - Preliminary performance (noise) budget
- …

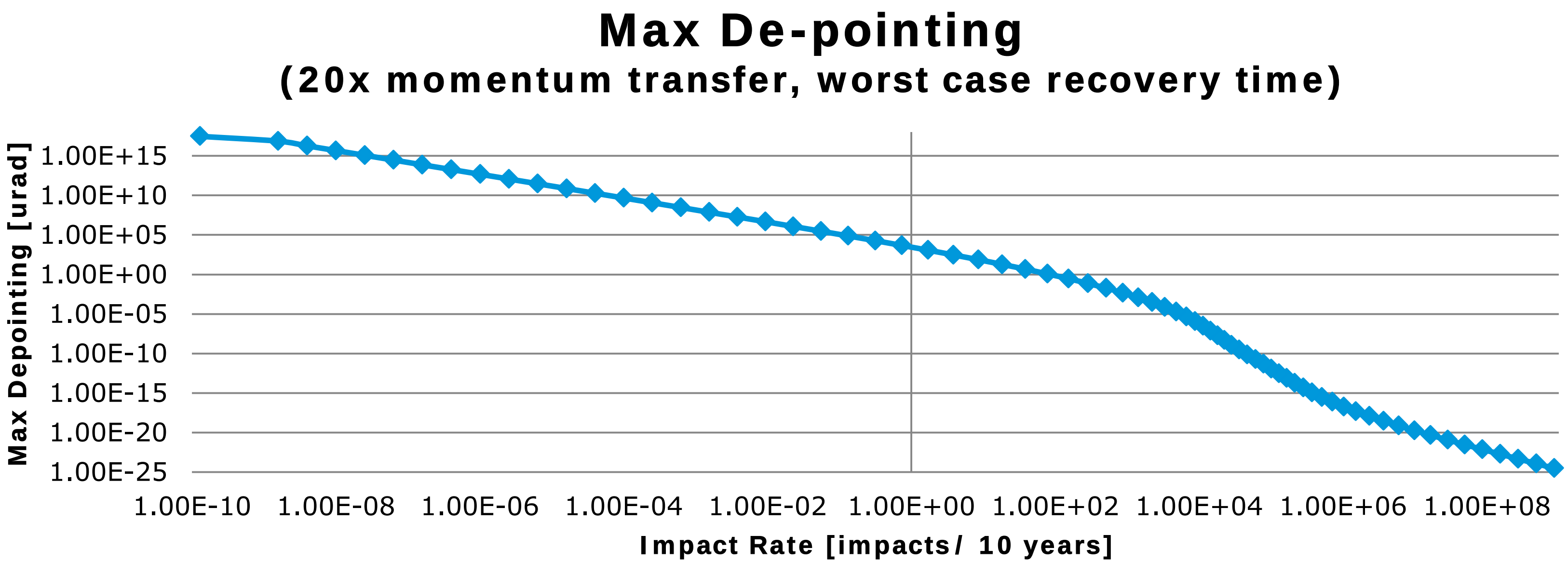
Thank you!

Back-up Slides

DFACS / AOCS Modes

Mission Phase	FUNCTION	MODE	Notes
Launch	AOCS	Standby	Propulsion module attached (if any)
Transfer		Attitude	
Orbit Insertion Maneuvers		Maneuver	
Anomaly		Safe	
Commissioning	DFACS	Attitude	Sciencecraft only (no propulsion module)
Constellation Acquisition		Scanning	
TM Release		Accelerometer	
Science		Drag Free	
Anomaly		Safe	

Spacecraft De-pointing vs. Impact Rate



- Loss of laser pointing (10urad) ~ once per month => Interruption in science
- One full safe mode (10 mrad de-pointing) every 10 years
- Loss of mission (10 deg de-pointing) once every 1000 years.

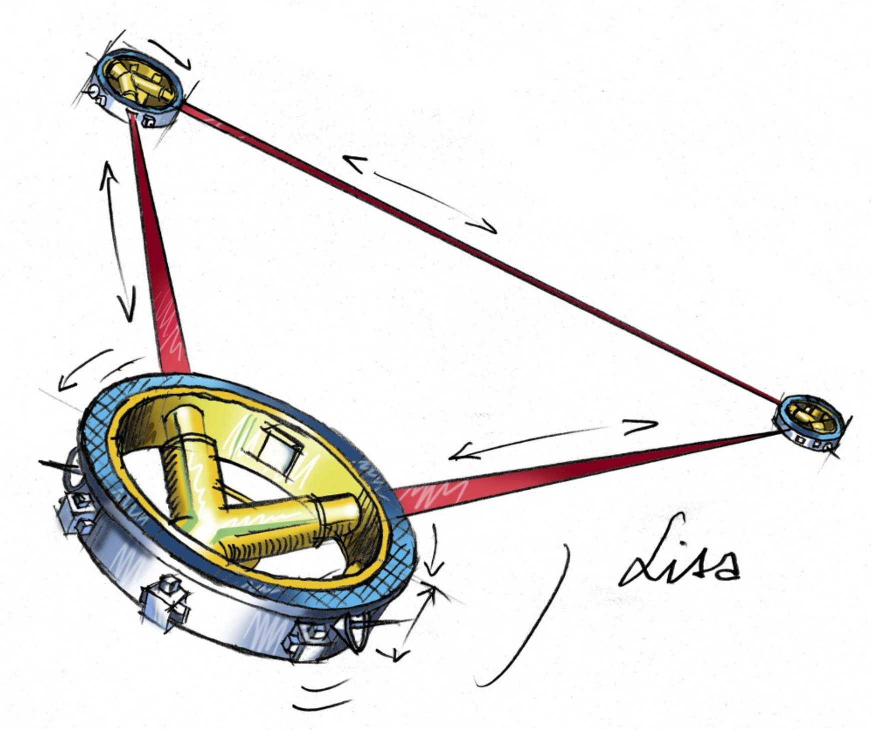
LISA

Chemical Propulsion

Internal Final Presentation
ESTEC, 05-05-2017

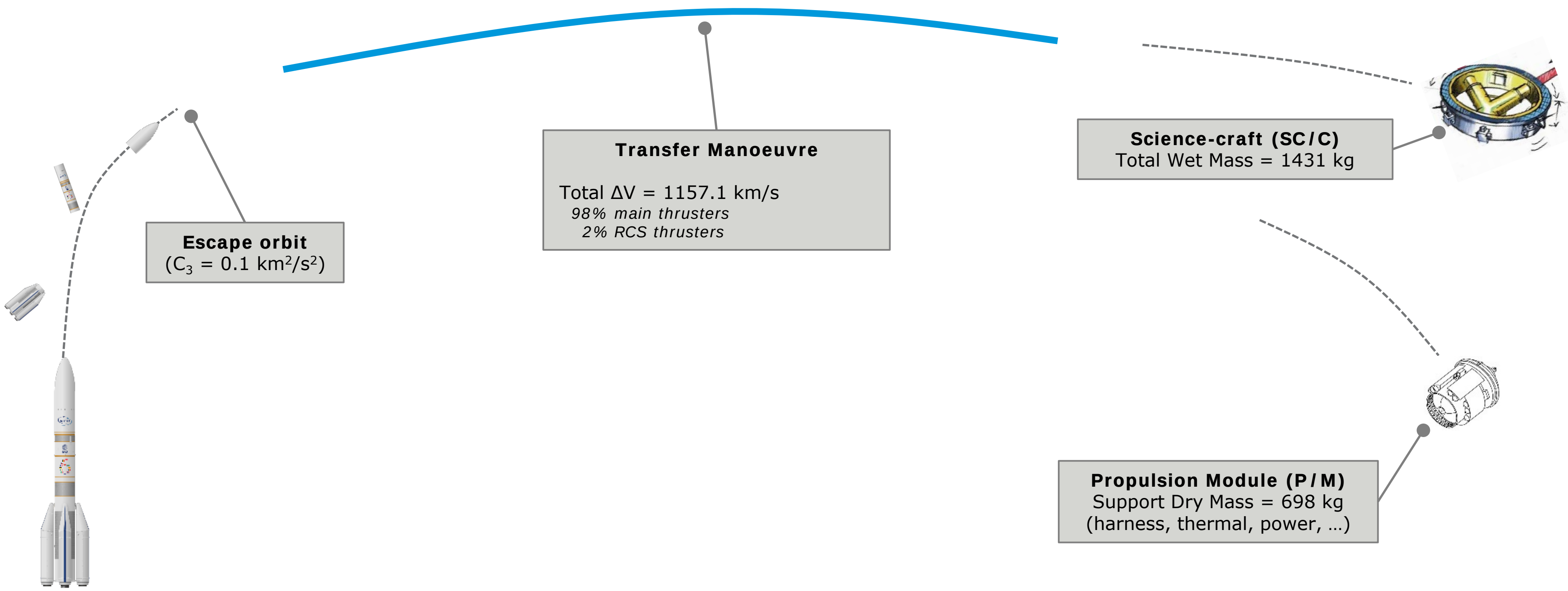
Prepared by the CDF* Team

(*) ESTEC Concurrent Design Facility



P/M Configuration for Transfer

The propulsion module (P/M) performs the orbit transfer of the science-craft (SC/C), needs to deliver main thruster and RCS burns, and is discarded before the science mode.



Propulsion System – Architecture Trade-offs

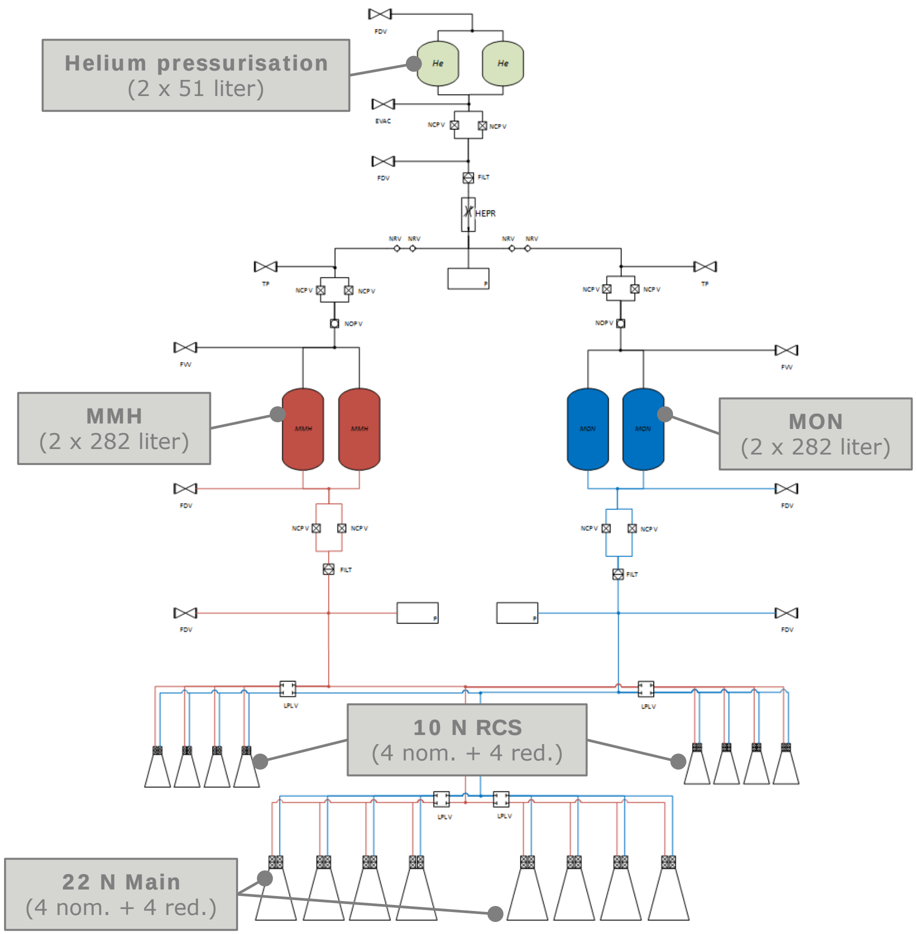
	Option A	Option B	Option C
Main Thruster	4 x 20 N Mono-Prop	400 N Bi-Prop	2 x 50 N Mono-prop
Propellant(s)	N2H4	MON+MMH	LMP-103S
I_{sp} [s]	218	321	256

Due to mass criticality, high I_{sp} **bi-propellant system (MON/MMH)** is chosen.

	Configuration B1	Configuration B2
Main Thrusters	1 x 400 N	(4+4) x 22 N
RCS Thrusters	(4+4) x 22 N	(4+4) x 10 N
I_{sp} [s]	321 (main) / 300 (RCS)	300 (main) / 291 (RCS)
Total Thrusters Mass [kg]	9.7	10.6
Propellant Mass for transfer [kg] (excl. margin & resid.)	971.1	1054.0
Max. thrust-on time [h]	1.9 (main) / 0.2 (RCS)	9.6 (main) / 0.4 (RCS)

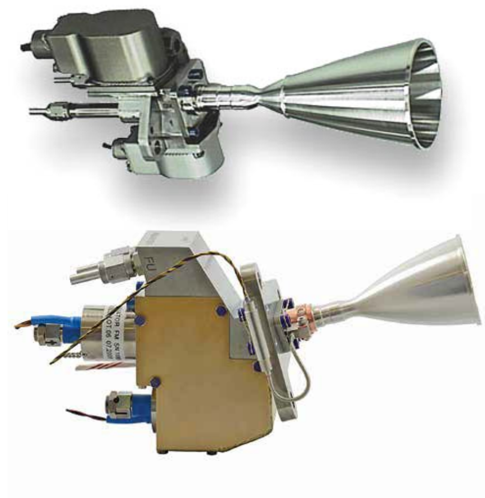
Due to potential issues with physical configuration, it was opted for **several smaller** (22 N) thrusters **instead of a larger** (400 N) thruster for the **main transfer burns**.

Propulsion System – Baseline Architecture



- Bi-propellant (MON / MMH) system
- Pressurisation using Helium
- 8 (4 nom. + 4 red.) 22 N thrusters
- 8 (4 nom. + 4 red.) 10 N thrusters

Proposed Airbus DS bi-propellant thrusters: 10 N (top) and 22 N (bottom)



Propulsion System – Mass Budget

- Total Propellant Mass (MON+MMH): **1079.2 kg**
  (excl. residuals, incl. AOCS)
- Pressurant Mass (He): **4.1 kg**

	Incl. Margin	
Mass Thrusters & Tanks	*122.9*	*[kg]*
Mass Valves, Regulators,…	*10.3*	*[kg]*
Propellant Mass	*1079.2*	*[kg]*
Residual Propellant Mass	21.6	*[kg]*
Pressurant Mass	*4.1*	*[kg]*
P/M Mass (dry)	697.6	*[kg]*
Total Mass (dry)	*2128.3*	*[kg]*
Total Mass (wet)	***3243.6***	*[kg]*

- 2 x 282 l Tank for **MMH**
  - Total mass: **42 kg**
  - Fill level: 94 %
- 2 x 282 l Tank for **MON**
  - Total mass: **42 kg**
  - Fill level: 94 %
- 2 x 51 l Tank for **He**
  - Total mass: **22.4 kg**

- Pyro, Latch, Fill & Drain, Non-return valves = **4.9 kg**
- Filters, press. transducers, press. regulator = **4.9 kg**

Propulsion System – Hardware / Equipment

Component	Mass [kg]	Mass Margin [%]	P_on [W]	P_stdby [W]	# items	Total Mass [kg]	Red k	TRL
Propellant Tank	21.000	5			4	88.200		9
Pressurant Tank	11.200	5			2	23.520		9
Thrusters (22N)	0.680	5	41	0	4	2.856	4	8
Thrusters RCS (10N)	0.650	5	30	0	8	5.460	4	9
Pressure Regulator	4.000	20	15	5	1	4.800		6
Pressure Transducer (HP)	0.125	5	0.3	0.3	1	0.131		9
Pressure Transducer (LP)	0.250	5	0.8	0.8	2	0.525		9
Filter (HP)	0.076	5			1	0.080		9
Filter (LP)	0.117	5			2	0.246		9
Pyrovalve (NO)	0.155	5	0	0	2	0.326		9
Pyrovalve (NC)	0.150	5	0	0	10	1.575	5	9
FD/FD/TP Valve	0.050	5			11	0.578		9
Non-Return Valve	0.085	5			4	0.357		9
Latch Valve (LP)	0.550	5	30	0	4	2.310		9

Mostly proven technology with flight heritage. Minimal technology development needed.

Coldgas System: Design Drivers

Two different thrust levels:

1.) High thrust level for detumbling phase: mN (EP and EP+)

→12 thrusters (+ 12 redundancy)

2.) Low thrust levels for EP transfer phase and DFACS in science phase: < 10 µN

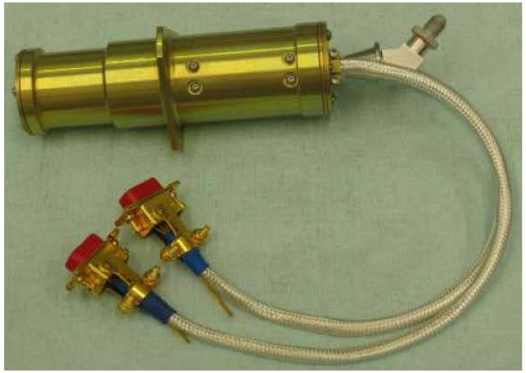
→ 4 thrusters (+ 4 redundancy)

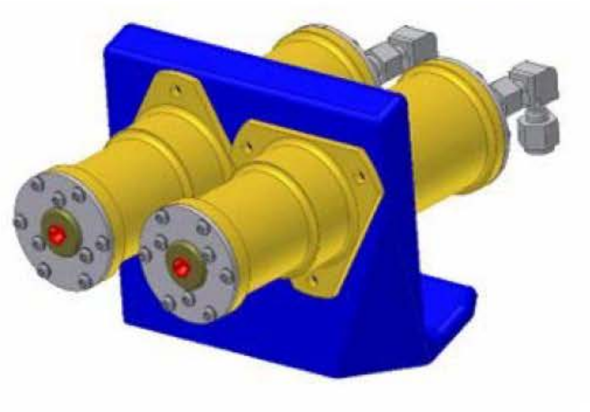
Requirements for thrusters	
Thrust level (science mode)	> 10 µN
Thrust level (detumbling)	>250 µN – 50 mN
Total impulse	11000 Ns
Thruster update rate	10 Hz
Thrust resolution	0.1 µN
Noise	Same as LPF

TAS-I: Micro Cold Gas Thruster

Thrust Range	1 – 500 μN
Thrust Resolution	Meet requirements
Isp	> 45 s
Noise Level	Meet requirements
Provided Lifetime	60000 h
TRL	9
Heritage	Gaia, Euclid

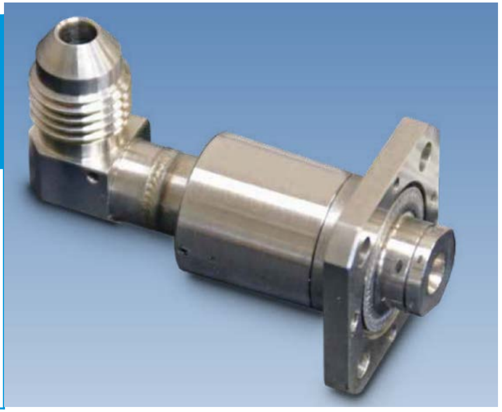
- Piezo actuated proportional valve and flow sensor
- Closed loop control of mass flow and thrust
- Temperature monitoring of valve and nozzle for flight corrections





50 mN Thruster for detumbling phase

Thruster	Thrust Range	Specific Impulse	Thruster Mass	Other missions
Moog SVT01 Cold Gas Thruster	10 – 50 mN	72 s (Nitrogen) 33 s (Xenon)	0.1 kg	**CryoSat-1 / -2, TanDEM-X, Swarm**

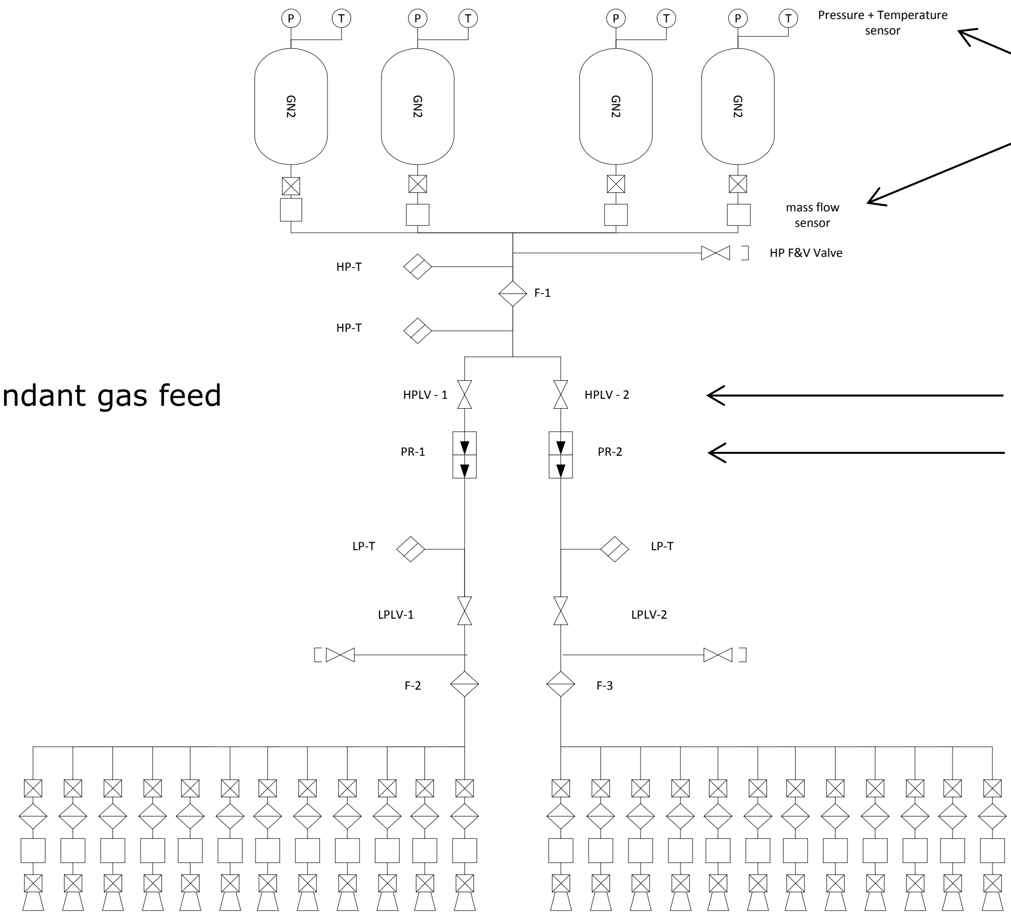


SVT01 Cold Gas Thruster

Flow Schematic: CP

Similar to Euclid

Primary and redundant gas feed



measurement of propellant mass in each tank

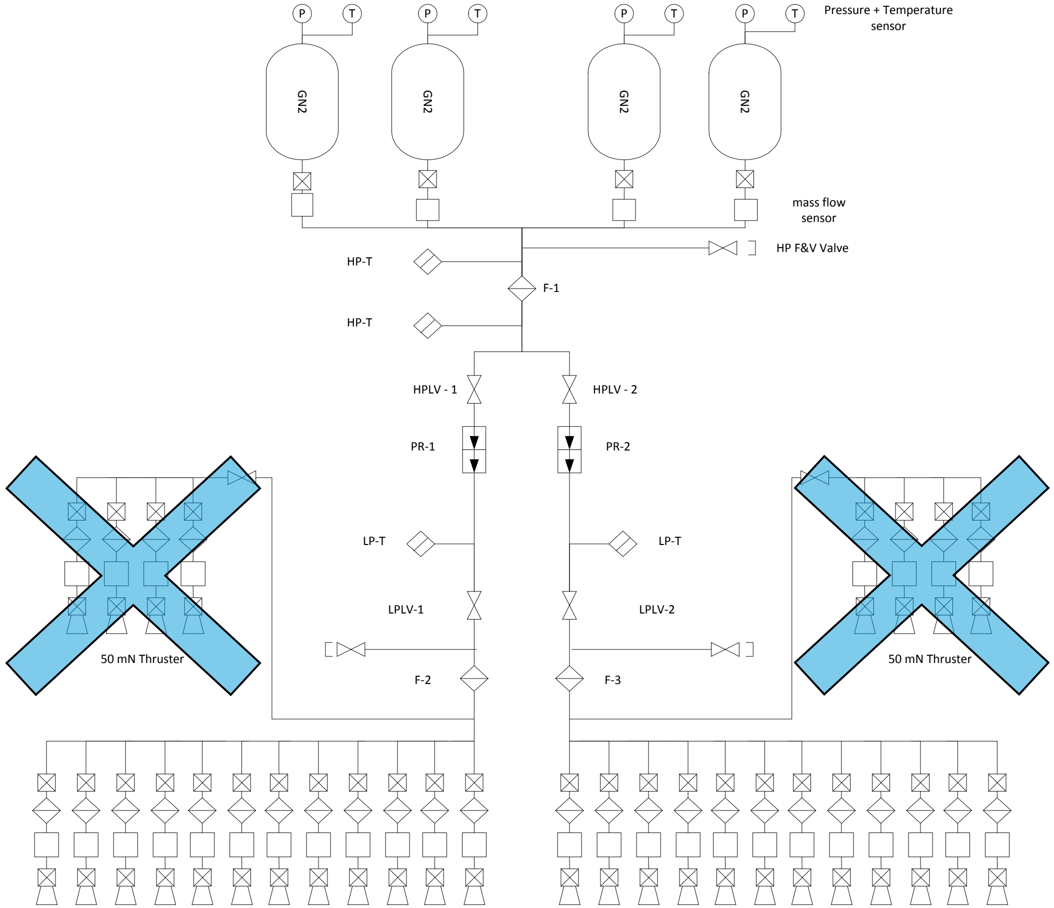
isolating latch valve

electronic pressure regulators

Micro Propulsion Thruster

Chemical Propulsion

Flow Schematic: EP



Detumbling thrusters included in EP system: Higher storage capability for Xenon

Requirements and Design Drivers

Requirements for Tanks	
Max. Diameter Tanks	750 mm
Number Tanks	4
Tanks	Engaged individually
Tank Temperature	310 K

Tank

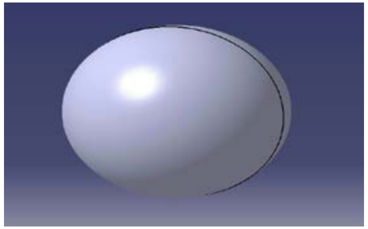
Artes Development: HeHPV (Helium High-Pressure Vessel)

- European development
- Nominal MEOP: 310 bar
- Burst Pressure > 600 bar

Maximum Tank: 239 kg Cold Gas

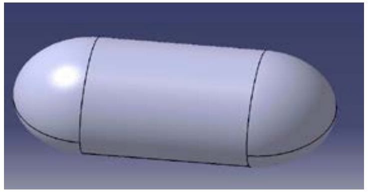
Spherical Tank

- Diameter: 733 mm
- TRL: 6
- Mass: 33.2 kg (w/o margin)

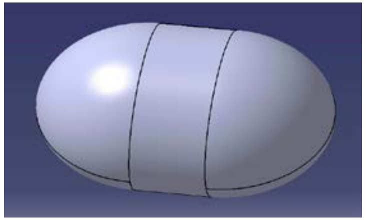


Other tank designs possible:

- Diameter: 0.5 m
- Overall length: 1.214 m



- Diameter: 0.6 m
- Overall length: 0.927 m



Conclusions Cold Gas

Cold Gas System Dry Mass + Margin	**152.1 kg**
Mostly proven technology with flight heritage	TRL 8 – 9
Tank development	Highest development needs
Cold Gas Thrusters	Meet all requirements - Improvement possible

Summary

- Several options were considered:
- Hydrazine monopropellant (transfer)
- NTO/MMH bipropellant (transfer)
  - 400 N main engine
  - 4 x 22 N thrusters
- Green Propellants (transfer)
- Nitrogen cold gas (science, EP transfer)

- **Due to high science payload mass per spacecraft CP mass budget exceeds launcher payload capacity**

- Nitrogen cold gas system requires very large tanks due to high propellant mass demand for 10 year science operation and storage temperature of 310 K

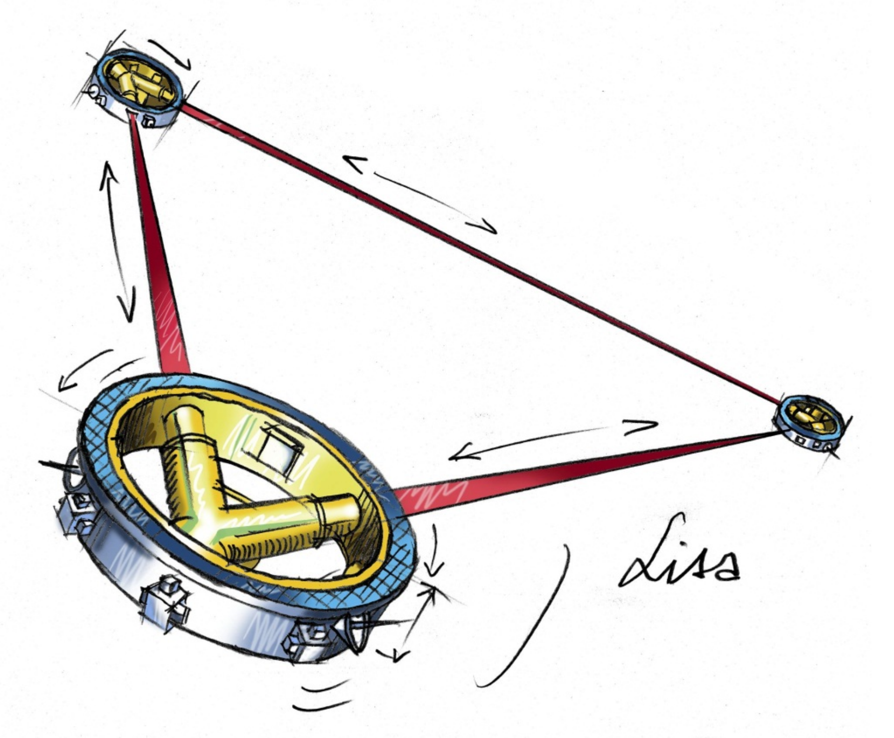
LISA

Electric Propulsion

Internal Final Presentation
ESTEC, 05-05-2017

Prepared by the CDF* Team

(*) ESTEC Concurrent Design Facility



OUTLINE

- Requirements and Design Drivers
- Assumptions and Trade-Offs
- Baseline Design
- Equipment list
- Options

EP for Transfer

Requirements and Design Drivers

- minimum thrust of 50 mN
- overall EPS power shall be minimized (Isp reduction is possible to reduce required power)
- initial mass of 1150 kg
  - ⇒ **EP option: 1500 kg**
  - ⇒ **EP+ option: 1380 kg**
- thrusting time of around 140 days (=3360 h)
  - ⇒ **EP option: 275 days (=6600 h)**
  - ⇒ **EP+ option: 229 days (=5500 h)**
- $\Delta v \approx 950$ m/s
  - ⇒ **EP option: 1164 m/s**
  - ⇒ **EP+ option: 1153 m/s**

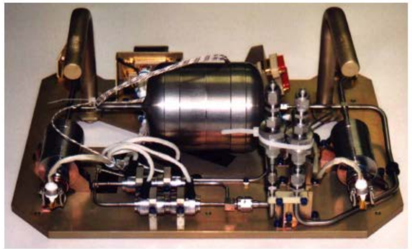
Assumptions and Trade-Offs

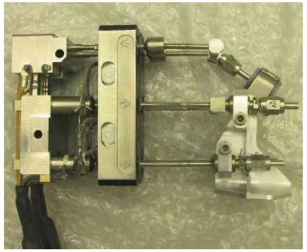
- Technology with high thrust-to-power-ratio to be selected
    - ⇒ Hall Effect Thruster (higher T/P-ratio than gridded ion engine)
- Technology with high TRL (ideally flight heritage) to be selected
- Available power: 1.5 kW
- Fully redundant system

Baseline Design

SMART1 Propulsion system as baseline: PPS®1350 (BUT: incl. redundant system)

- development status: fully developed
- Thruster flight heritage: SMART1, telecommunication satellites (over 30 thrusters flown or ordered)
- Thruster nominal operating point: P = 1.5 kW, T = 90 mN, Isp = 1650 s
- total impulse: 3.39 MNs (10530 h, 7300 cycles, > 5000 h demonstrated in-flight)
- variable input power: on SMART1 operated at 1.42 kW (70 mN, 1610 s)
- PPS®1350-E: up to 2.5 kW achieving 140 mN and 1800 s


BPRU


XFC


PPU

FU


PPS®1350-G

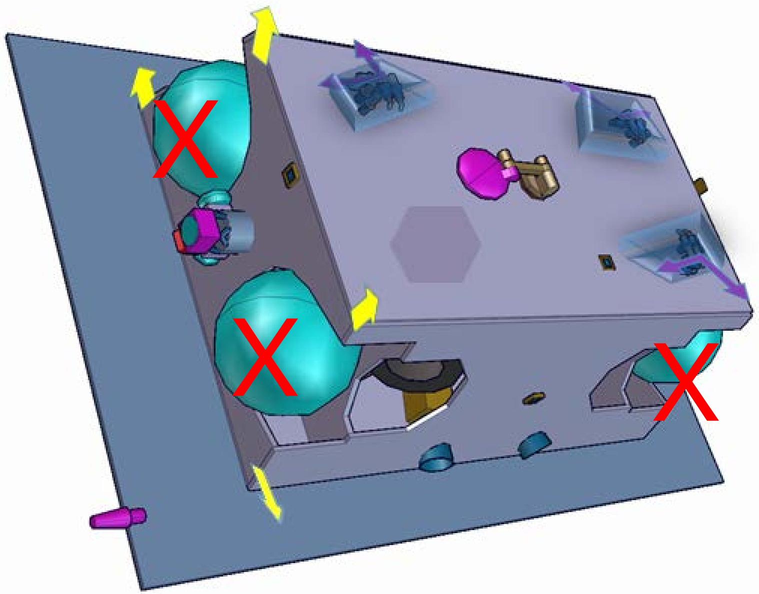
Equipment list

Item	Quantity	Mass per unit [kg]	Total Mass [kg]
PPS®1350	2	4.35	8.7
PPU	2	10.66	21.32
XFC	2	0.82	1.64
FU	2	0.675	1.35
BPRU	2	2.75	5.5
PRE Card	2	1.27	2.54
Miscellaneous	1	3.5	3.5
Tank	4	7.7	30.8
Total			**75.35**
Total incl. margins			**80.68**

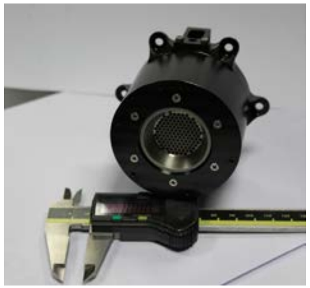
	EP option	EP+ option
Propellant mass for transfer [kg]	145	114.5
Propellant mass incl. margins [kg]	147.9	116.8

Micropropulsion System for EP+ Option

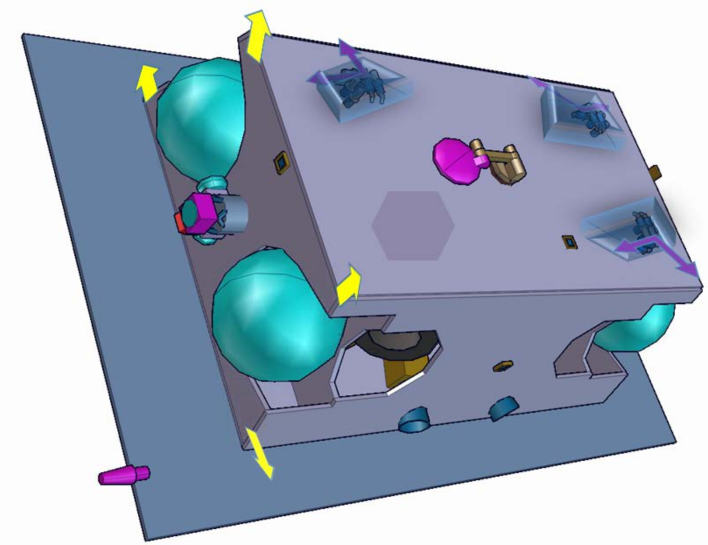
- Requirements and Design Drivers
- Architectures and Trade-Offs
- Options



Mini Ion Engines (1000µN; NGGM)



Parameter	Unit	Collinear Lateral Thrusters	
Minimum Thrust	mN	0.05 (0*)	
Maximum Thrust	mN	>2.5	
Thrust Resolution	µN	0.5	
Thrust Noise		<1µN/√Hz above 0.08Hz	
Rise/Fall Time	ms	< 50	
Slew Rate	mN/s	> 0.5	
Update command rate	Hz	10	
Thrust non linearity		< 2%	
Lifetime	yr	> 10	
Specific Power	W/mN	< 40	
* Thrust has to be turned off completely if thruster is not operating			



Requirements and Design Drivers

- The Micro-Propulsion Subsystem (MPS) requirements for LISA:

  - Thrust level during science: 0-100uN
  - Thrust level during transfer : 100/1000µN
  - Total impulse: 10 years of science operation
  - Thruster update rate: 10 Hz
  - Thrust resolution: 0.1 µN
  - Noise: The same as for LPF

Requirements and Design Drivers

Phase	Activities	Duration [days]	Thrusters in use
LEOP	Detumbling	2	Xe cold gas
Cruise EP off	Sun Pointing, periodic pointings towards Earth (once per week)	150	6xminiRIT100uN
Cruise EP on	EP thrusting, periodic pointings towards Earth (once per week)	300	6xminiRIT100uN + 4xminiRIT1000uN
Acquisition and commisioning	Scan manoeuvres, Science attitude maintenance, SRP compensation, antenna repointing	180	6xminiRIT100uN
Science	Scan manoeuvres, Science attitude maintenance, SRP compensation, antenna repointing	1460	6xminiRIT100uN
Extended Science	Scan manoeuvres, Science attitude maintenance, SRP compensation, antenna repointing	2190	6xminiRIT100uN

Assumptions and Trade-Offs

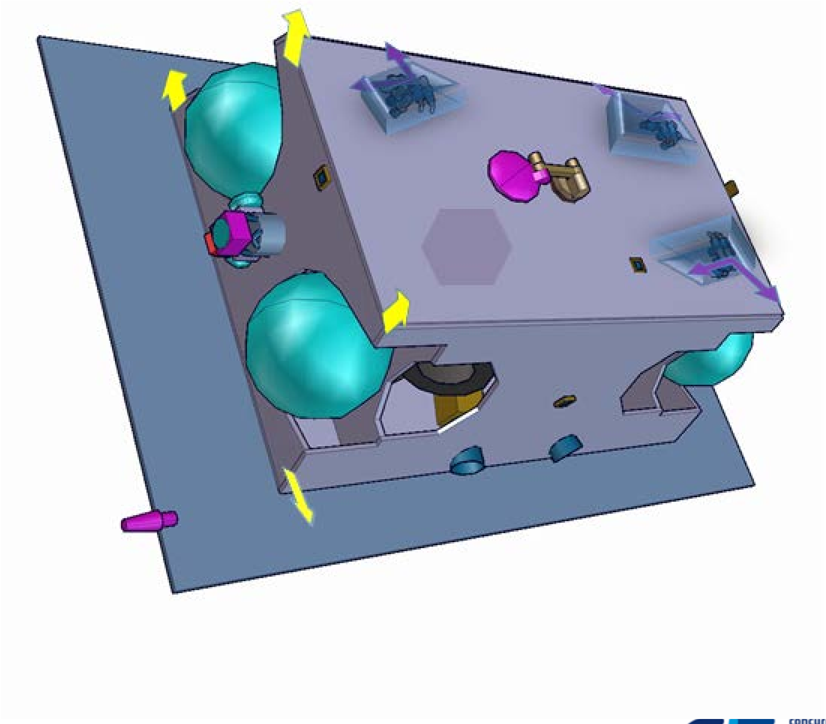
Four MPS options evaluated for LISA:

- MPS based on 6+4 MiniRIT thrusters
  - Developed under GSTP, DLR, EOP up to EM (TRL 5/6)
  - Assessed for LPF and EUCLID
- MPS based on 6+4 Indium FEEP thrusters
  - Developed under GSTP, EOP up to flight model (not for LISA Requirements)
- MPS based on 6+4 Colloid Thrusters
  - NASA Technology, Flown on LPF
- MPS based on 6+4 Caesium FEEP thrusters
  - Developed under LPF project
  - Qualification on hold due to the growing leak current (Iacc) observed during pre-qualification

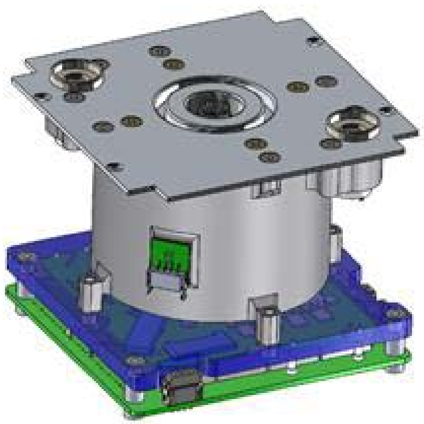
Configuration for science to be updated to 9+9 thrusters

µpropulsion Architecture with 6 / 12 thrusters

- Very high necessary Thrust Dynamic (1-100µN)
- Minimum propellant consumption
- Power consumption during Transfer and Science Modes
- Single PPU for all µThrusters
- Applicable for all EP systems



In-FEEP (FOTEC)



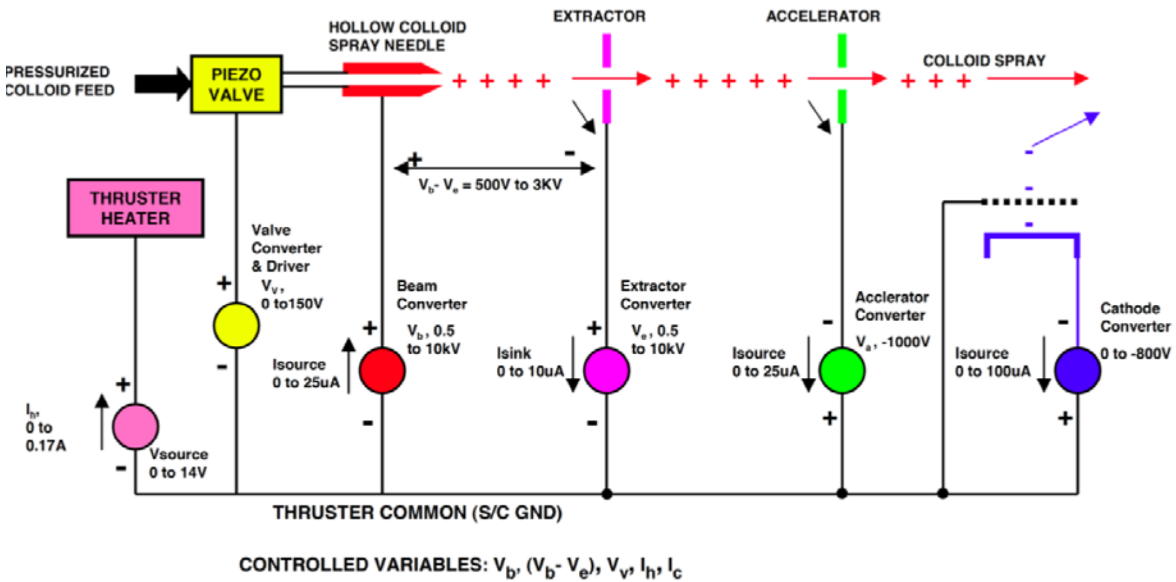
Component	Mass (g)
Thruster including extractor and PPU	640
Propellant	108
Propellant margin 10%	10.8
Total	758.8

Parameter	LISA	FOTEC FEEP
Volume	TBD	1 l
Dry mass	TBD	640 g
Thrust range	1 – 100 µN	1 – 100 µN
Thrust noise	TBD	TBD
Specific impulse at 20 µN	TBD	6000 s
Power at 20 µN	TBD	7 W
Total impulse	6.3 kNs	6.3 kNs

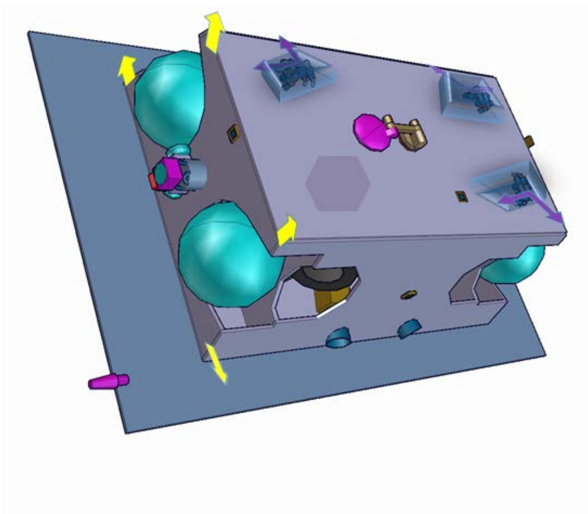
Avg. Thrust Per Thruster	0	10	20		100	250
Power (W)	72	78	84		156	252

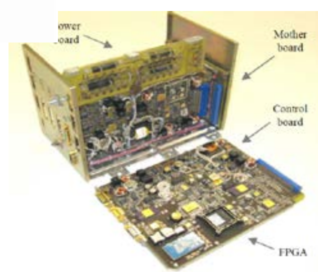
Colloidal Thrusters

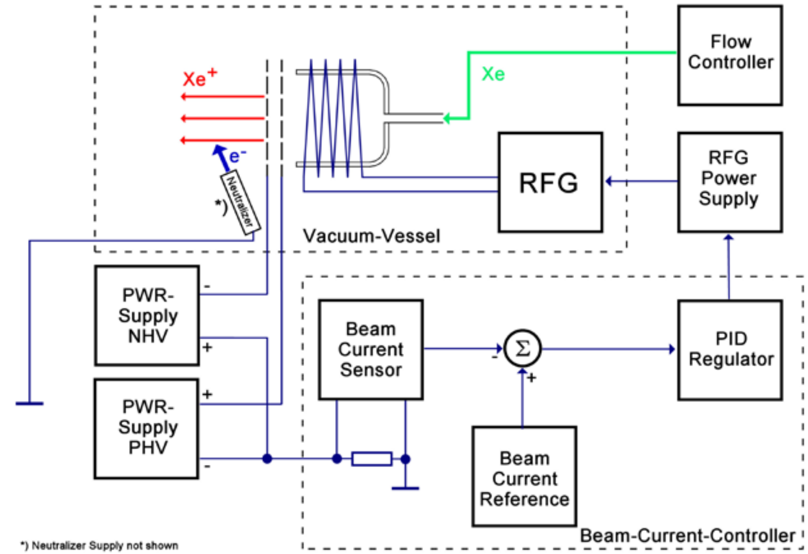
- Colloidal thrusters are part of the ST7-DRS payload on LPF
  a. This is an AOCS payload, using the same LTP inertial sensor, but different control system and colloidal actuators

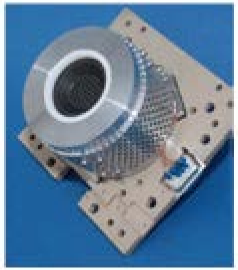


µ Ion Engines

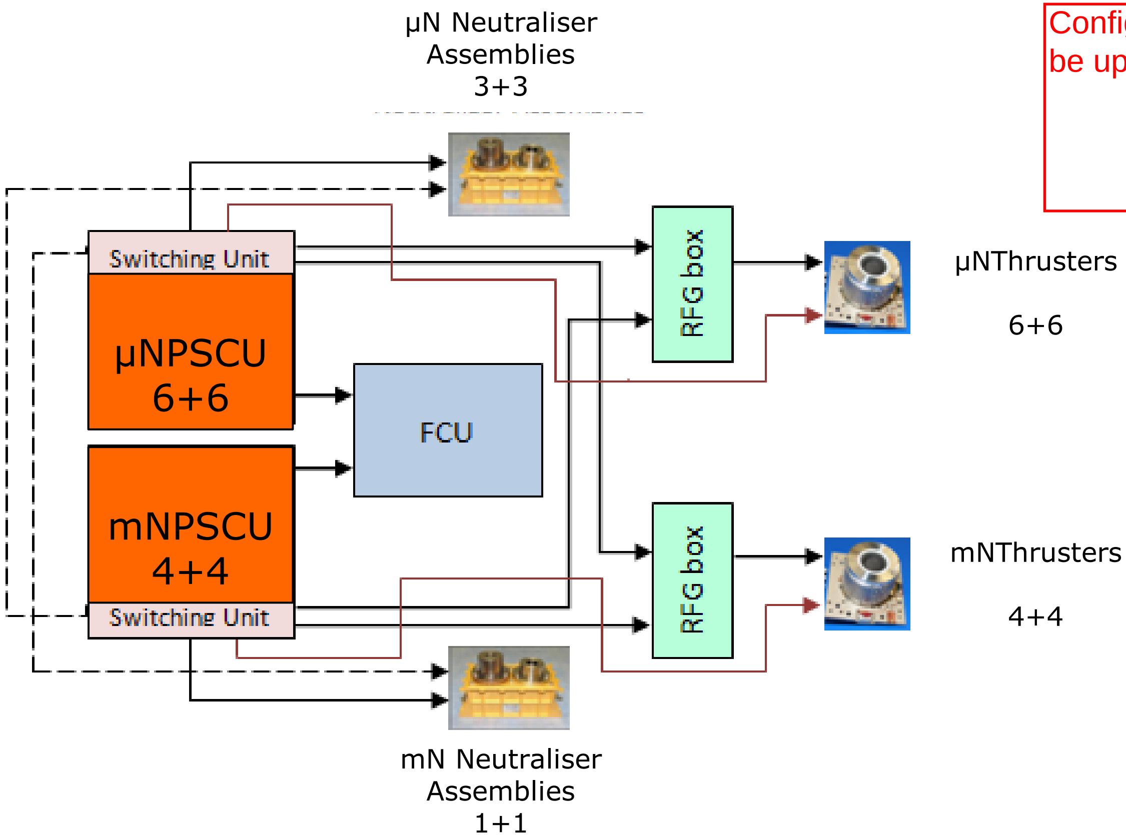






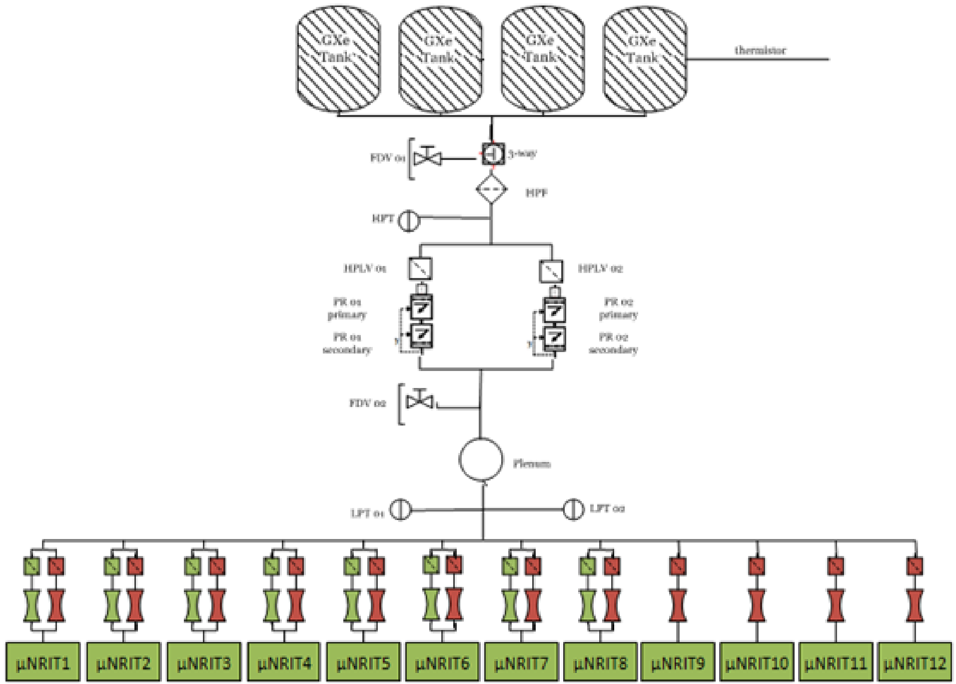


Subsystem block-diagram



Configuration for science to be updated to 9+9 thrusters

Configuration for science to be updated to 9+9 thrusters



Flow-Control Schematic

Mass Budget for Architecture F

Configuration for science to be updated to 9+9 thrusters

Architecture F							
Unit	**Unit Mass (g)**	**Qty**	**Tot Mass (g)**	**Equip Cat**	**Margin**	**Total mass with margin (kg)**	**TRL**
PCUs	9000	3	27000		20%	32.40	6 for LPF
swithcing unit	3000	3	9000		20%	10.80	5 for LPF
RFGs	500	20	10000		20%	12.00	6 for LPF/EUCL
Harness	1349	1	1349		20%	1.62	guesstimated
miniRITs	300	20	6000		20%	7.20	6 for µThruste (Euclid/NGGM)
Thruster supports	177	20	3540		20%	4.25	6 for LPF
Neutraliser Assemblies	438	3	1314		5%	1.38	6 for LPF
Tank & support	4000	0	0		5%	0.00	9
3-way hand valve	400	0	0		5%	0.00	9
HP FDV	45	0	0		5%	0.00	9
µFCu	150	20	3000		5%	3.15	6 for EUCLID
HP Pressure Transducer	265	0	0		5%	0.00	9
HP Latch Valve	369	0	0		5%	0.00	9
Pressure Regulator	1195	0	0		5%	0.00	9
LP Pressure Transducer	288	20	5760		5%	6.05	6 for EUCLID
LP FDV	45	3	135		5%	0.14	Flight Hardware
Plenum	671	3	2013		10%	2.21	Flight Hardware
LP Latch Valve	60	3	180		5%	0.19	Flight Hardware
Pipes	2400	1	2400		20%	2.88	needs satellite ICD
Brackets	707	3	2121		20%	2.55	needs satellite ICD
Orifices	66	0	0		20%	0.00	replaced by proportional valve per thruster capable to regulate 7-25µg/sec
AOCS thrusters	100	8	800		5%	0.84	9 (Small Geo)
CG piping	1000	1	1000		20%	1.20	Need S/C ICD
Total Mass						75.61	
Total Dry Mass including contingency						88.85	

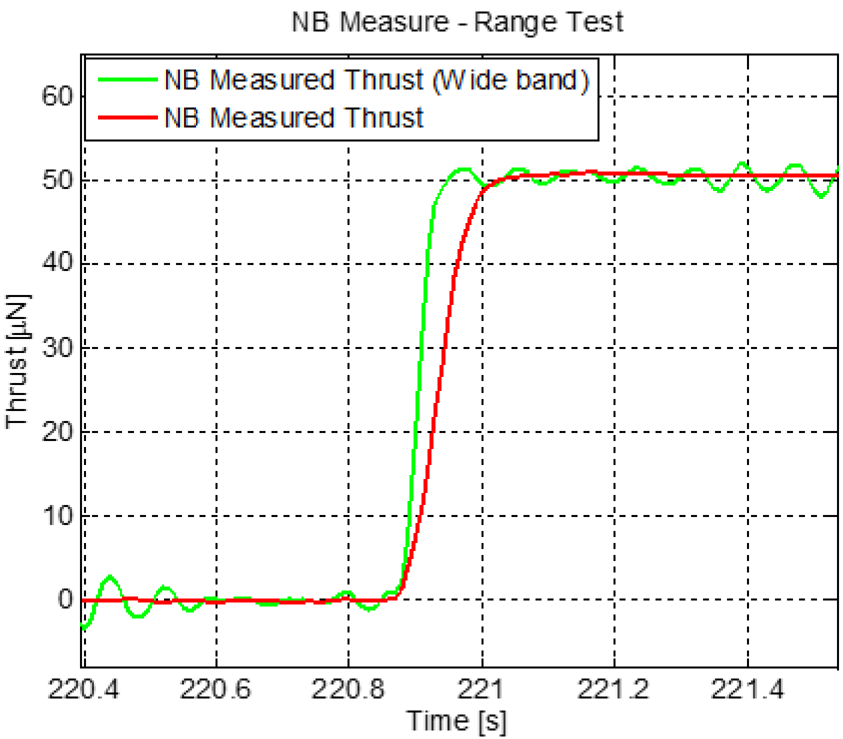
Mission Phases and µProp. Requirements

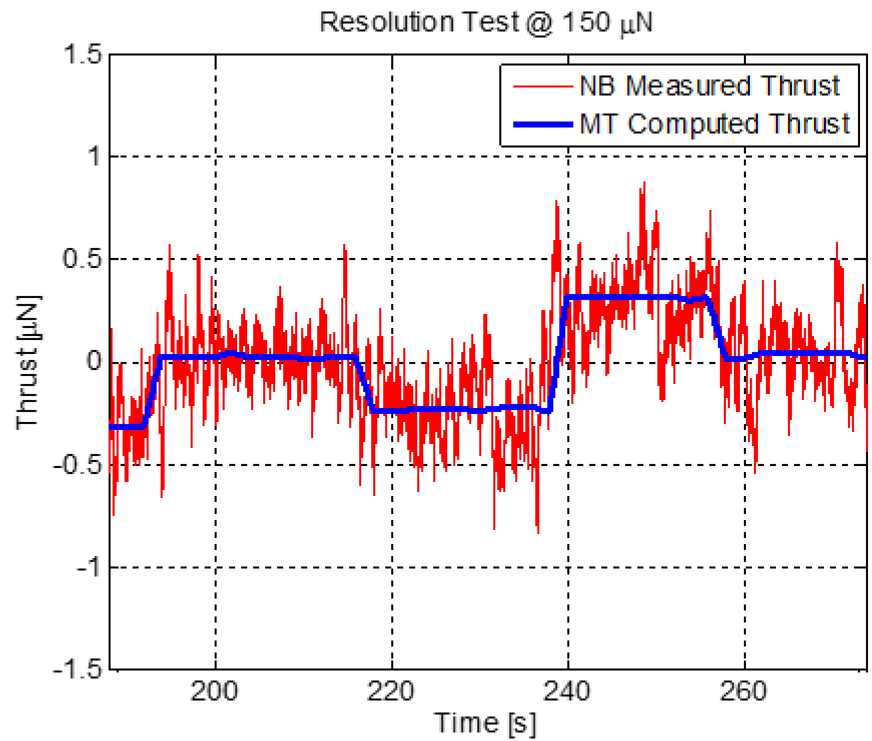
Configuration for science to be updated to 9+9 thrusters

Phase	Activities	Duration [days]	Thrusters in use		
LEOP	Detumbling	2	Xe cold gas		
Cruise EP off	Sun Pointing, periodic pointings towards Earth (once per week)	150	6xminiRIT100uN		
Cruise EP on	EP thrusting, periodic pointings towards Earth (once per week)	300	6xminiRIT100uN	71	4.790
			4xminiRIT1000uN	272	
Acquisition and commisioning	Scan manoeuvres, Science attitude maintenance, SRP compensation, antenna repointing	180	6xminiRIT100uN	71	0.616
Science	Scan manoeuvres, Science attitude maintenance, SRP compensation, antenna repointing	1460	6xminiRIT100uN	71	4.995
Extended Science	Scan manoeuvres, Science attitude maintenance, SRP compensation, antenna repointing	2190	6xminiRIT100uN	71	7.493
Total					**19.607**

Thrust Control

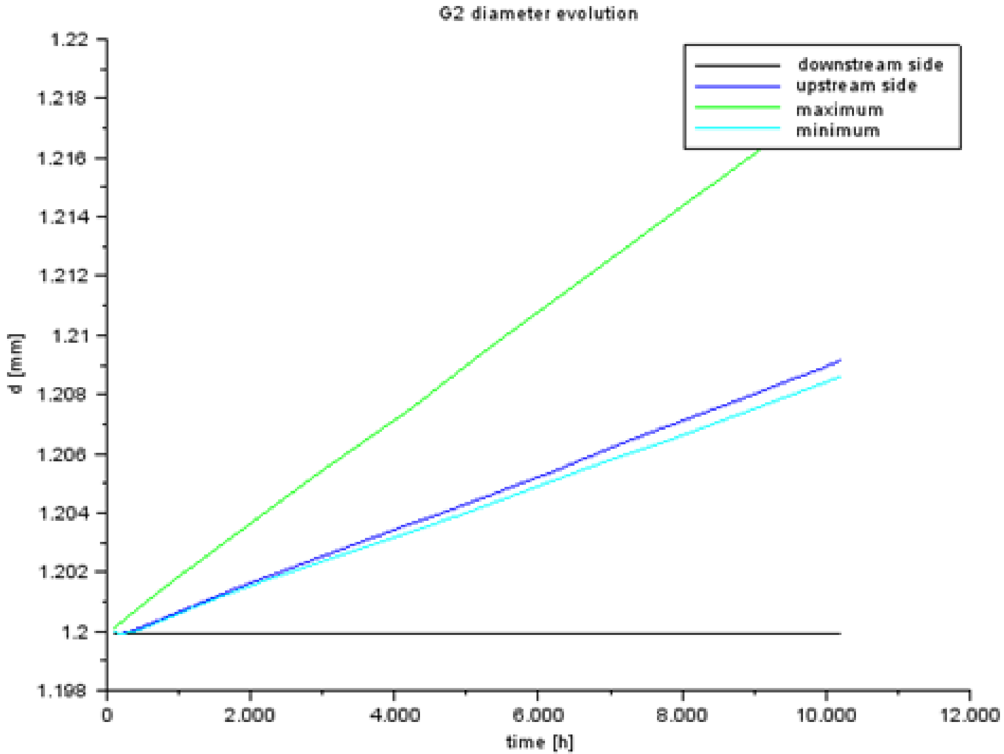
Thrust Control Resolution dependent on Thrust Dynamic Range and the ADC electronics (Here MiniRIT (1000µN) test on Nanobalnce)





Life Time (modelling)

Thruster Life Time model based on RIT-10 thruster (23000hours testing and 6000 hours flight heritage during the ARTEMIS rescue mission



Thrust Noise

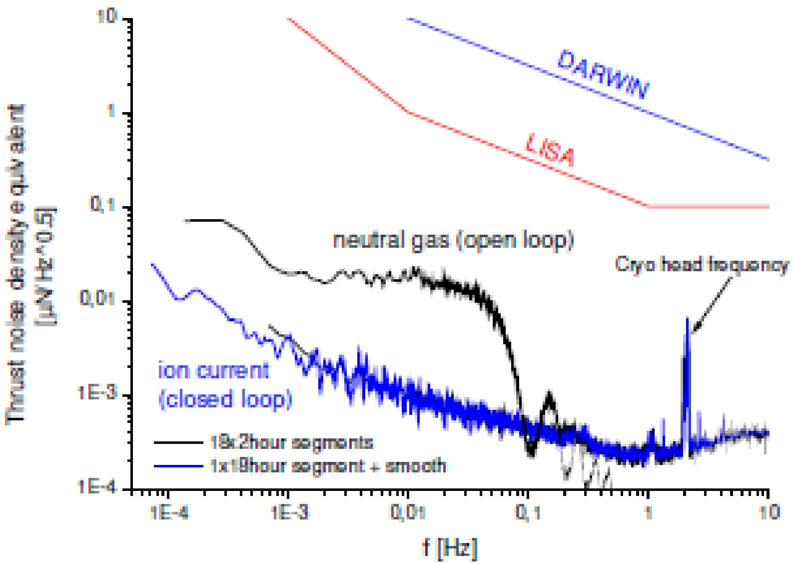
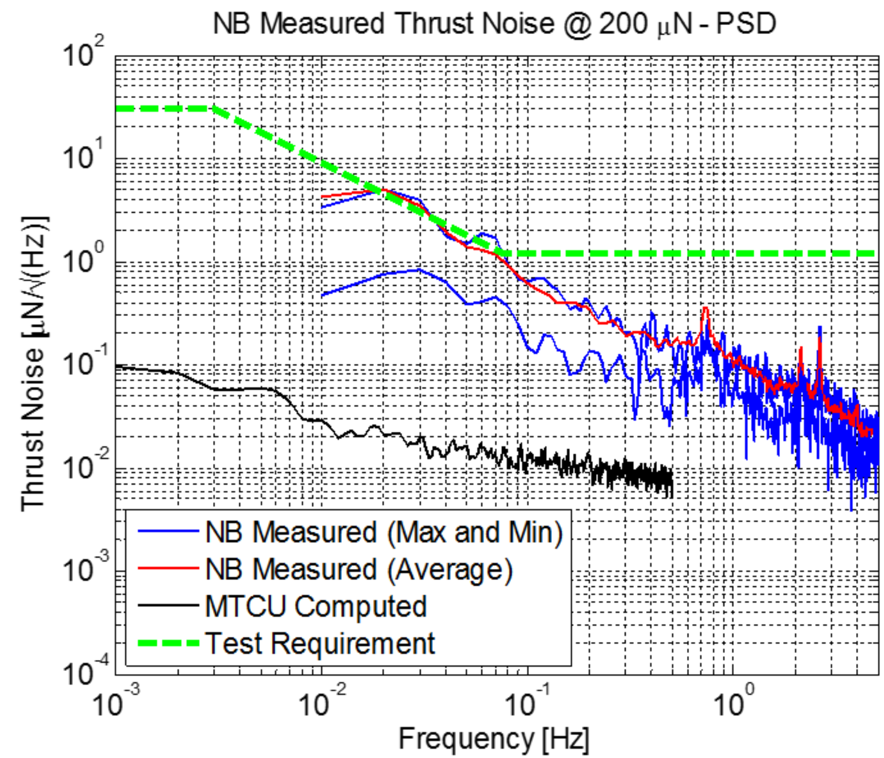


Figure 6 Thrust noise density calculated from current noise density in comparison with DARWIN and LISA thrust noise requirements



S. Weiss et all. ; Thrust noise contribution of µN-RIT with respect to DARWIN and LISA requirements

LISA

TT&C

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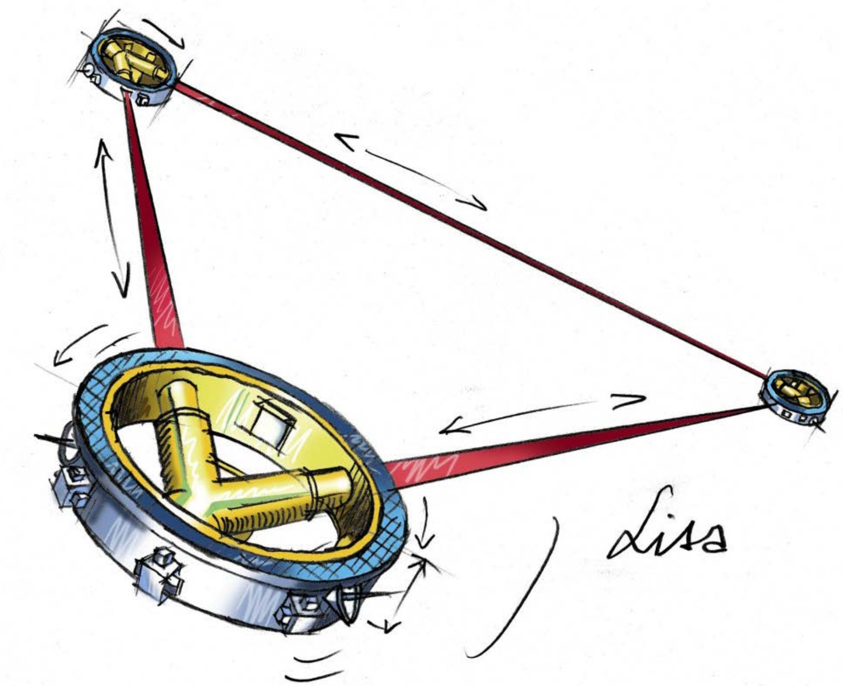
ESTEC, 05-05-2017

Prepared by the CDF* Team



(*) ESTEC Concurrent Design Facility

Main requirements and Design Drivers

- The communication subsystem shall download all Science+H/K data
- Distance from Earth stations – 65 Million of km
- Cover +/- 3.5 deg in elevation, and 360 in azimuth over the year
- No interruptions of science, meaning:
  - Repointing every 14 days (or higher)
  - No thermal variations (power consumption and dissipation the most constant possible)
  - No Center of Mass variations

Assumptions and Trade-offs

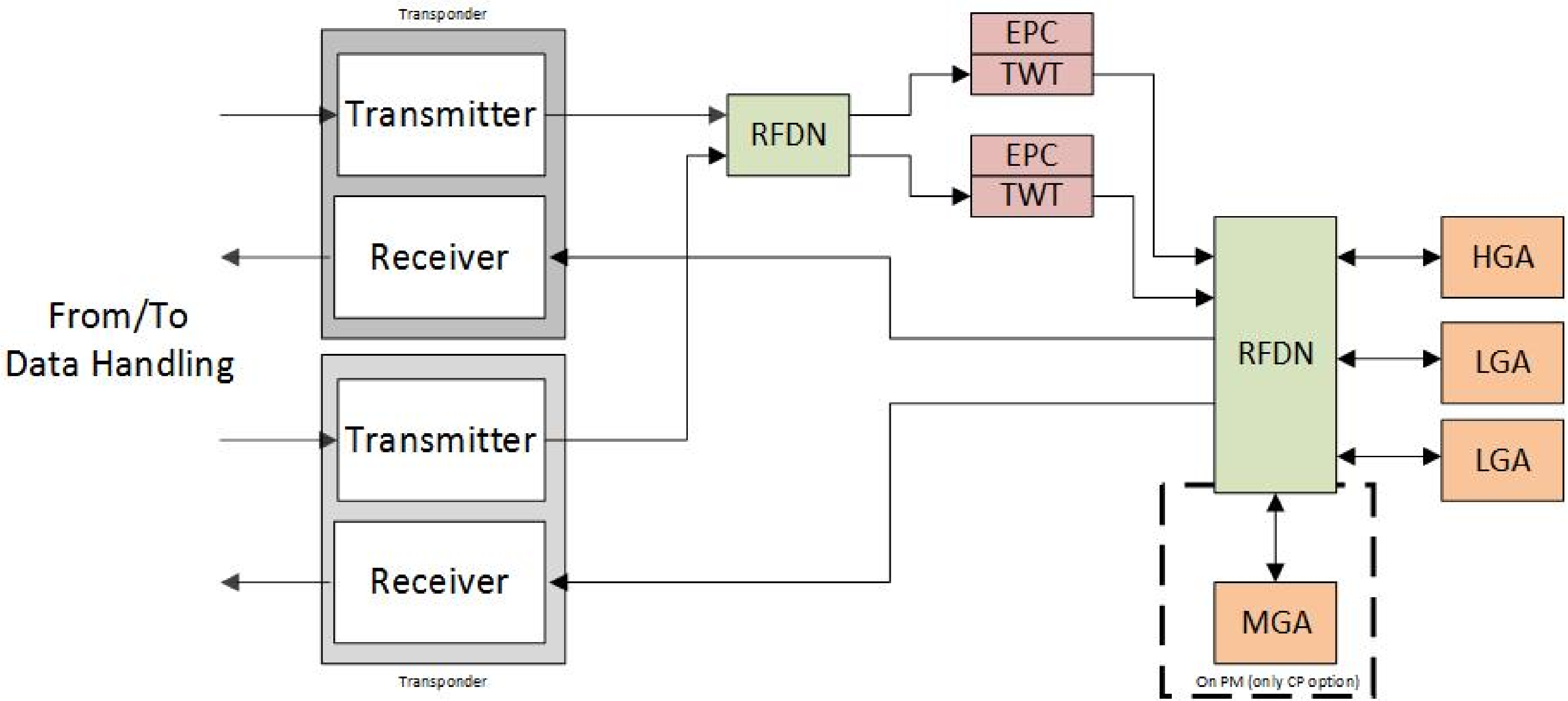
Assumptions

- Constellation data rate generation of 51 kbps
- G/S coverage 10 h/day

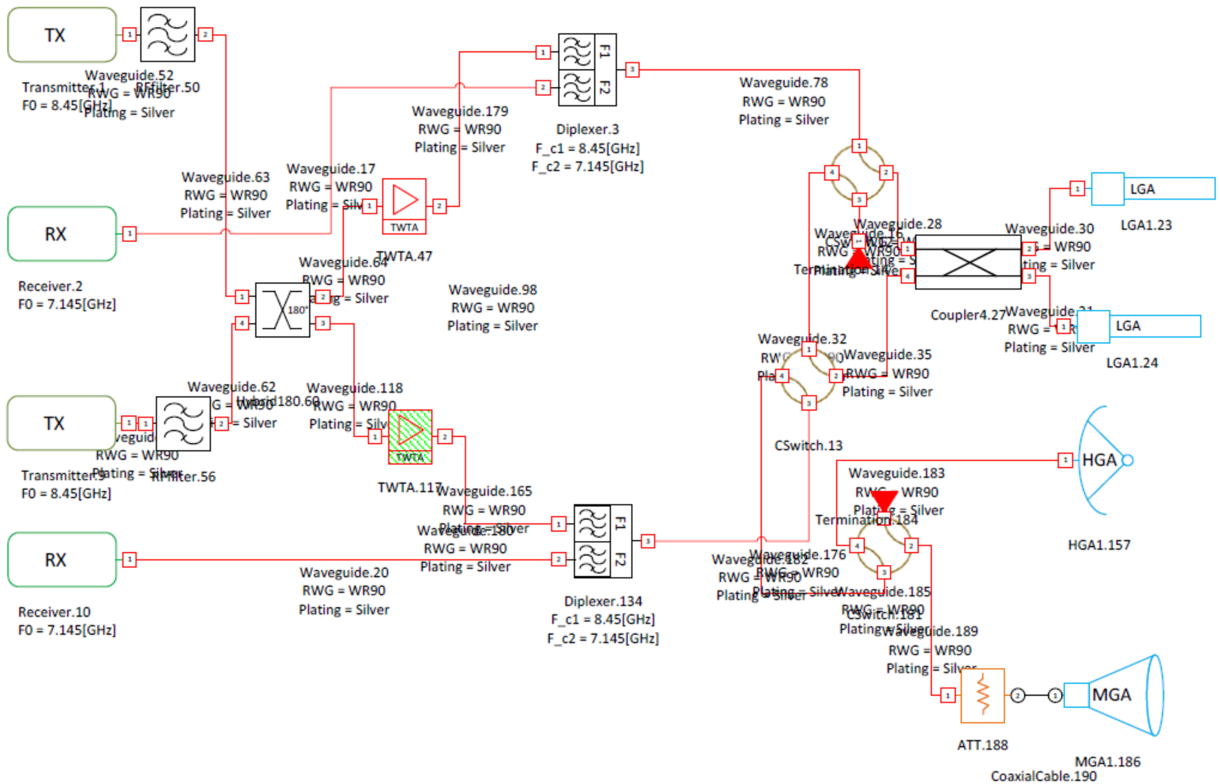
Trade-offs

- Frequency Allocation
- Antenna
  - Mechanical Steerable
  - Phased Array

Baseline Design

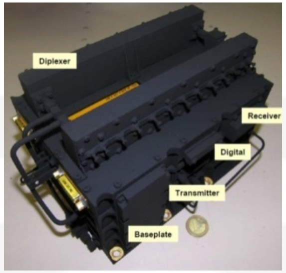


Baseline Design



Baseline Design

- X-Band XPND (X^2PND)
  - 3 kg
  - TX+RX: 25W
    RX: 10W
  - TRL 9
- X-Band TWTA (TWT+EPC)
  - 2.4 kg
  - TX: 281W
  - TRL 9

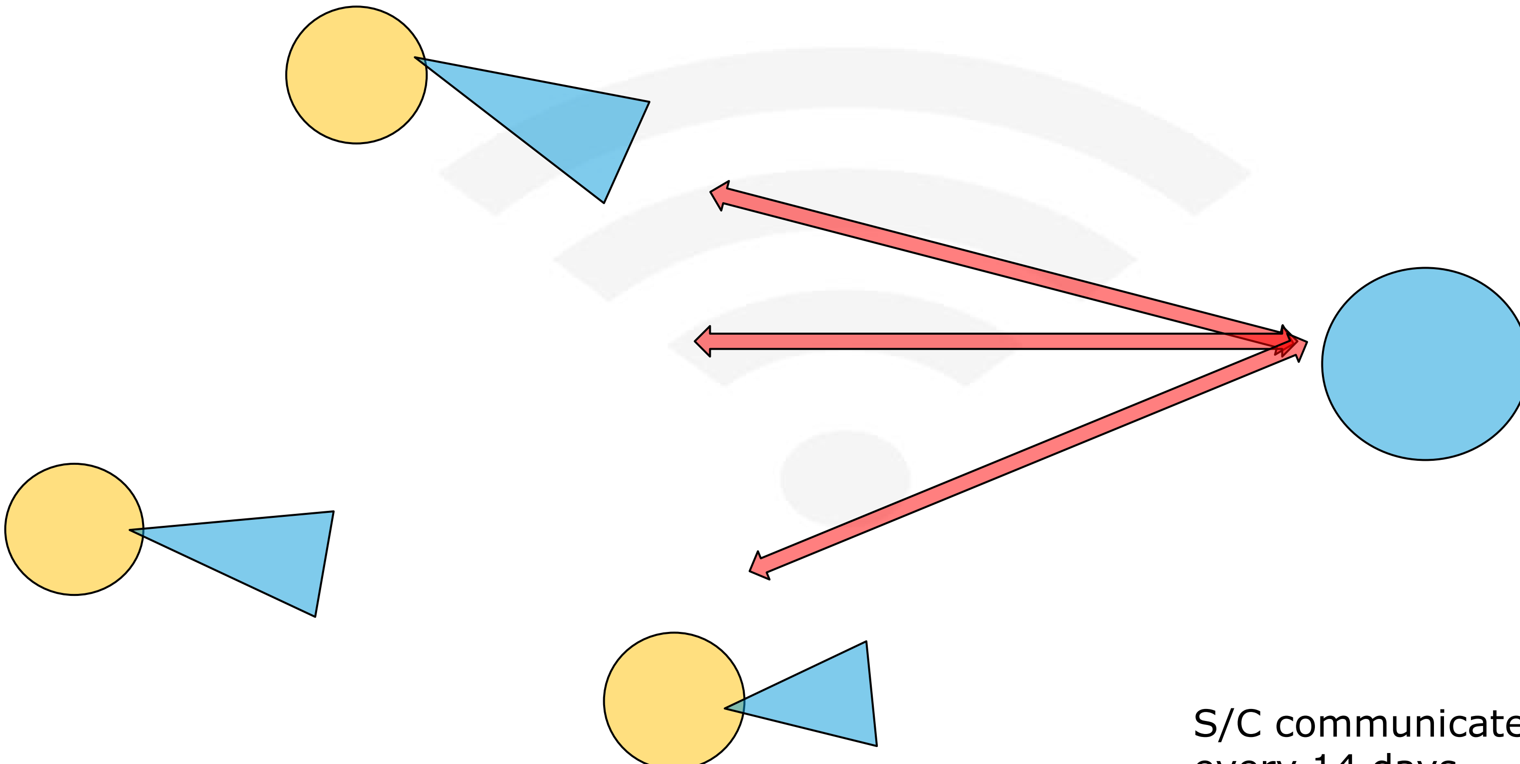


Baseline Design

- X-Band Helix (LGA)
  - 0.4 kg
  - TRL 9

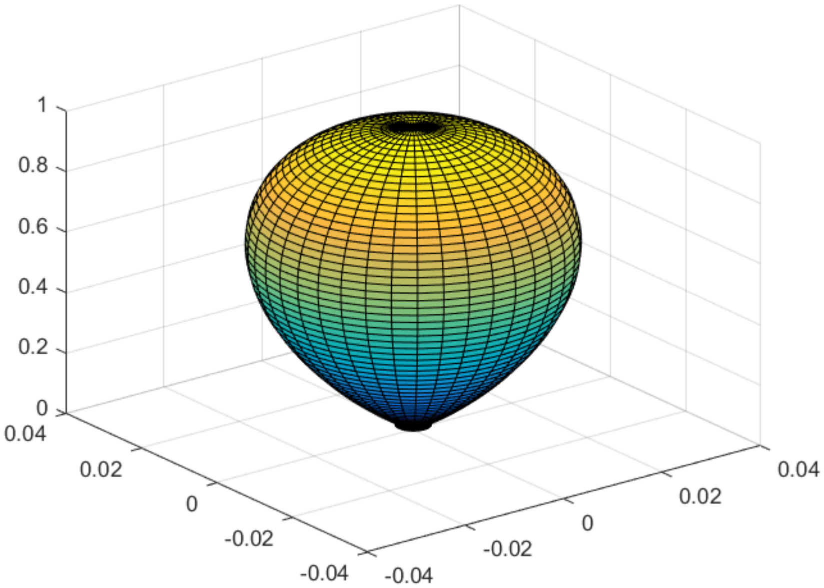
- X-Band Dish
  - Solar Orbiter Heritage
- X-Band Horn (CP option only)
  - 1 kg
  - Syracuse heritage
- X-Band RFDN
  - Solar Orbiter, Sentinels, etc heritage
  - For CP qualification for the PM/SVM RFDN interface

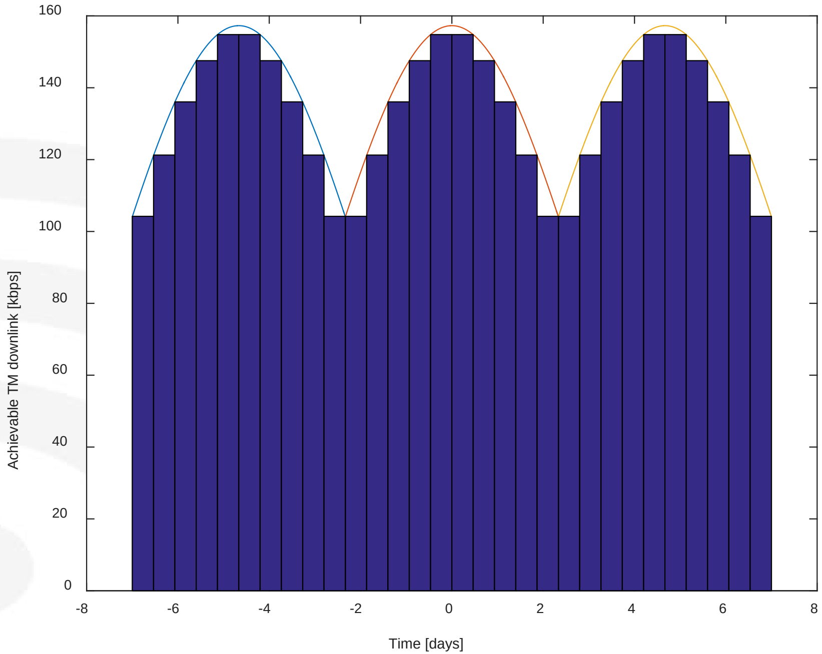
Baseline Design



S/C communicates 4.6 days
every 14 days

Baseline Design – Data rates





Dish
Average rate: 132.7 kbps
Max rate: 154.8 kbps
Min rate: 104.2 kbps
Margin: 4.1 dB (>3 dB)

MGA (CP option)
Max rate: 13 kbps
Margin: 4.0 dB (>3 dB)

LGA
Max rate: 52 bps
Margin: 3.1 dB

Mass & Power Budget, and Power Flux Density

~22.5 kg (25.5 kg for CP option)

316 W consumption
156 W dissipation

Power Flux Density compliant to Regulations down to 200 km of altitude

Options

- Set of all possible options for Mechanical steerable antennas

1 DoF:

Option: Advanced design of Antenna that covers 3.0 deg in elevation
→ difficult to be assessed in CDF

2 DoF

Option: Circular Diameter 0.35 m (Current BASELINE)

- RF 160 W (Consumption 316W, Dissipation 156W)
- TM downlink 52 bps Safe Mode

Option: Circular Diameter 0.50 m

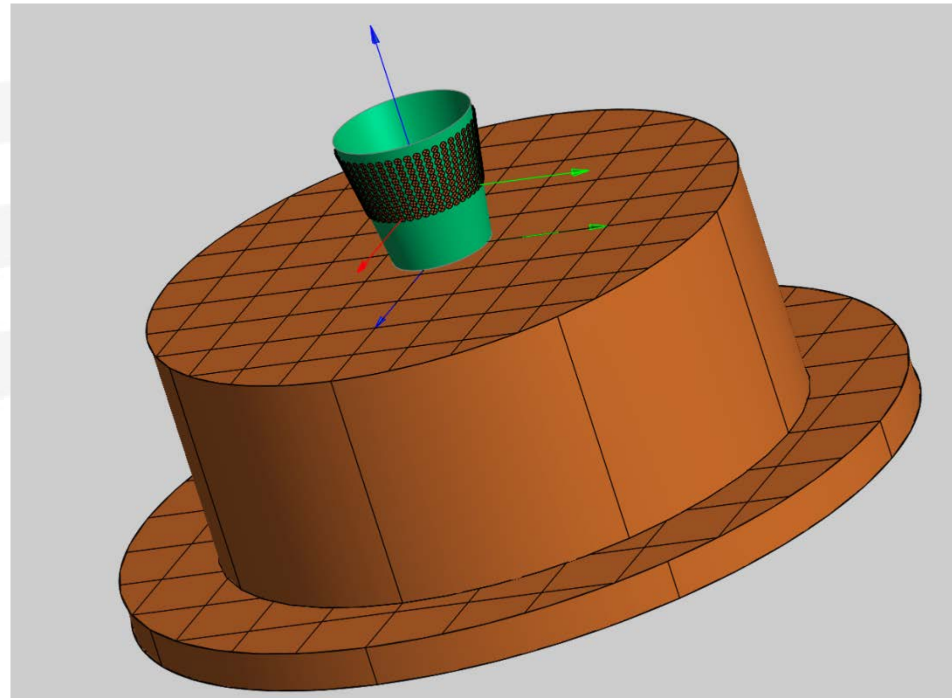
- RF 80 W (Consumption 175W, Dissipation 96W)
- TM downlink 26 bps Safe Mode

Options

Phase Array Antenna

- 600x315 mm
- 27 dBi (sufficient for closing the link at 132 kbps with 160W of RF)
- Beam generated by 15x10 elements

- Beam Forming Network To Be Defined
    - Mass and power consumption are To Be Defined as well

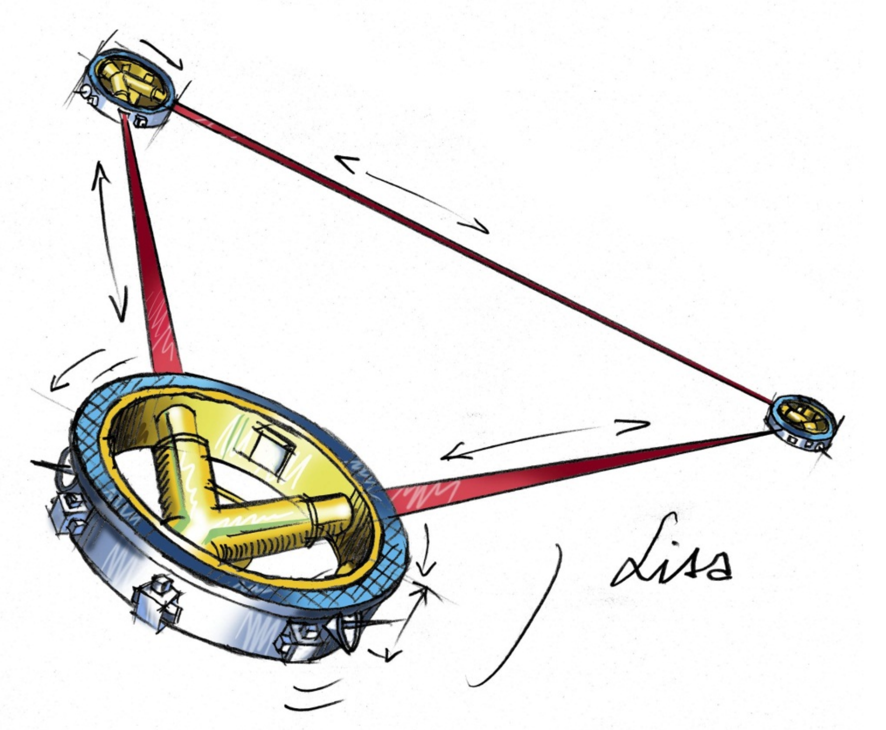


LISA

Data Handling

Internal Final Presentation
ESTEC, 05-05-2017

Prepared by the CDF* Team



(*) ESTEC Concurrent Design Facility

Outline

- Requirements and Design Drivers
- Options
- Baseline Design
- Equipment list
- Budgets

Requirements and design drivers

Telecommands

- DHS shall demodulate, decode, validate, distribute and execute time-tagged or Essential ground Telecommands (TC) allocated to spacecraft (S/C) or payload (P/L) units.
- If S/C is in direct ground contact during Science Mode then DHS shall demodulate, decode, validate and distribute to the other constellation S/C's time-tagged and Essential ground TCs.
- When in Science Mode, DHS shall acquire, validate and distribute to S/C and P/L units time-tagged or Essential TCs received from the S/C in ground contact.

Telemetry

- DHS shall collect S/C and P/L health telemetry (HTM) during all mission phases including transfer and science phases.
- If S/C is in direct ground contact, DHS shall support science TM and HTM data relaying from the other constellation S/C's.

On board Time

- In Science Mode all DHS functions shall be synchronized to a single centralized Ultra Stable Oscillator (USO).
- In Safe Mode or in case of USO failure, DHS shall use an internal oscillator.

Autonomy

- DHS shall support autonomous science operations and communication with the other constellation S/Cs.
- DHS shall support autonomous FDIR functions and transition to Safe Mode.

Data processing

- In Science Mode DHS shall be in charge to run DFACS algorithm to control u-propulsion.
- DHS shall perform data filtering, downsampling and then compression if needed to meet downlink bandwidth constraints.

Data storage

- DHS shall acquire and store on-board all DFACS, science (268 Kbit/s) and HTM data to a max of 256 GByte EOL.

DHS performance requirements

P/F tasks

H/K
mem. management
TC manager
TM manager
Ext. data control
PUS services
Monitoring
MTL
Event actions

~ 7 MIPS
from LISA-PF

P/L tasks

DFACS	~ 8MIPS
Filtering/downsampling	~ 18 MIPS
Data compression	~ 2,5 MIPS

→ ~ 70 MIPS
assumption for LISA

DHS options: OBC and MM

RUAG OBC

- Heritage from Sentinel 3, Small GEO, Juice.
- All-in-one OBC+RTU
- > 65 MIPS (> 3x LISA-PF OBC) suitable for P/F + P/L tasks
- 14 kg, 38 W (Juice CDMU)
- 318L x 260W x 277H mm
- TRL 8 for OBC part: full reuse for LISA except for centralized oscillator.
- TRL 7-8 for RTU part: possible modifications for LISA

Airbus mass memory

- Heritage from Sentinel 2 and 5, SEOSAT, MetOp-SG etc
- Higher density flash memory device needed for LISA storage requirement (128 Gbit devices)
- 128 Gbit flash device → ~370 Gbyte per board
- Integrated Transfer Frame Generator for data formatting and transmission to ground.
- 8 Kg, 20 W (MetOp-SG MM)
- 340L x 130W x 234H
- TRL 6

DHS options: Centralized vs distributed I/O system

Two different approaches for distribution of I/O interfaces over the S/C:

- centralized approach: single box as star point for most of data harness
- distributed approach: OBC box comprises only core functionalities and MM. A data bus connects a number of uRTUs distributed over S/C

	Distributed	Centralized
Budgets	- reduced OBC mass - reduced harness mass - possible overall minor mass saving - potentially higher power consumption - reduced OBC size but possible overall higher footprint	- high harness mass - centralized mass in OBC - fewer DC/DC converter → less power consumption - optimized volume
Design	- high scalability - limited number of I/O's in OBC	- fixed scalability - high number of I/O's in OBC
Heritage	- reuse of already developed and qualified OBC	- impact on qualification due to specific LISA req.
Performance	- I/O's are sourced from uRTUs connected to OBC via 1553 or CAN bus - Better real-time capability if data processing capability are supported by each uRTU.	- I/O boards are integrated with OBC and connected with an internal link - Real-time capability depends on internal bus

⇩

No clear advantage of one of the two approaches

DHS baseline

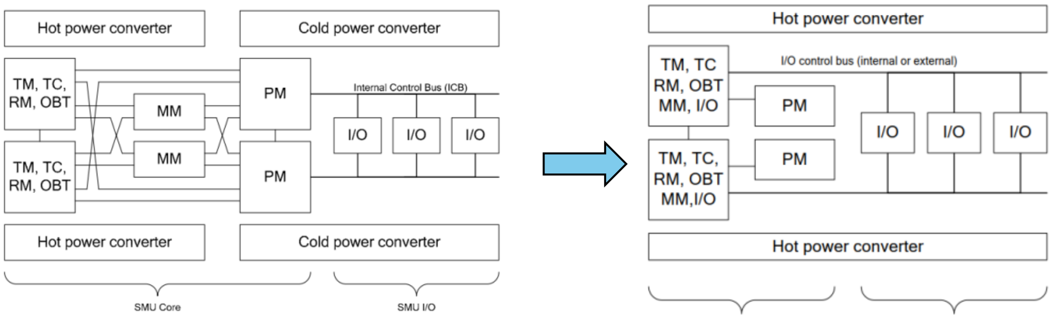
RUAG Next Generation Spacecraft Management Unit (SMU)

	JUICE CDMU
Mass	14 Kg
Volume	22.9 dm^3
Dimensions	318 mm x 260 mm x 277 mm (W x D x H)
Power	38 W

	CDMU-NG
Mass	8.25 Kg
Volume	11.66 dm^3
Dimensions	174 mm x 242 mm x 277 mm (W x D x H)
Power	33 W

- Evolution of the existing SMU used for example in Juice mission. Goal of the new design is twofold:
  - to support new functionalities foreseen in future missions: new TM/TC standards for data relaying, higher CPU performance, file based MM, sensor bus I/Fs, increased Essential TM etc
  - to reduce weight, power and size by merging functionalities in few complex ASICs.
- Possibility to integrate in the same OBC box both RTU and Mass Memory functionalities (baseline for the proposed LISA DHS model)
- Technologies developed in the frame of three ESA studies:
  - SBCC (Single Board Computer Core)
  - AFIO (Advanced Flexible I/O)
  - MMOBC (Mass Memory for OBC)

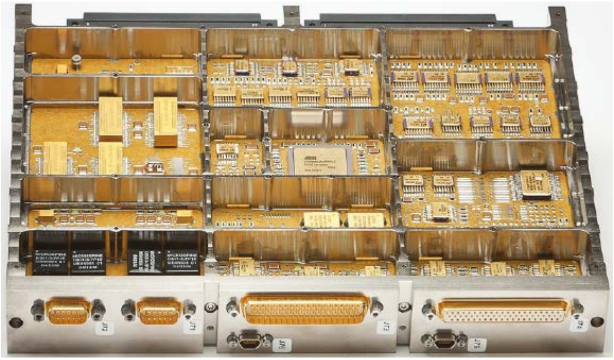
DHS redundancy concept



From current to a new S/C Management Unit architecture concept

- Redundancy and FDIR concept is changed due to low boards number and high integration
- Reduced number of power converters
- Fewer cross-strappings and less circuitry
- Application processor is separated to handle future performance increase
- All-in-one OBC+RTU+MM with 5 independent redundant modules:
  - 2 Processor Modules (hot, cold)
  - 2 TC / TM / Reconfiguration / Safeguard Memory/ OBT modules (hot, hot except nominal TM encoder)
  - 2 AOCS I/F modules (hot, cold)
  - 2 Standard I/O I/F modules (hot, cold)
  - 2 Mass Memory Units (hot, hot)
  - 2 Power Converter (hot, hot)

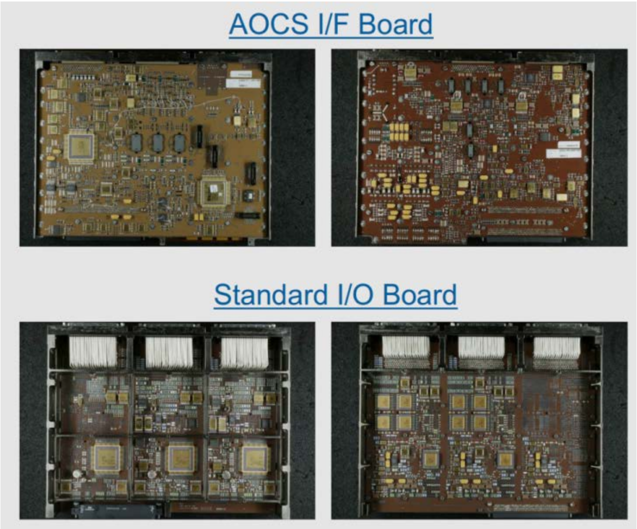
Next Generation SMU: Computer Core

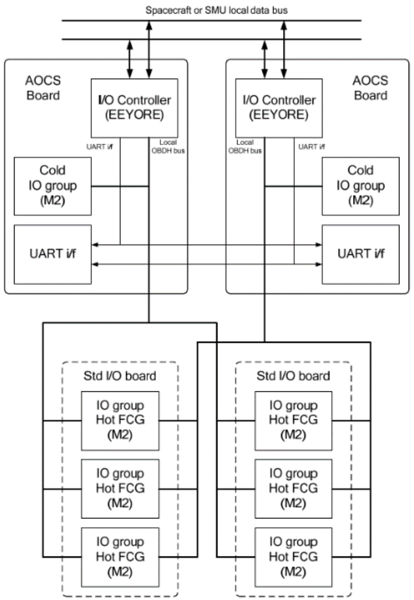


SBCC (Single Board Computer Core)

- All S/C management specific functions are integrated in a single ASIC: TC, TM, Reconfiguration, Safeguard Memory, OBT
- All I/Os are managed by a specific I/O processor
- Application processor is separated to handle future performance increase.
- Current SBCC under development will use NGMP processor: 4-core LEON4, up to ~800 MIPS (first assumption for LISA is ~70 MIPS)
- Increased reliability of the on-board SW when running critical tasks (P/F management) and lower critical tasks (P/L management) in the same computer
- Main I/O interfaces
  - 2 CAN
  - 2 MIL-STD-1553
  - 13 SpaceWire links
- EQM of the SBCC in 2018

Next Generation SMU: I/O system

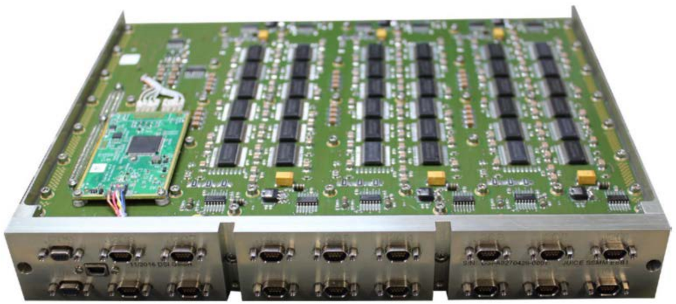


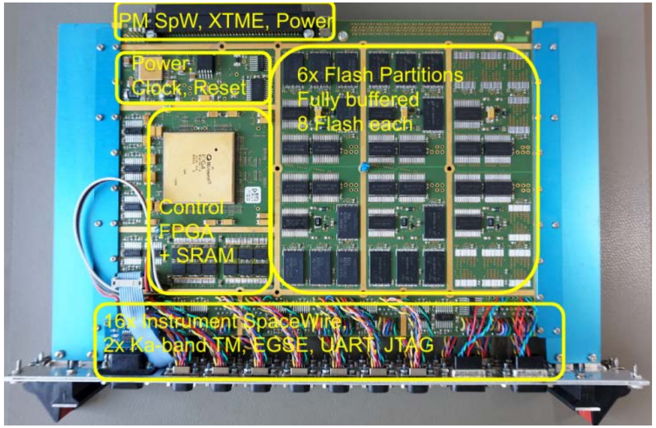


AFIO (Advanced Flexible I/O system)

- The RTU part is based on two board types:
  - Standard I/O board providing: thermistor acquisitions, analog measurements, relay acquisitions etc. Number of boards can be increased according to mission I/O's requirements.
  - AOCS I/O board with DC/DC converter: propulsion, magnetorquers, magnetometers etc
- Autonomous acquisition and commanding based on local instructions list.
- Reduced power (mainly idle) and mass of the I/O system, by 60% respectively 50% compared to previous I/O board designs.
- EQM already available

Next Generation SMU: Mass Memory

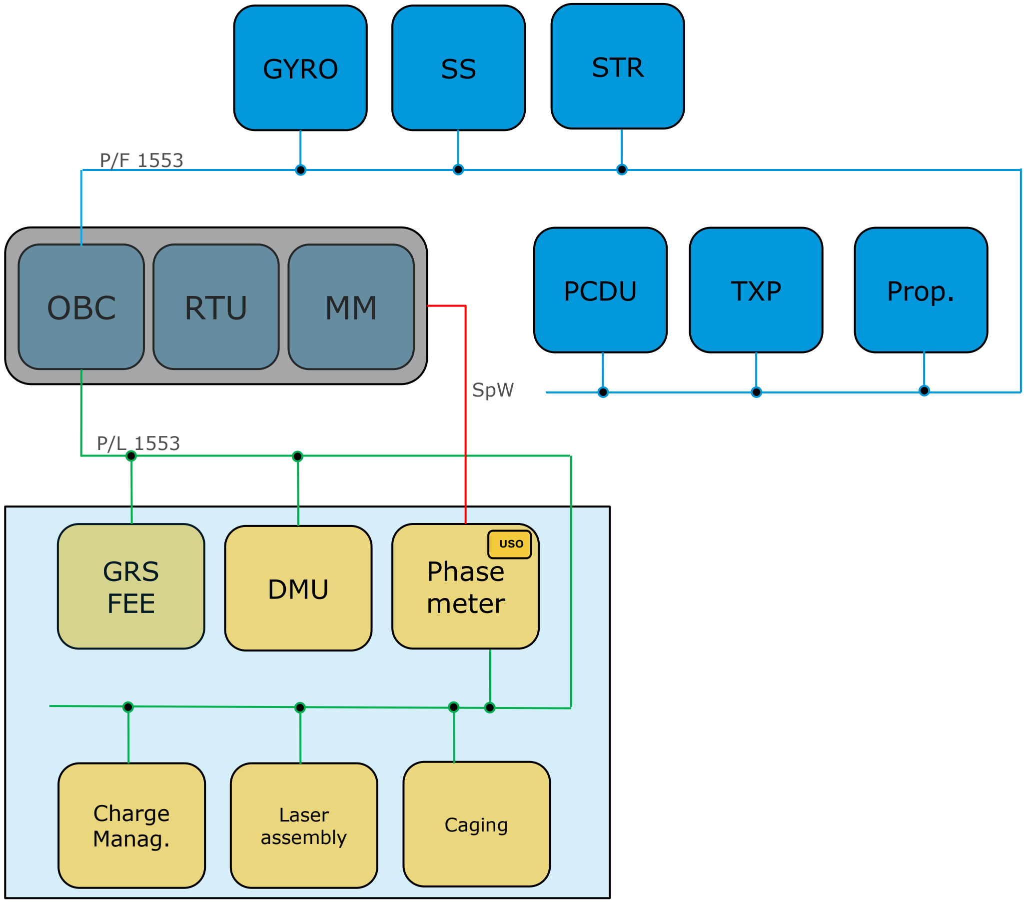




MMOBC (Mass Memory for OBC)

- Trend in MM design is towards non-volatile MM based on Flash memory devices.
- In space MM, non volatile Flash technologies will soon replace DRAM memories.
- Each MMOBC board can store 750 GByte using 128 GBit flash memory devices
- Implementation of CFDP standard as file transfer protocol
- Files store management
- Autonomous downlink of files from flash storage to x-band.
- Configurable as self-standing MM with a dedicated processor or with OBC providing the processing function.
- 16 SpaceWire interfaces to instruments + 6 SpW for internal x-coupling and OBC interface
- Breadboard model available

DHS Architecture



- Backbone for platform and P/L data transfer is Mil-1553 bus or CAN bus
- Separated bus for platform and P/L
- OBC acts as platform and P/L Bus Controller
- Data from Phasemeter via SpW link to MM.

DHS Budgets

Mass

Boards	N. boards	Mass (Kg) board (1)
SBCC	2	0.9
OBCMM	2	0.96
AOCS	2	1.43
IO	2	1.43
DC/DC SBCC	2	1.12
Motherboard	1	0.88
		13.728

(1) Mass per board excluding housing mechanics
- Final value includes 10% for housing
- SBCC board mass from Herschel Processor Module
- DC/DC mass from Herschel
- AOCS and I/O board mass from AFIO study
- Motherboard mass from Herschel

Power

Boards	N. boards	Power operational(W) (1)	Power stand-by (2)
SBCC	2	6.84	6.47
OBCMM	2	6.66	3.53
AOCS	2	5.55	4
IO	2	5.55	4
DC/DC SBCC	2	11.48	8.50
		36.08	**26.50**

Power in operational mode derived from COLE ASIC PM board + GR740 processor consumption

(1) - GR740 processor with 4 core 50% load and 1 SpW active
- MM all SpW active and 1 mem. Partition on
- AOCS/IO autonomous acquisitions are running and data is retrieved via 1553, but no power interfaces are active.

(2) - GR740 with 4 core 0% load and no SpW
- MM stand-by: no links active
- AOCS/IO as in operational mode but u-stepping SADM is off.

Dimensions

Boards	N. boards	Width (mm) (1)
SBCC	2	36
OBCMM	2	36
AOCS	2	36
IO	2	36
DC/DC SBCC	2	36
		396

396W x 277H x 242D

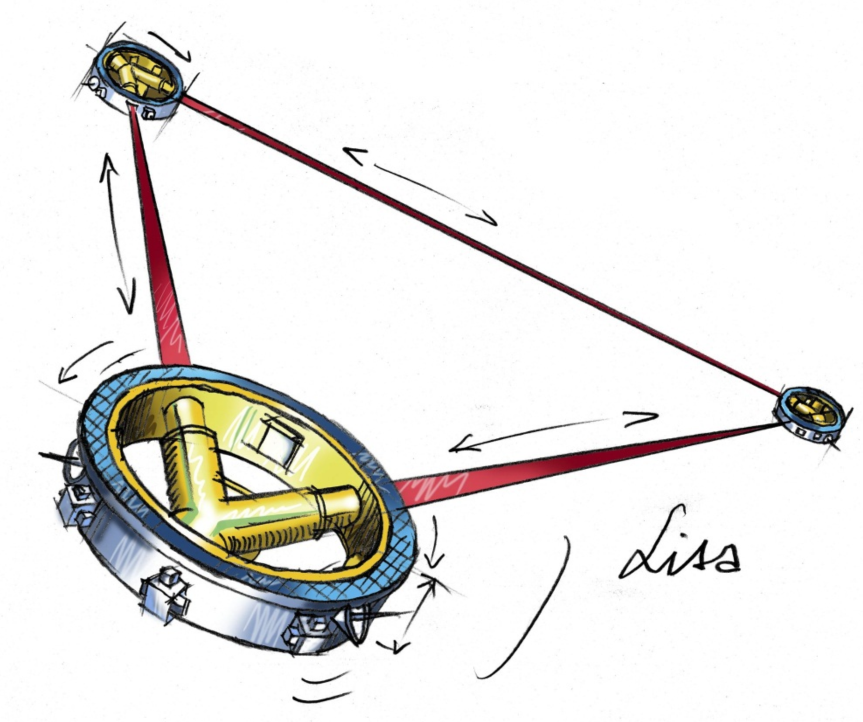
(1) Width includes 36mm including feet
H and D are from CDMU-NG (SBCC activity)

LISA

Power

Internal Final Presentation
ESTEC, 05-05-2017

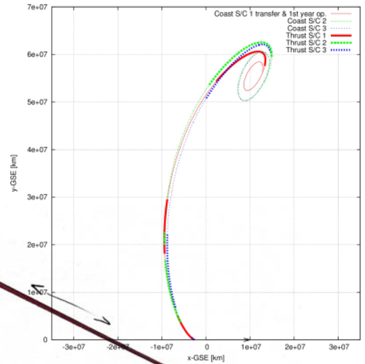
Prepared by the CDF* Team

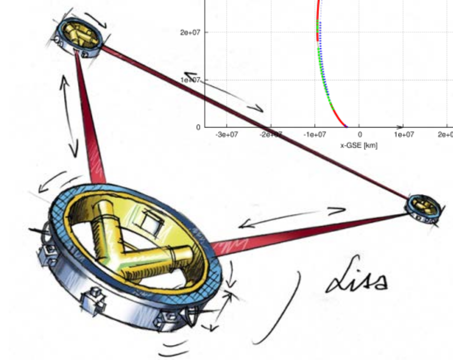


(*) ESTEC Concurrent Design Facility

- Requirements and Design Drivers
- Assumptions and Trade-Offs
- Baseline Design
- Equipment list

Requirements and Design Drivers





EPS shall provide the required power during all mission phases

- **Launch**: EPS must provide, without sunlight to panels, the load requirements from disconnection of launcher umbilical to completion of detumbling. This total time will be < 2 hours [TBC] (detumbling approx. 15 min).

- **Transfer**:
  - CP option: Spacecraft shall be turned to point the main thruster for burns of 2 hours duration. During this time, worst case of zero sun on solar panels must be assumed.
  - EP and EP_plus options: Large power requirement for main electrical thruster. Thruster must be pointed appropriately at all times. Thruster axis is fixed at 90° to solar panel normal. Solar aspect angle can therefore not be optimised, and may be up to 40° in worst case. No eclipses.

- **Science operations**: Plane of constellation shall be 60° to ecliptic plane. When combined with the favoured practical SC configuration of "solar panel on top", this means a solar aspect angle of 30°. No eclipses.

Requirements and Design Drivers

Payload requires high thermal stability

- So provision of a perfectly stable voltage electrical bus (during sci ops) is preferred
  - To avoid variations in thermal outputs from e.g. DC-DC converters in payload units.

Payload requires high mechanical stability

- So thermoelastic loads from the solar array shall be isolated from the SC as far as possible.
  - As was the case in LISA Pathfinder

Mission lifetime is long

- Considering transfer time and extended science mission lifetime goal leads us to 12.5 years total lifetime
  - Solar array performance degradation must be correctly accounted for.

Requirements and Design Drivers: Load budgets:

- 10% power margin is applied to the Hall effect thruster load (PPU)
- 30% margin is applied to all other equipment

Requirements and Design Drivers: Load requirements

SIZING CASES:

- Solar Array.
  - CP option
    - Largest power requirement is 1581 W in SciM
    - SciM means: sun aspect angle is 30°, age of solar arrays = 12.5 years
  - EP option
    - Largest power requirement is 2406 W in TFM
    - Worst case sun aspect angle is 40°, age of solar arrays = 1.5 years
  - EP_plus option.
    - Largest power requirement is 2750 W in TFM
    - Worst case sun aspect angle is 40°, age of solar arrays = 1.5 years

Requirements and Design Drivers: Load requirements

SIZING CASES:

- Battery
  - CP option.
    - Sized to support 2 hours of TFM mode @ 1149 W
    - (It is assumed that in worst case, the SC must point the engine such that there is zero sun on the array.)
    - 2 hours does **not** allow for an entire ΔV manoeuvre, but it is of negligible cost to split the burns into 2 hour episodes.

  - EP and EP_plus option.
    - Worst case is launch mode @ 481 W. Must be supported from disconnection of launcher umbilical to completion of detumbling.

Assumptions and Trade-Offs

- Currently available SOTA solar cells (AZUR 3G30C) are used for the solar array sizing. This is conservative, because:
  - launch date of 2034 means that future generation cells of higher efficiency are very likely to be used.
  - Next generation ~33% efficiency cells would mean a ~10% reduction in solar cell requirement / PVA area.
    - Saving ~ 1.3 m^2 and ~7 kg from EP solar array
    - Saving ~ 1.5 m^2 and ~8 kg from EP_plus solar array
  - This will NOT translate to a 10% reduction in solar array mass in the cases where the panel area is fixed for sunshield purposes (CP option).
  - Beware of over-optimism: The major improvement in solar array W/kg foreseen in the coming years is due mainly to lighter panel technologies in deployable wings.
    - This is probably NOT applicable to the LISA thermally and mechanically isolated body-mounted panel.

- Currently available SOTA space qualified Li-ion battery technology is assumed for battery sizing (namely SAFT VES16).
  - Other space qualified batteries are also applicable.
  - Launch date of 2034 means that future generation Li-ion secondary cells of higher energy density are likely to be used.
  - Energy density at cell level could improve from ~150 to ~250 Wh/kg
  - This could lower the mass and volume of the battery(s) by up to 40%.
    - Saving e.g. ~ 11 kg

Assumptions and Trade-Offs

Power system architecture:

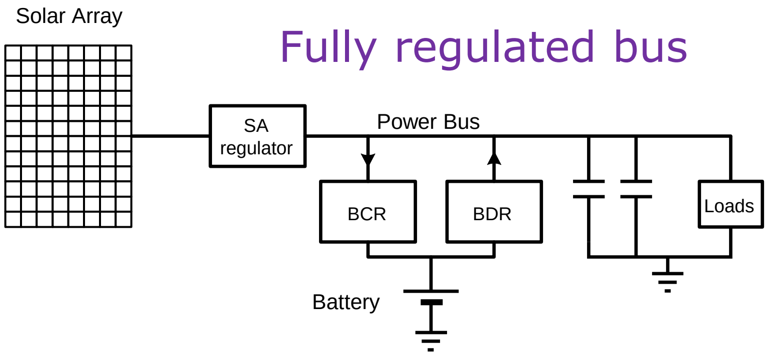
- Solar regulation – shunt or MPPT?:
  - In EP and EP_plus cases, the **solar array sizing case is in transfer**, close to beginning of life.
    - For shunt regulation/direct energy transfer/S3R, the solar cell string length must be sized to achieve the minimum required **voltage** at end of life, when degradation is highest and voltage is lowest. So SA would be "oversized" at 1.5 years (it would be forced to operate below MPP).
    - This strongly leads us to an **MPPT** solution, so that the array size can be optimised.
  - For the CP option, either S3R or MPPT may be suitable. The higher efficiency of the S3R would likely give a small saving in array size, but for the CP option the panel is sized for sunshield, so overall impact negligible at SC level.
  - In all cases, an MPPT was implemented in the EPS model used for the sizing calculations

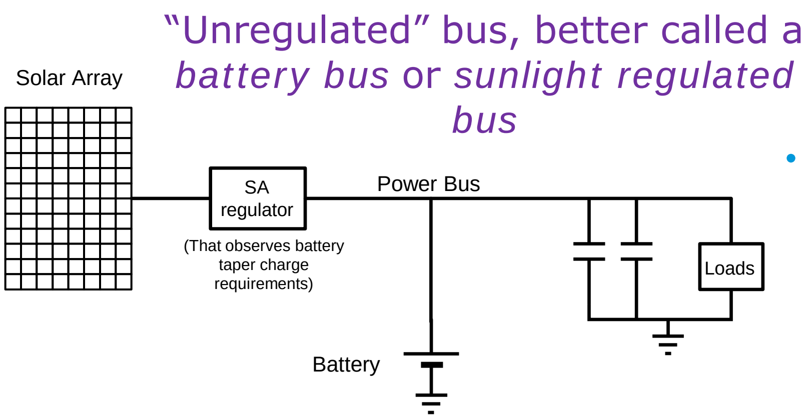
Assumptions and Trade-Offs

Power system architecture:

- Bus voltage – 28, 50, even higher?
  - In EP and EP_plus cases, the PPUs for the Hall thrusters require a 50 V feed.
  - In **all** options, the total SC power requirement is above 1.57 kW
    - (ECSS says 50V for 1.57 kW < P < 5 kW)
  - So **50 V** is appropriate for all options.

Assumptions and Trade-Offs





Power system architecture:

Main bus type – regulated? battery?

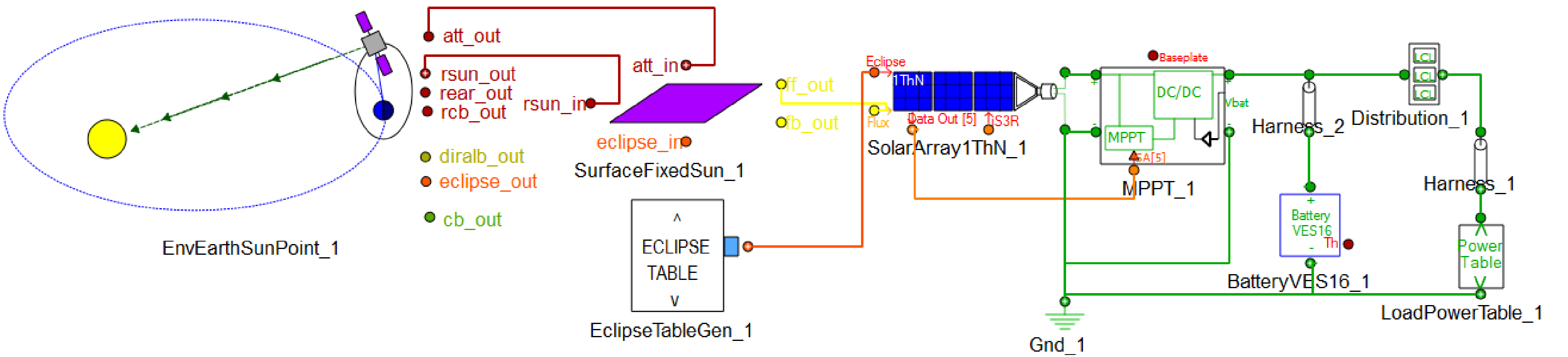
- Provision of a perfectly stable voltage electrical bus (during SciM) is preferred
  - To avoid variations in thermal outputs from e.g. DC-DC converters in payload units.
- This requirement is met equally well by a regulated or "unregulated" bus
  - Because in the case of LISA, there are no eclipses (and therefore no periods of battery discharge) when on-station.
  - (A battery bus is also called a "sunlight regulated bus", to reflect the fact that the SA regulator limits the bus voltage to V_{EOC} when battery is charged and generation meets load demand.)
- Battery bus solution would optimise battery sizing by avoiding DC/DC conversion losses during discharge.
  - And it would be cheaper and lighter.
  - So battery bus is baselined.

Baseline design – solar array panel

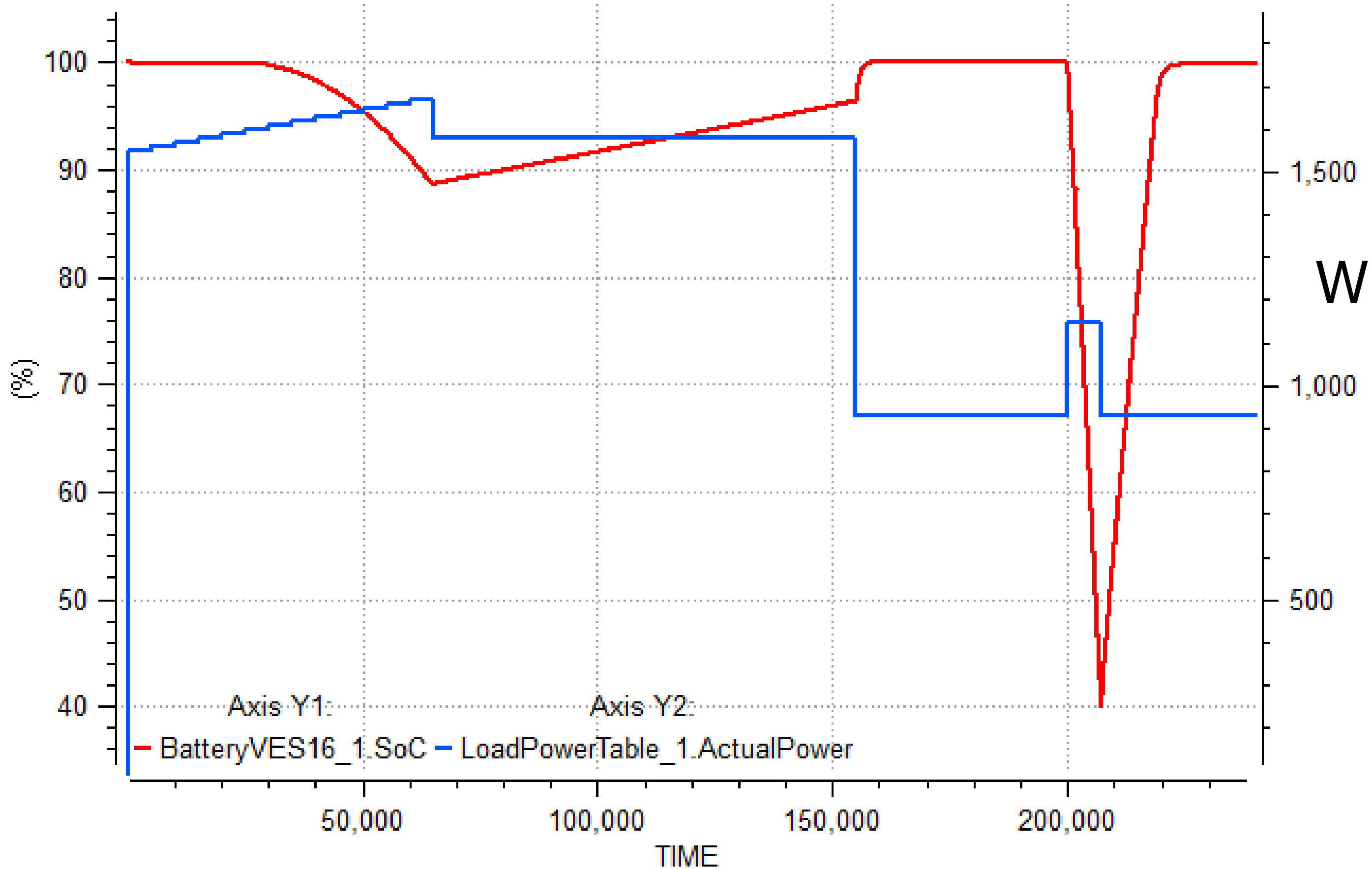
Solar panel mechanical aspects:

- The LISA pathfinder solution is assumed
  - Special blade mountings to minimise the transmission of thermoelastic loads from the solar array to the SC structure.
- For mass estimation, I take the LPF panel area-specific mass for all components except the PVA and the MLI:
  - Mass of substrate + inserts + blades = **3.68 kg / m²**.
  - PVA mass is from the *PEPS* EPS model.
  - MLI is accounted for by THERMAL discipline

Baseline design – sizing with *PEPS* model

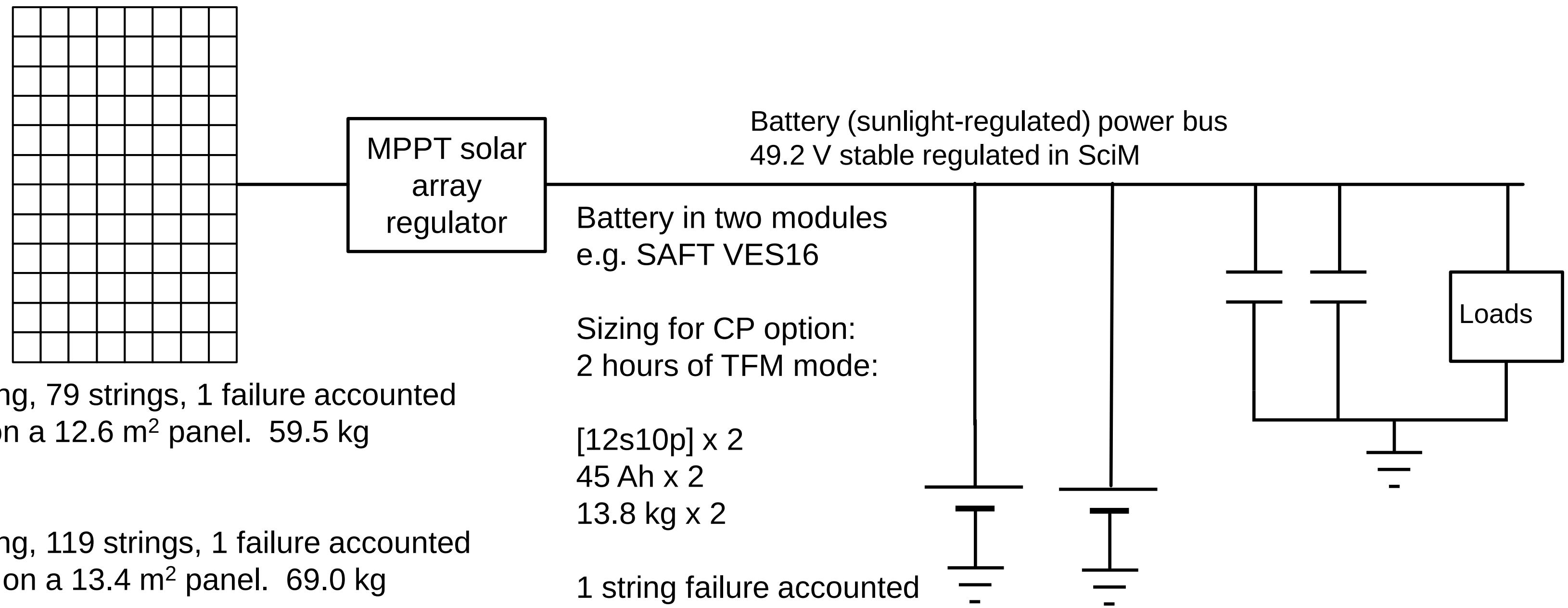


Baseline design – sizing with PEPS model



Baseline design & sizing

Solar Array
AZUR 3G30C   8 x 4cm cells



CP:
30 cells per string, 79 strings, 1 failure accounted
8.9 m^2 of PVA on a 12.6 m^2 panel. 59.5 kg

EP:
30 cells per string, 119 strings, 1 failure accounted
13.4 m^2 of PVA on a 13.4 m^2 panel. 69.0 kg

EP_Plus:
30 cells per string ,135 strings, 1 failure accounted
15.2 m^2 of PVA on a 15.2 m^2 panel. 78.3 kg

Battery in two modules
e.g. SAFT VES16

Sizing for CP option:
2 hours of TFM mode:

[12s10p] x 2
45 Ah x 2
13.8 kg x 2

1 string failure accounted

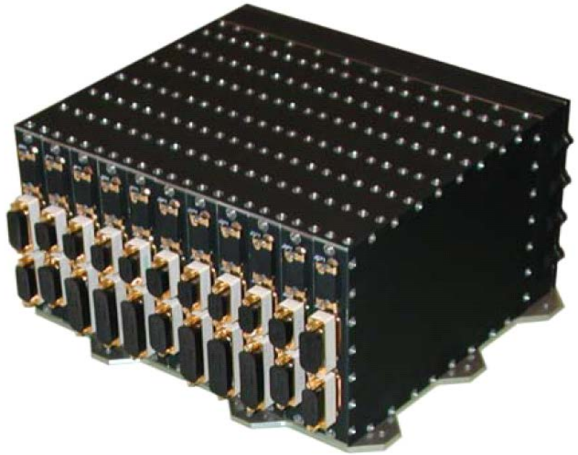
The same battery is baselined for EP and EP_plus. It provides the Launch Mode power requirement for 5 hours (to 40% SoC)

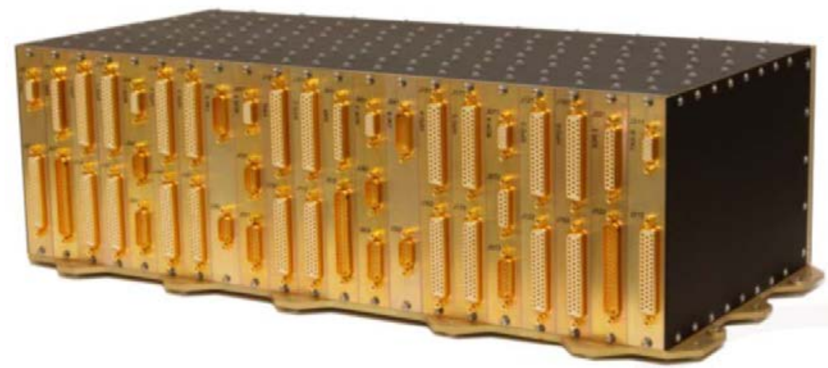
Baseline design & sizing: PCDU

PCDU sizing (for mass and volume) is a uniquely inaccurate science

- (Seems to depend more on the manufacturer than the functionality.)
- In this case, I make a mass and volume estimation using the TERMA modular power system products of Galileo IOV heritage.

- **CP option: 15.5 kg, 18.5 litres**
- **EP and EP_plus: 19.6 kg, 23.1 litres**





Equipment List

CP

	mass (kg)	mass margin (%)	mass incl. margin (kg)
Bat_SVM_1 (Battery_SVM #1)	13.80	4.49	14.42
Bat_SVM_2 (Battery_SVM #2)	13.80	4.49	14.42
PCDU_Small (Power Conditioning & Distribution Unit_Small)	15.50	10.00	17.05
SA_SVM (SolarArray_SVM)	59.50	5.01	62.48
Grand Total	**102.60**	**5.62**	**108.37**

EP

	mass (kg)	mass margin (%)	mass incl. margin (kg)
Bat_SVM_1 (Battery_SVM #1)	13.80	4.49	14.42
Bat_SVM_2 (Battery_SVM #2)	13.80	4.49	14.42
PCDU_Large (Power Conditioning & Distribution Unit_Large)	19.60	10.00	21.56
SA_SVM (SolarArray_SVM)	69.00	5.05	72.49
Grand Total	**116.20**	**5.75**	**122.89**

EP plus

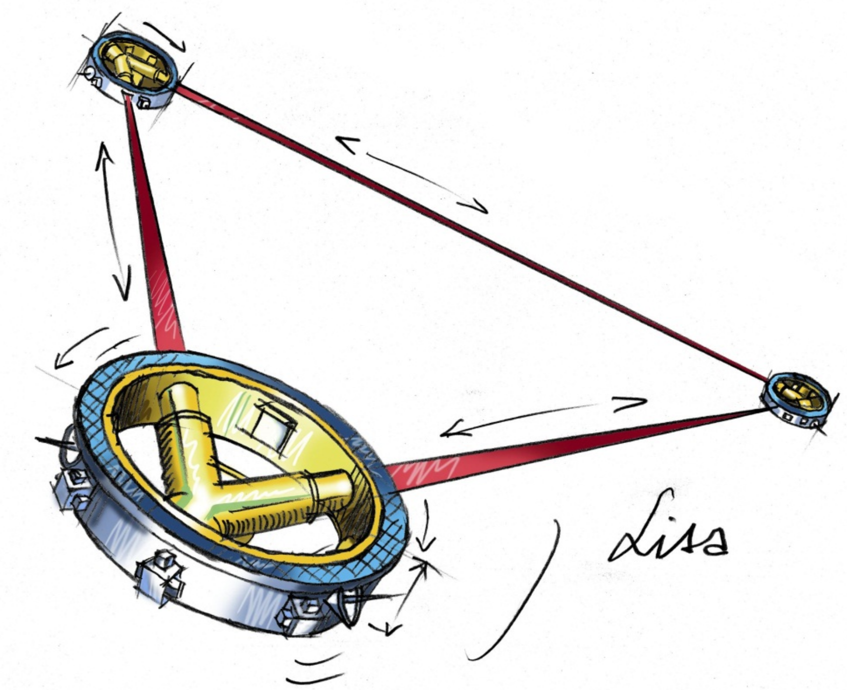
	mass (kg)	mass margin (%)	mass incl. margin (kg)
Bat_SVM_1 (Battery_SVM #1)	13.80	4.49	14.42
Bat_SVM_2 (Battery_SVM #2)	13.80	4.49	14.42
PCDU_Large (Power Conditioning & Distribution Unit_Large)	19.60	10.00	21.56
SA_SVM (SolarArray_SVM)	78.30	4.60	81.90
Grand Total	**125.50**	**5.42**	**132.30**

LISA

Mechanisms

Internal Final Presentation
ESTEC, 05-05-2017

Prepared by the CDF* Team



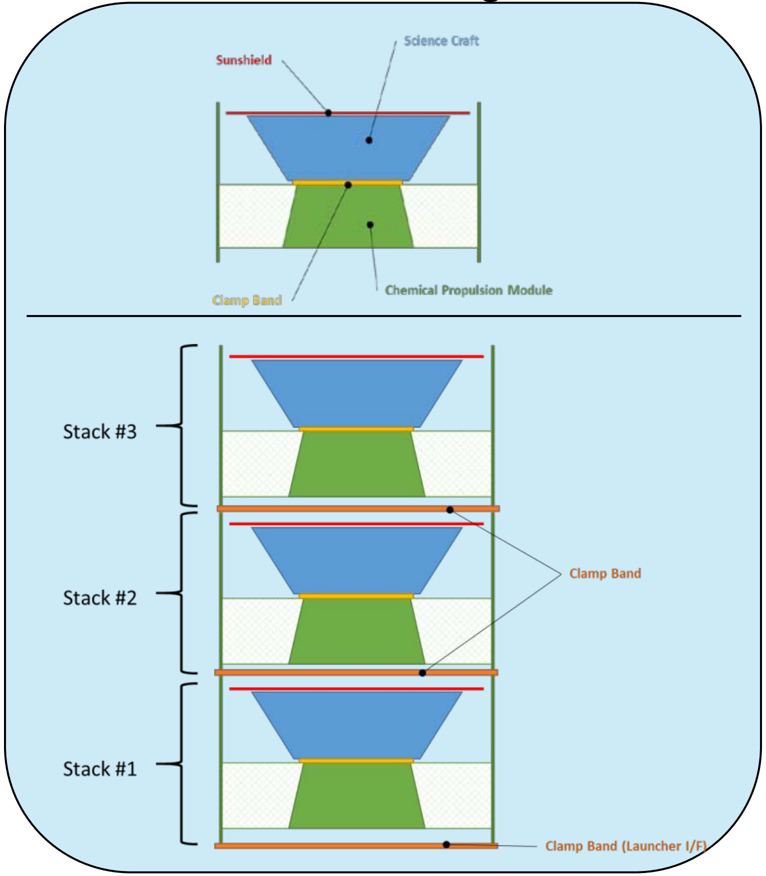
(*) ESTEC Concurrent Design Facility

Requirements

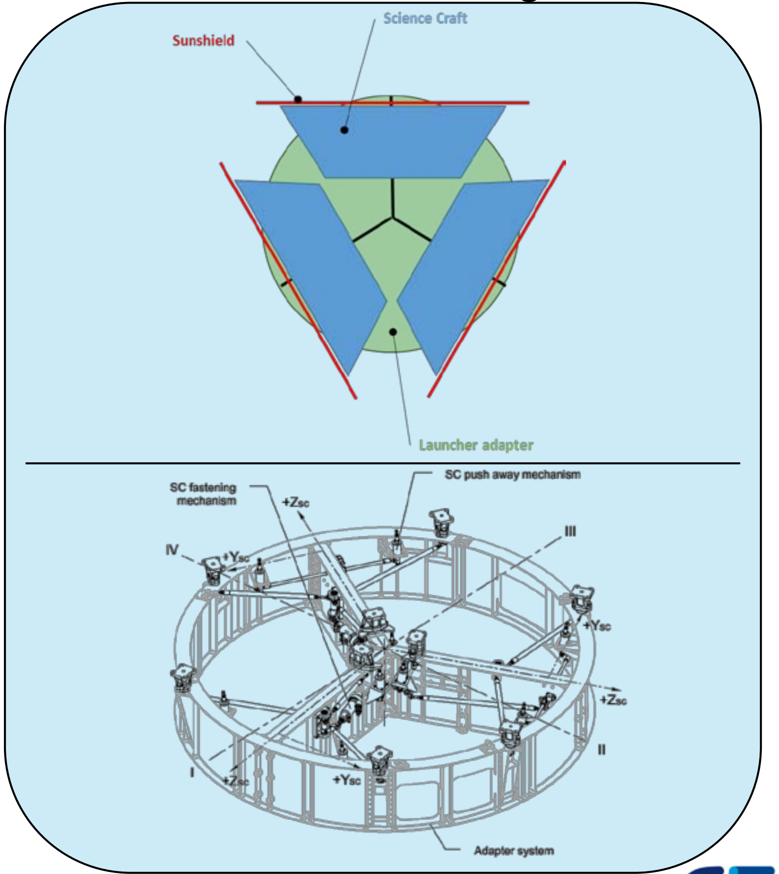
- The maximum tumbling rate after separation from the launcher shall be less than 5°/s (RSS)
- Antenna Pointing mechanism elevation range shall be ±3.5°
- Antenna Pointing mechanism azimuth range shall be [0-360]°
- TM residual (not compensable) linear acceleration, due to Antenna rotation (in its final deployed configuration), shall be less than 10 pm/s^2 (TBC)
- TM residual (not compensable) angular acceleration, due to Antenna rotation (in its final deployed configuration), shall be less than 50 frad/s^2 (TBC)

Deployment: options

STACKED Configuration



SWARM Like Configuration



Deployment: options

- **SWARM like**
  - Pro
    - Low shock/smooth release
    - Mechanical pusher synchronisation
    - Only one pyro actuator per spacecraft
    - Similar loads at the spacecrafts I/F
  - Con
    - Custom design: new development, testing issues
    - Uncertainties (cog, friction, etc…) will lead to tumbling
      - Req. 5°/s RSS
      - Goal 1.4°/s RSS

Deployment: options

- **Stacked**
  - Pro
    - Can be spin stabilized
    - Clamp band connection (heritage)
    - Easier ejection
  - Con
    - Six release system (reliability issue) Many shocks
    - Load on spacecraft are different
    - Command for separation must be binged up to the upper stack
    - Issue on configuration/ structure : high diameter

Deployment: Baseline

Baseline is SWARM like configuration

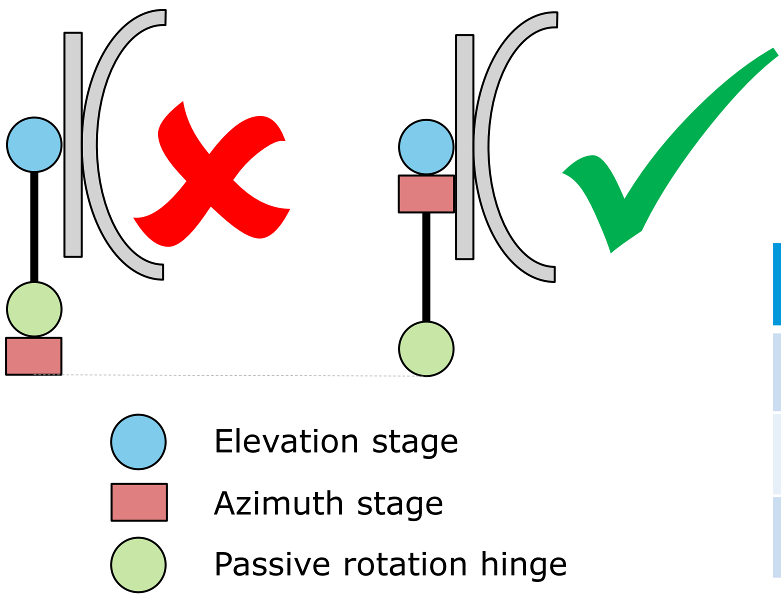
- 4 holding point (mechanically synchronized)
- 1 Pyro
- 4 pusher (mechanically synchronized)
- Requirement on tumbling → max 5°/s (RSS)

EUROCKOT courtesy

Antenna Pointing / Deployment Mechanism

Antenna criticalities:

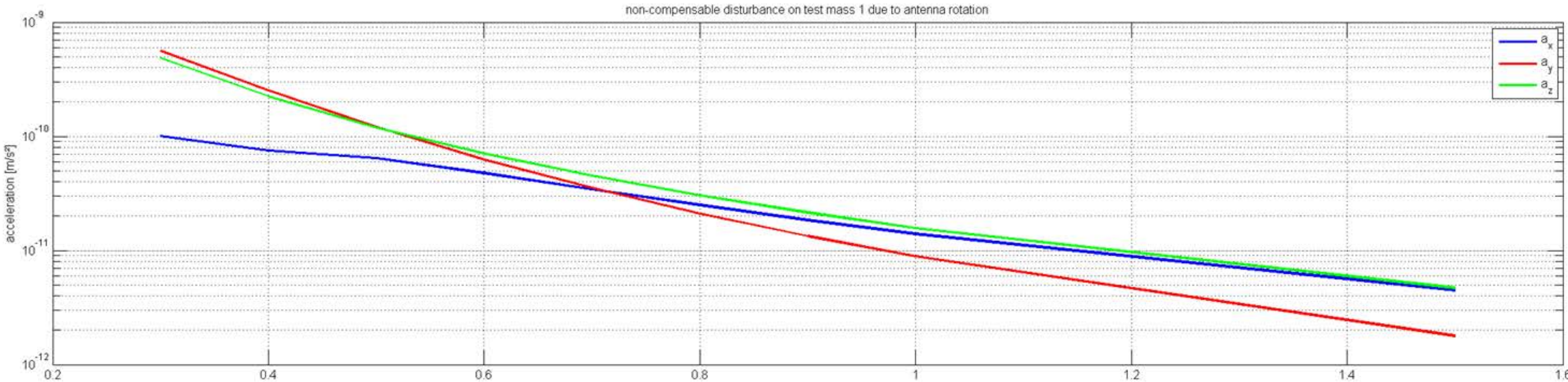
- Disturbances during actuation (action-reaction and microvibration) → impact on DFACS
- Gravitational disturbances → impact on configuration



- In the chosen configuration both the azimuth and the elevation stage are accommodated on the end of a short bracket.
- The Antenna Deployment Mechanism employs 1 HDRM and one passive rotation hinge

	Mass [kg] (excl. margin)	Mass [kg] (incl. 20% margin)
Antenna Pointing Mechanism	12.0	14.4
Antenna Deployment Mechanism	2.0	2.4
Total	**14.0**	**16.8**

Antenna Pointing – Gravitational Disturbance

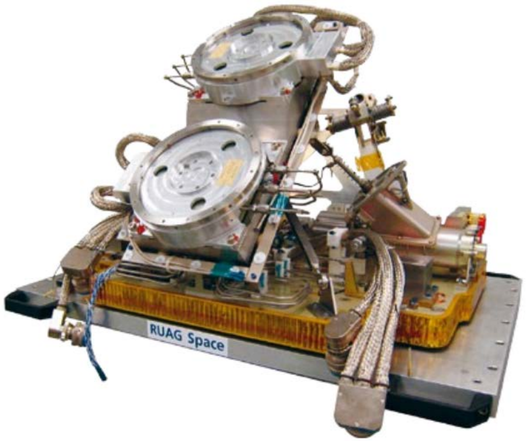


- It was shown that the maximum (non compensable) gravitational disturbance on the test masses due to antenna azimuth rotation can be disregarded as long as the distance between TMs and antenna is sufficiently high (> 1m) and the diameter is not too big (< 350 mm)
- It was shown that the antenna dish can be simplified to a point mass for this assessment with good approximation (as long as the dish diameter is less than 350 mm and the distance is above 1m)

Electric Propulsion: Thruster Pointing Mechanism

The RUAG “TPM” was selected for thruster pointing

- Designed to accommodate up to two Snecma PPS1350 Hall Effect Thrusters
- Pointing range: ±6.5° half cone
- Off the Shelf
- Satisfies requirements
- Relatively light

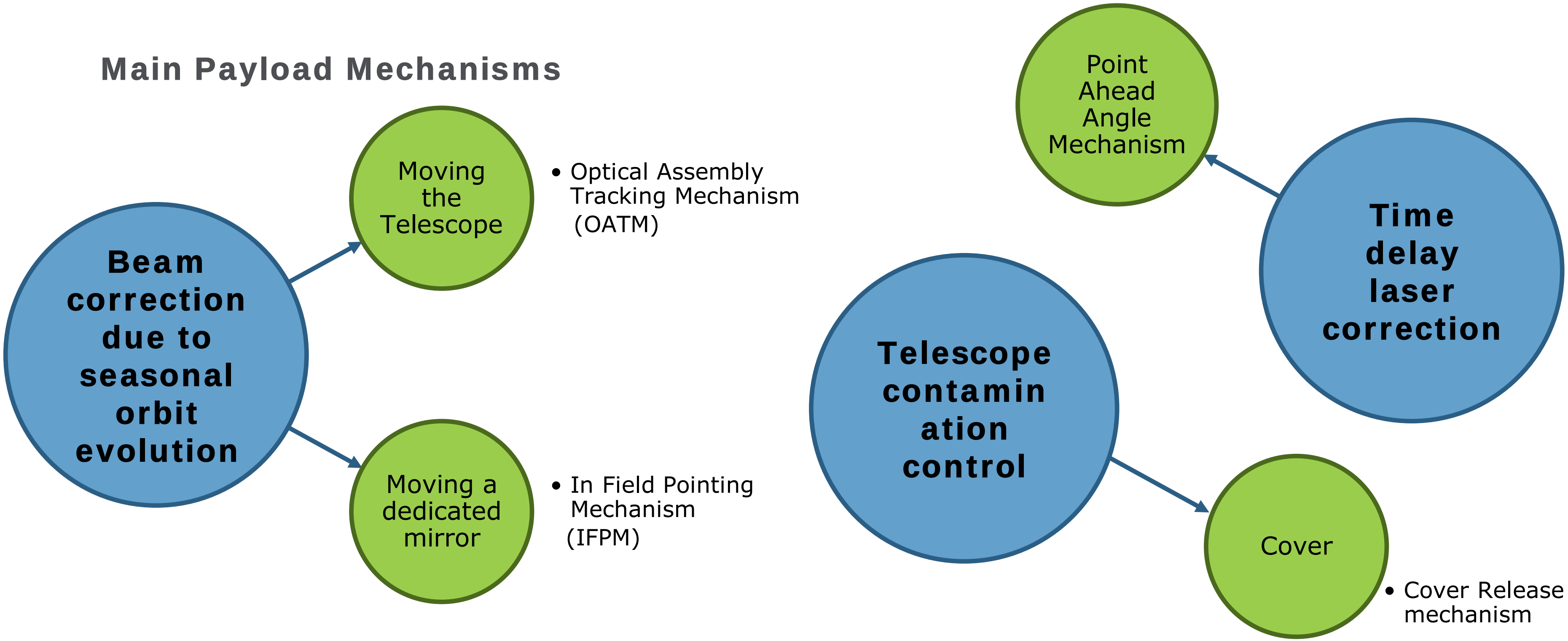


TPM

	Mass [kg] (excl. margin)	Mass [kg] (incl. 5 % margin)
Thruster Pointing Mechanism	**10.6**	**11.13**

Payload Mechanisms: summary

Main Payload Mechanisms



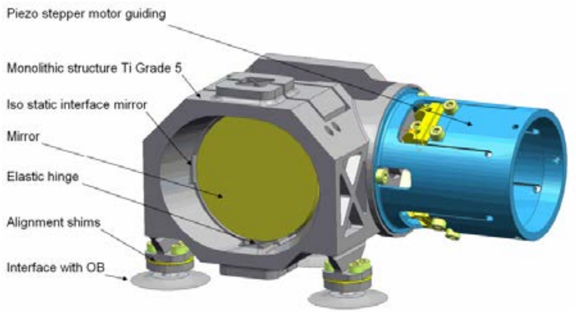
Payload Mechanisms: options

Optical Assembly Tracking Mechanism (OATM)

- No additional noise in the optical path
- Qualification of inchworm/piezowalk ?
- Additional mass (launch lock)

In field pointing Mechanism (IFPM)

- Reduced self gravity perturbation
- Bulky layout
- Demanding optical design
- Tilt/piston coupling effects
- Qualification of inchworm/piezowalk ?
- Already in development but not yet conclusive results.



TNO courtesy

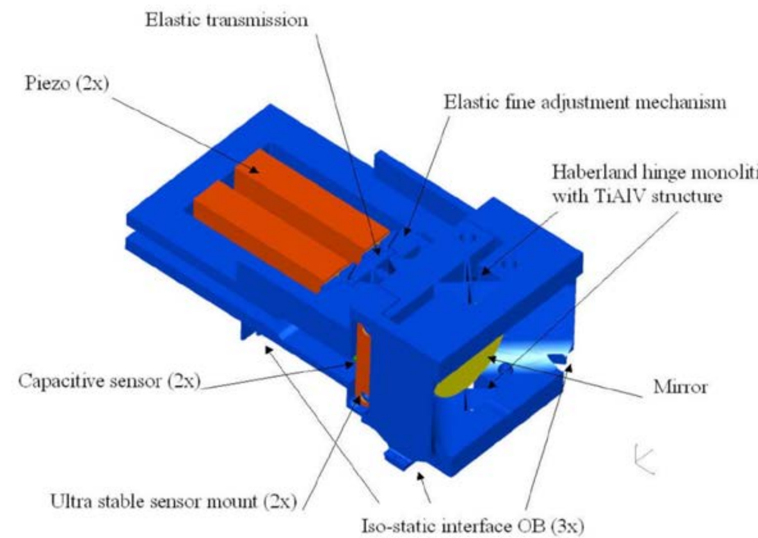
Payload Mechanisms: baseline

IFPM vs OATM

- Not enough info to close the trade.

Point Ahead Angle Mechanism (PAAM)

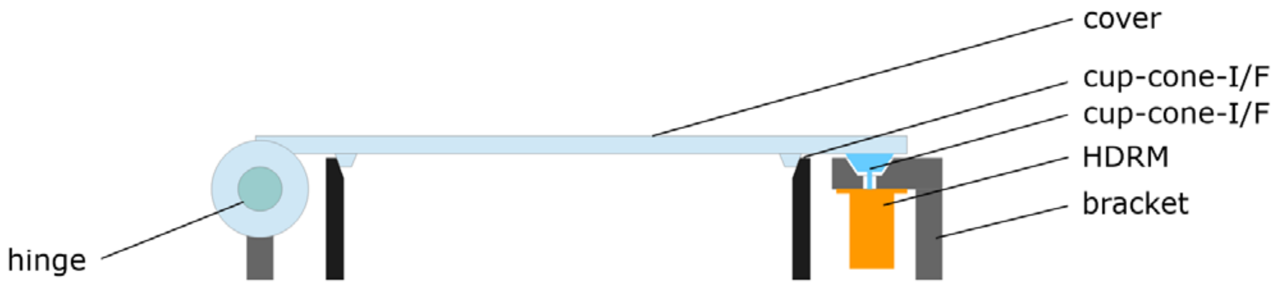
- PAAM mechanism compliant to extreme performance requirements

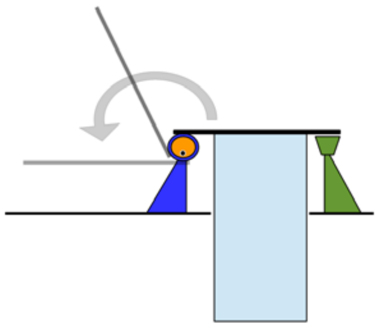


Payload Mechanisms: Telescope Cover

Several trade-offs presented:

1. 1 HDRM vs multiple HDRMs (1 HDRM as baseline)
2. Hinge/HDRM attachment to s/c vs telescope structure (attachment to s/c structure as baseline)
3. External vs internal (at hinge) latching elements (internal latching as baseline)





Mass estimate (including cover): 2.0 kg

Mass Budget - CP option

Service Module	Mass [kg] (excl. margin)	Mass [kg] (incl. margin)
Antenna Pointing Mechanism	12	14.4
Antenna Deployment Mechanism	2	2.4

Propulsion Module	Mass [kg] (excl. margin)	Mass [kg] (incl. margin)
Separation Clamp Band (1666)	9	10.8
Separation Clamp Band (4m)	25	30

Launch Segment	Mass [kg] (excl. margin)	Mass [kg] (incl. margin)
Launch Ring Adapter	200	240

Total	**248**	**297.6**

*Payload mechanisms not considered

Power Budget - CP option

Service Module	Power [W] (while on)	Power [W] (standby)
Antenna Pointing Mechanism	30	5
Antenna Deployment Mechanism	TBD	0

Propulsion Module	Power [W] (while on)	Power [W] (standby)
Separation Clamp Band (1666)	140	0
Separation Clamp Band (4m)	140	0

Launch Segment	Power [W] (while on)	Power [W] (standby)
Launch Ring Adapter	-	0

*Payload mechanisms not considered

Mass Budget – EP & EP optimized

Service Module	Mass [kg] (excl. margin)	Mass [kg] (incl. margin)
Antenna Pointing Mechanism	12	14.4
Antenna Deployment Mechanism	2	2.4
Thruster Pointing Mechanism (2pcs)	2 x 10.6	2 x 11.13
Residual HDRM mass on S/C	5	6

Launch Segment	Mass [kg] (excl. margin)	Mass [kg] (incl. margin)
Launch Ring Adapter	200	240
S/C Separation Mechanism	60	72

Total	**297.2**	**357.1**

*Payload mechanisms not considered

Power Budget – EP & EP optimized

Service Module	Power [W] (while on)	Power [W] (standby)
Antenna Pointing Mechanism	30	5
Antenna Deployment Mechanism	TBD	0
Thruster Pointing Mechanism (2pcs)	12 (per TPM)	0
Residual HDRM mass on S/C	-	-

Launch Segment	Power [W] (while on)	Power [W] (standby)
Launch Ring Adapter	-	-
S/C Separation Mechanism	TBD	0

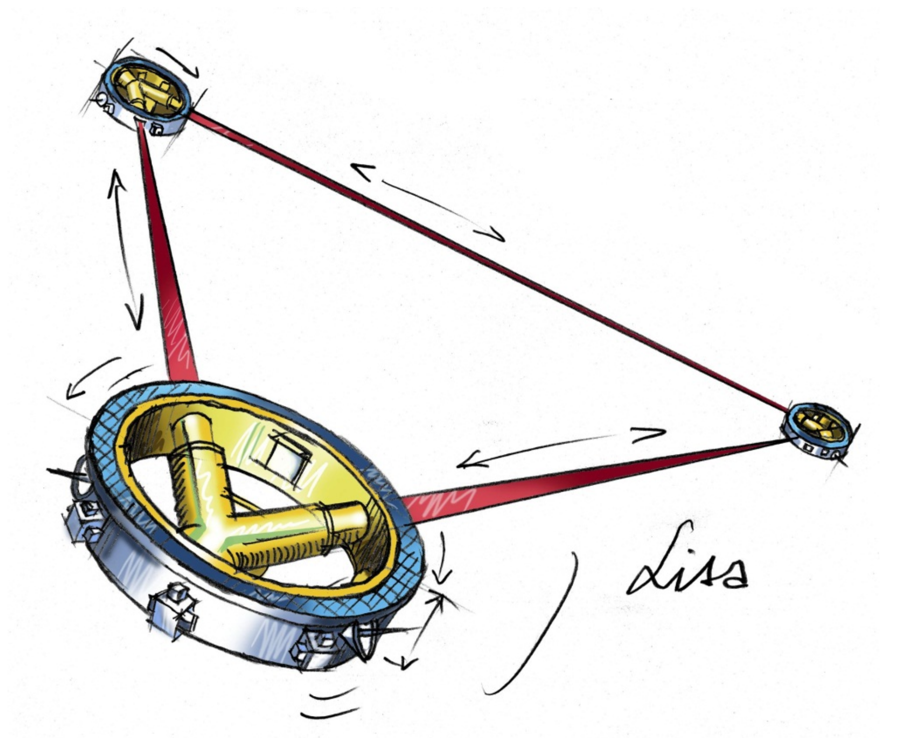
*Payload mechanisms not considered

LISA

Configuration

Internal Final Presentation
ESTEC, 05-05-2017

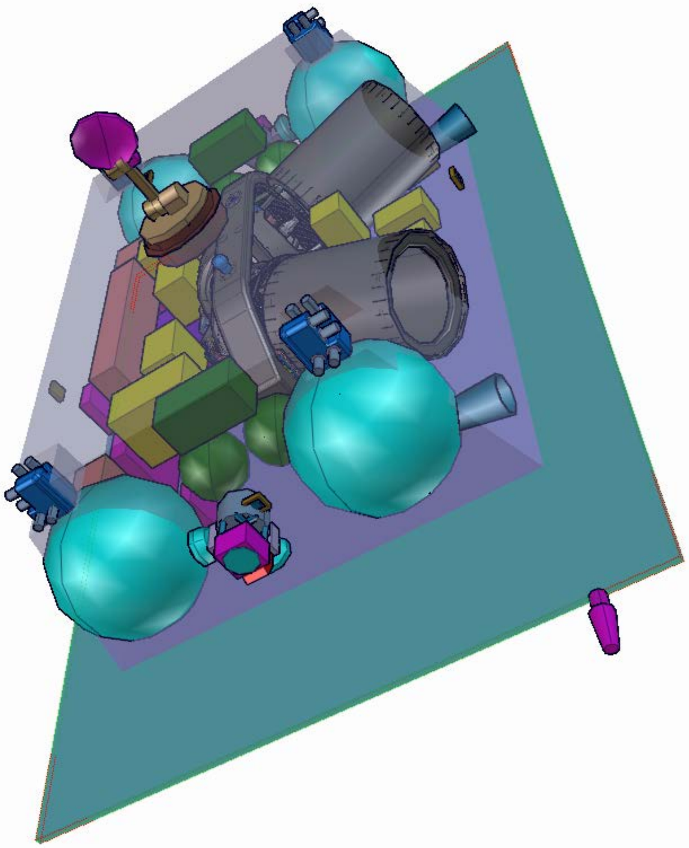
Prepared by the CDF* Team



(*) ESTEC Concurrent Design Facility

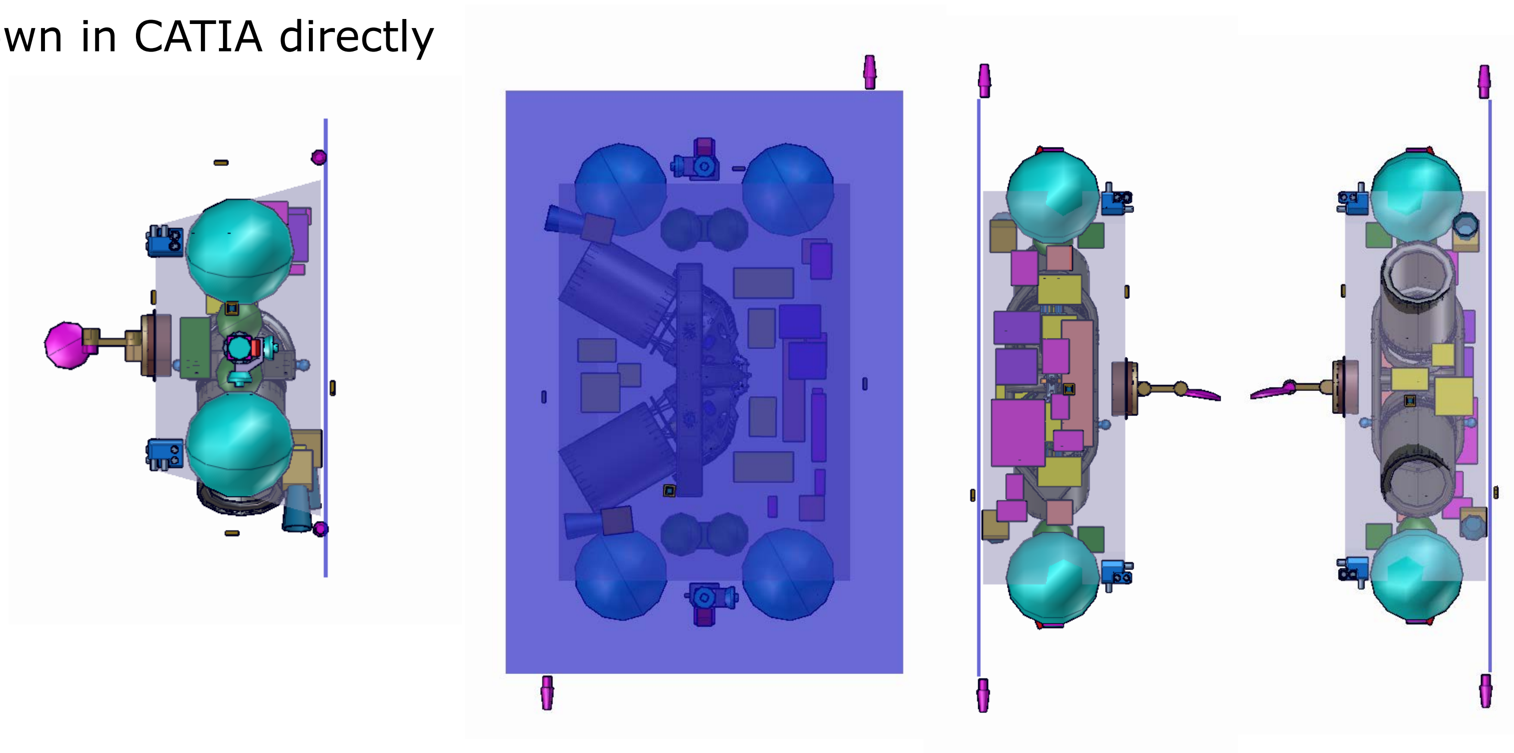
EP Option Configuratin

Shown in CATIA directly



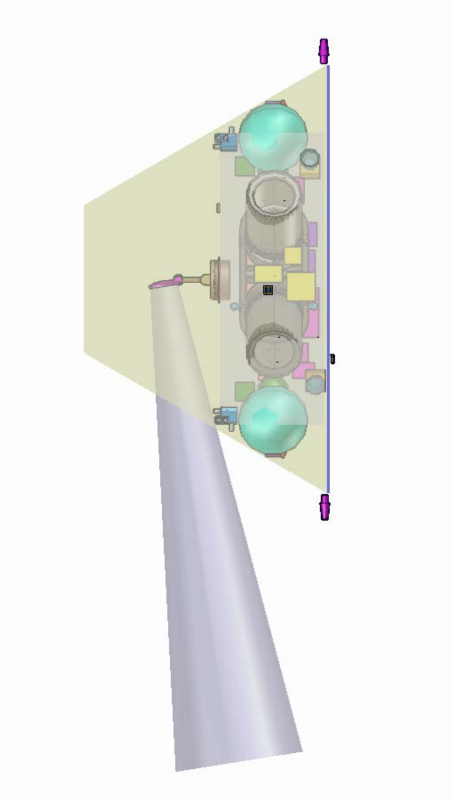
EP Option Configuratin

Shown in CATIA directly



EP Option Configuratin

Shown in CATIA directly

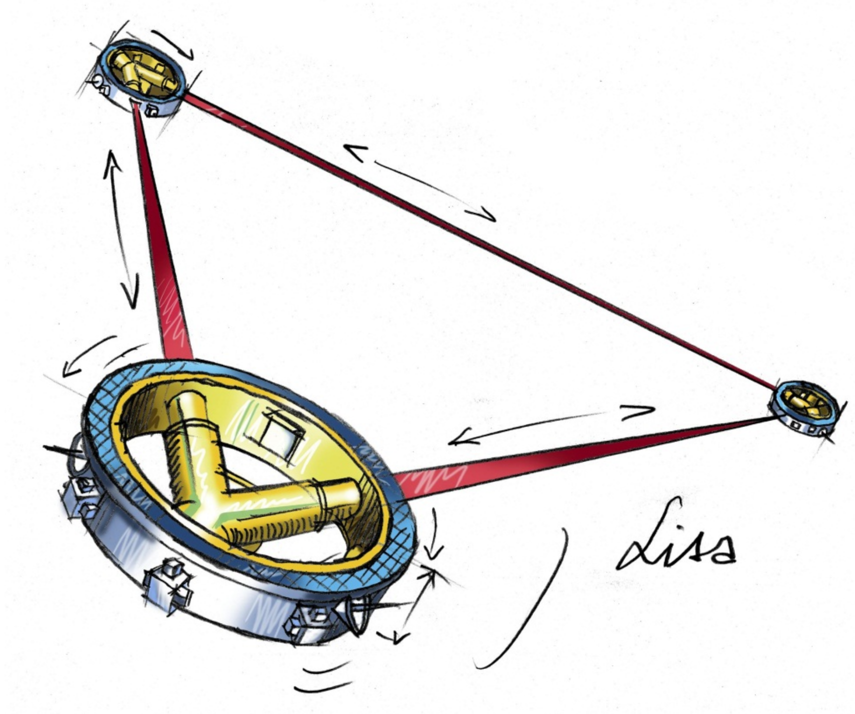


LISA

Structures

Internal Final Presentation
ESTEC, 05-05-2017

Prepared by the CDF* Team



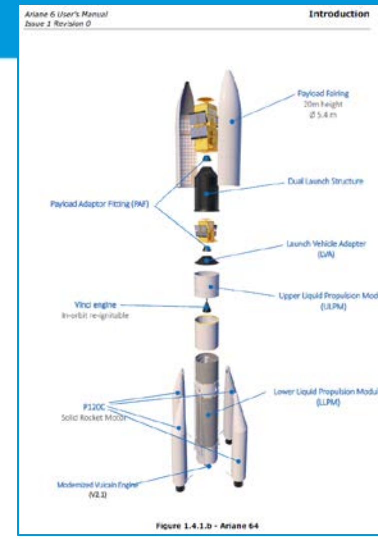
(*) ESTEC Concurrent Design Facility

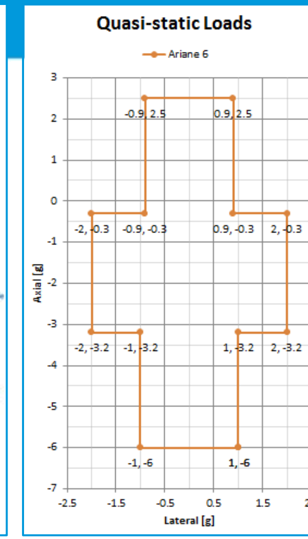
Overview

- Requirements wrt Structure
- Structure – CP option
  - Service Module
- Structure – EP & EP+ option
  - S/C structure
  - Dispenser
  - Stiffness
- Conclusion

Requirements wrt Structure

- Provide stiffness and strength
  - L/V requirements Ariane6
    - 1st Lateral Frequency ≥ 6 Hz
    - 1st Longitudinal Frequency ≥ 20 Hz
    - [Avoid PO range 43 ± 10 Hz]
  - Launch loads (QS, sine, …)
  - ‘Triple launch’ requirements
- Contribute to science mission requirements (minimum disturbance), in particular stiffness
- Provide area, volume, interfaces for the payload and equipment (propulsion, power, …)
- Contribute to protection of payload and other systems from space environment
- Mass-efficient design
- Facilitate AIT operations



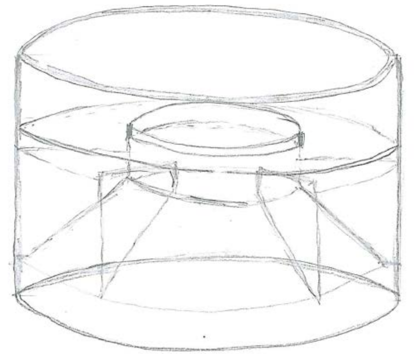


Structure – CP option

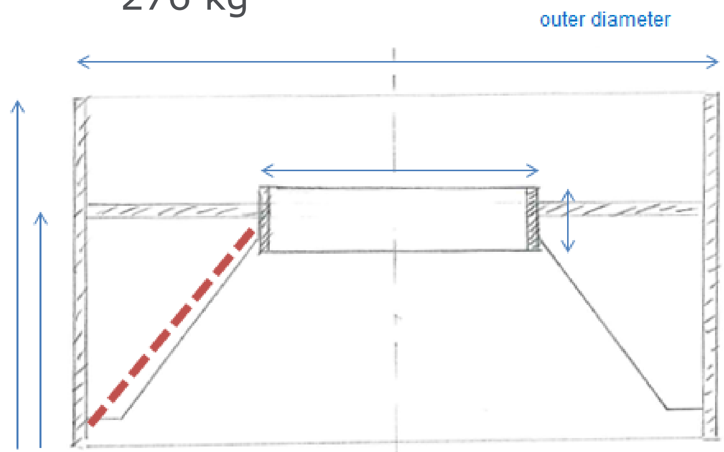
- Service Module including Propulsion Module (interfacing with S/C and L/V)
- Spacecraft S/C Cylindrical configuration
- Loads differ per S/C
  - lower S/C see higher axial loads and bending moments
  - upper S/C see higher lateral acceleration
- Sun-shield with SA is on top of S/C » large diameter of support structure is required which results in a mass-inefficient structure for the PM module interfacing with S/C and L/V
- Mass estimation (tbd)
- Trapezoidal-type design could be envisaged similar to the EP option (not studied here)

Structure – CP option: SVM

- Service Module <u>ex</u>cluding Propulsion Module
  - Large diameter cylinder
  - I/F for spacecraft for separation
  - Shear web or strut design
  - Mass estimation (based on geometry)
    - shear web 311 kg
    - strut design 276 kg

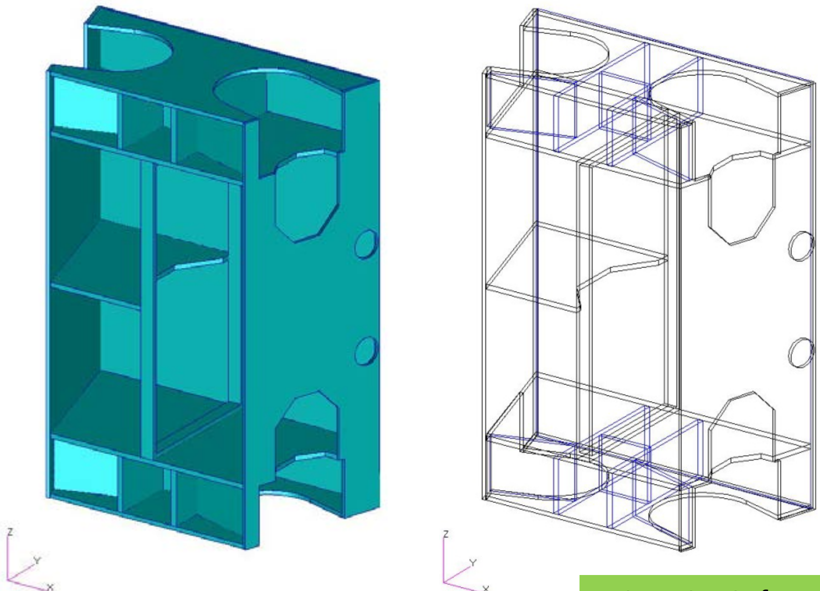


outer diameter	4.000 m
inner diameter	1.666 m
height total	2.500 m
height cylinder inner	0.300 m
location of inner platform	1.600 m
m40 cfrp struts/panels	1580.0 kg/m3
aluminium	2800.0 kg/m3
alu 3/16-5056-0.001	50.0 kg/m3



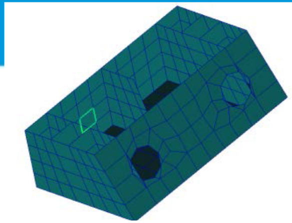
Structure – EP & EP+ option

- Trapezoidal configuration
  - Minimum disturbance for Science
  - Stiffness-driven
- Release of 3 S/C challenging
  - Stiffness of S/C important
  - Simulation using 3 (dummy) S/C
- Dispenser above Ariane 6.4 LVA
  - LVA 2624 / LVA 3664 / etc
  - Static moment at base of dispenser seems non-problematic (tbc)
  - Overflux @ LVA I/F <10% needs to be monitored
- Mass estimation (based on geometry), tbc by stiffness analysis
  - 158 kg using M55J fibres
  - 173 kg using YS-90A fibres (+ 10%) (currently in system mass budget)



note: cut-outs for tanks are not part of latest configuration

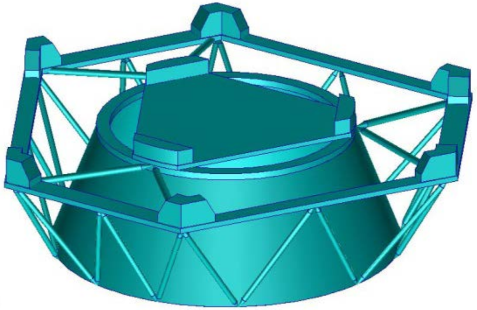
Structure – EP & EP+ option: S/C structure



- Trapezoidal / Swarm-type S/C Configuration
- 'Boat-type' structure with bulkheads and closed volumes to provide high stiffness in particular in torsion
- Sandwich panels based on high-modulus CFRP fibres and Aluminium HC
  - Baseline fibre Toray M55J
  - Option NIPPON GRAPHITE YS-90A (pitch fibre for thermal reasons, 8-10% heavier wrt M55J for structure only): higher modulus, lower strength

	TYPE	FIBER PROPERTIES			COMPOSITE PROPERTIES									
60 V_f epoxy resin RT		Tensile Strength	Tensile Modulus	Strain	Tensile Strength	Tensile Modulus	Tensile Strain	Compressive Strength	Flexural Strength	Flexural Modulus	ILSS	Density	CTE	Thermal Conductivity
		MPa	GPa	%	MPa	GPa	%	MPa	MPa	GPa	kgf/mm2	kg/m3	10^{-6}/K	Cal/cm·s·°
M40 CFRP		2,740	392	0.7	1,470	240	0.6	1,030	1,270	200	8	1,810		
M55J CFRP	UHM	4,020	540	0.8	2,010	340	0.6	880	1,230	280	7	1,910	-1.1	0.372
YS-90A CFRP (pitch fibre)	UHM	3,530	880	0.3	1,900	520	0.3	360		520		2,180	-1.5	

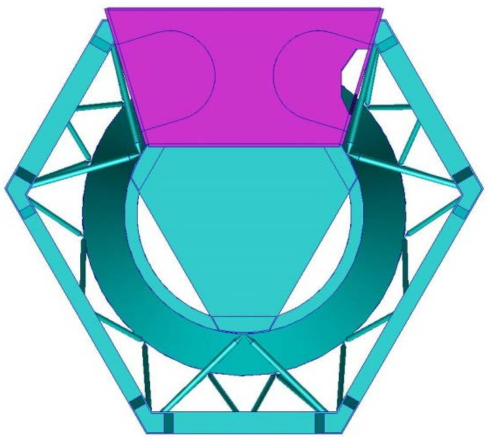
Dispenser – EP & EP+ option : dispenser

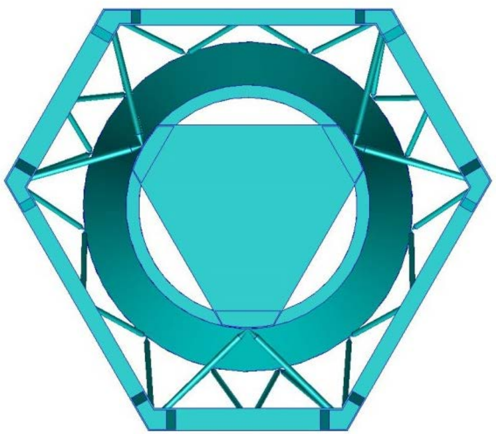


Dispenser above A6.4 LVA 2624

- Smaller L/V I/F diameter allows more space for struts, …
- Smaller L/V I/F diameter provides less bending stiffness
- Overflux wrt LVA 2624 needs to be monitored
- Better mass-efficiency than 'LVA 3664 option'
- Mass estimation tbd based on stiffness analysis, target $f_{lateral}$ » $f_{lateral,LV}$

A6UM: … Off-the-shelf adapters, with separation interface diameter of 937 mm, 1,194 mm, 1,663 mm, 1,666 mm and 2,624 mm are available.





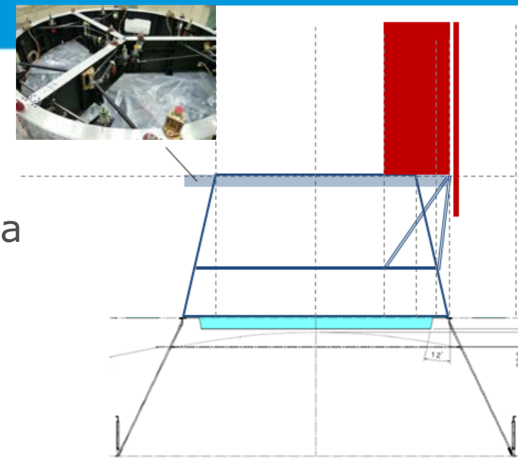
Dispenser – EP & EP+ option: dispenser

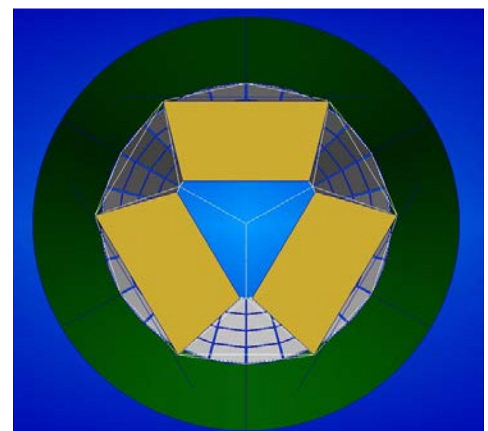
Dispenser above A6.4 LVA 3664

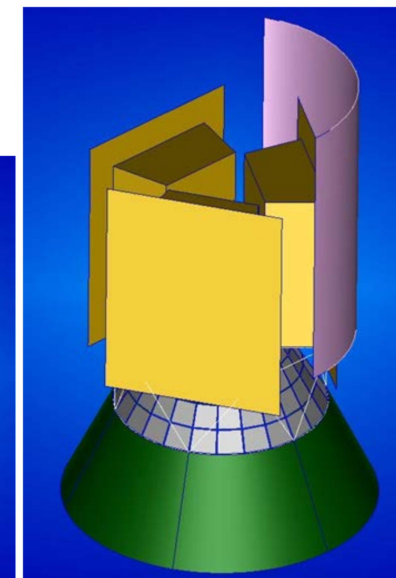
- Cone similar to LVA foreseen for Athena
- Mass estimation

mass impact wrt standard LVA			160.0 kg
volume/height impact wrt standard LVA			0.000 m
	mass (tbc)		600.0 kg
	diameter upper		3.664 m
	height		1.900 m
	diameter lower		5.400 m
	surface area		29.741 m2
dispenser	height		2.000 m
	mass structure		686.2 kg
		cone	*464.4 kg*
		cross, struts, …	*221.8 kg*
	mass mechanisms		*60.0 kg*
	mass total		*746.2 kg*
	diameter upper		3.200 m
	diameter lower		3.664 m

- Stiff, good overflux performance
- Options seems <u>not feasible</u> due to radial clearance of SA and dispenser struts





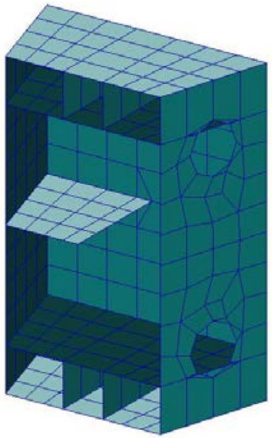


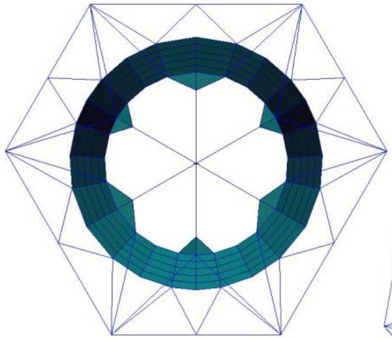
wrt A6 this is a non-standard I/F

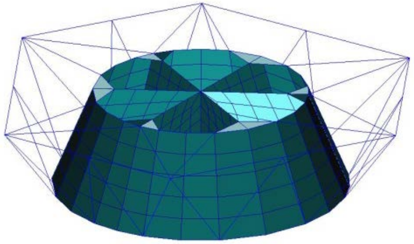
Structure – EP & EP+ option: stiffness

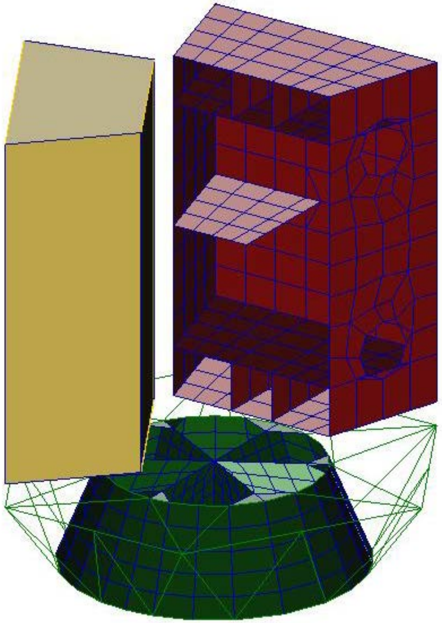
In order to verify the feasibility of the 3 S/C on a dispenser an eigenmode analysis will allow to get an indication that the dispenser & S/C stiffness is sufficient

- S/C based on trapezoidal Configuration
- Dispenser based on A6 LVA 2624 interface
- FEMs of dispenser and S/C are available (mass correlation tbd)
- Target minimum lateral frequency tbd (on Swarm experience?)









Conclusion

- The structural concept for the S/C foresees a design based on high-modulus CFRP fibres to provide minimum disturbance during Science mode and to allow a well-controlled separation from the dispenser
- Launching 3 S/C in a single launch is a challenge mainly for
  - Dispenser design (Swarm- / Galileo- / mixed-type / …)
  - Minimum interference between the S/C on the dispenser required
  - Simulation by analysis & test of S/C separation from dispenser, qualification of dispenser including separation system

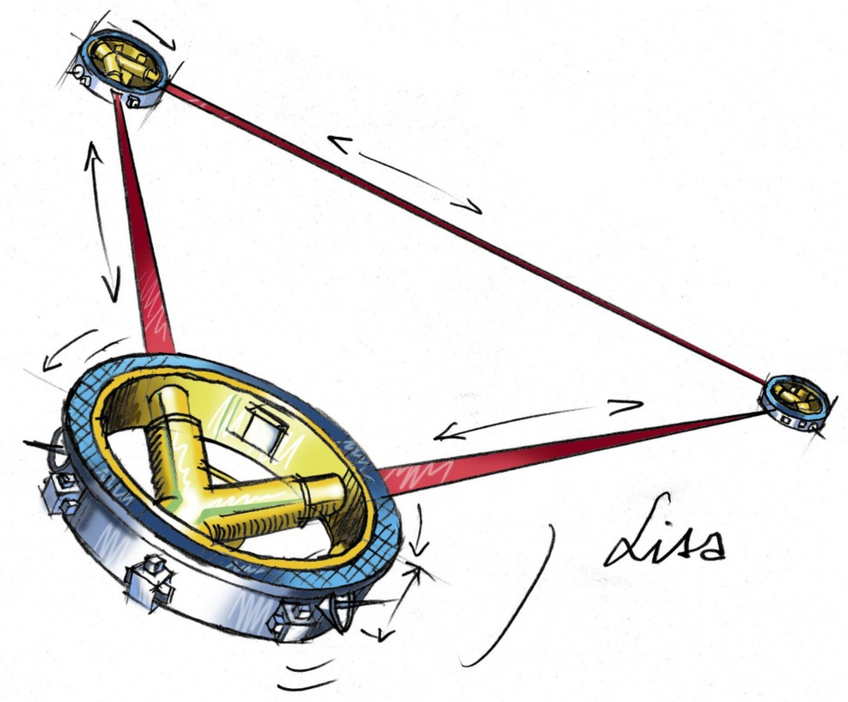
- Next steps
  - Study and trade-off on dispenser configuration(s) and separation scenario (increase TRL, improve robustness of design, evaluate cost impact on overall cost)

LISA

Thermal Control

Internal Final Presentation
ESTEC, 5th May 2017

Prepared by the CDF* Team



(*) ESTEC Concurrent Design Facility

Requirements and Design Drivers (1)

Item	Amount	Margin	Science	Acquisition	Accelerometer	Fast Discharge	Peak power [W]	Dissipated power [%]
Telescope	**2**	**20**	**0.52**	**5.00**	**0.52**	**0.52**		**99.00**
Big mirror	3		0.00	0.00	0.00	0.00		
Small mirror	4		0.00	0.00	0.00	0.00		
Mechanism in-field pointing	1		0.00	2.50	0.00	0.00		99.00
Mechanism telescope motion	1		0.00	2.50	0.00	0.00		99.00
Structure (rods)	2		0.00	0.00	0.00	0.00		
Cover + Mechanism	2		0.00	0.00	0.00	0.00	10.00	99.00
Baffle	1		0.00	0.00	0.00	0.00		0.00
Optical Truss SED	4		0.13	0.00	0.13	0.13		99.00
Optical Bench	**2**	**20**	**3.75**	**5.45**	**3.75**	**3.75**		**99**
Photodiodes + pre-amplifiers	30		0.13	0.13	0.13	0.13		
Fibre Injector Switch	2		0.00	0.00	0.00	0.00		
Re-focussing mechanism	1		0.00	0.00	0.00	0.00		
Acquisition sensor	1		0.00	1.70	0.00	0.00		
Baseplate/Mirrors	1		0.00	0.00	0.00	0.00		
Point Ahead Angle Mechanism	1		0.00	0.00	0.00	0.00		
Gravitational Reference Sensor	**2**	**5**	**0.00**	**0.00**	**0.00**	**0.00**		**0**
Caging Release			0.00	0.00	0.00	0.00	2.50	
GRS Front-End Electronics	**2**	**10**	**56.00**	**84.00**	**84.00**	**56.00**		**99**
Phasemeter	**2**	**20**	**50.00**	**50.00**	**50.00**	**50.00**		**99**
Laser (4, 2=OFF)	**2**	**20**	**100.00**	**100.00**	**100.00**	**100.00**		**98**
Laser Frequency Stabilisation (2, 1=OFF)	**1**	**50**	**6.00**	**6.00**	**6.00**	**6.00**		**99**
Charge Management System	**2**	**10**	**5.00**	**0.00**	**0.00**	**10.00**		**99**
Caging Control Unit	**2**	**10**	**0.00**	**0.00**	**0.00**	**0.00**	**18.00**	**99**
Diagnostics	**1**	**10**	**15.80**	**0.00**	**0.00**	**0.00**		**99.00**
Magnetometers	11		0.80	0.00	0.00	0.00		
Radiation Monitors	1		6.00	0.00	0.00	0.00		
Heaters	1		1.00	0.00	0.00	0.00	5.00	
(Payload PU included) Electronics	1		0.00	0.00	0.00	0.00	0.00	
Payload Processing Unit	**1**	**20**	**30.00**	**30.00**	**30.00**	**30.00**		**99**
Acquisition CCD Electronics	**1**	**20**	**0**	**6.70**	**0.00**	**0.00**		**99**
GRAND TOTAL			**566.83**	**622.92**	**600.05**	**560.45**		

	Dissipation (W)	
	Transfer	Science
PCDU	175	120
OBC	40	40
TWT	11	106
XB TRSB	25	25
PPU	75	0

Requirements and Design Drivers (2)

- The main requirement and design driver for LISA thermal control is to ensure the proper conditions for the LISA payload, i.e. preserving the payload within the ranges for temperature and thermal stability listed below

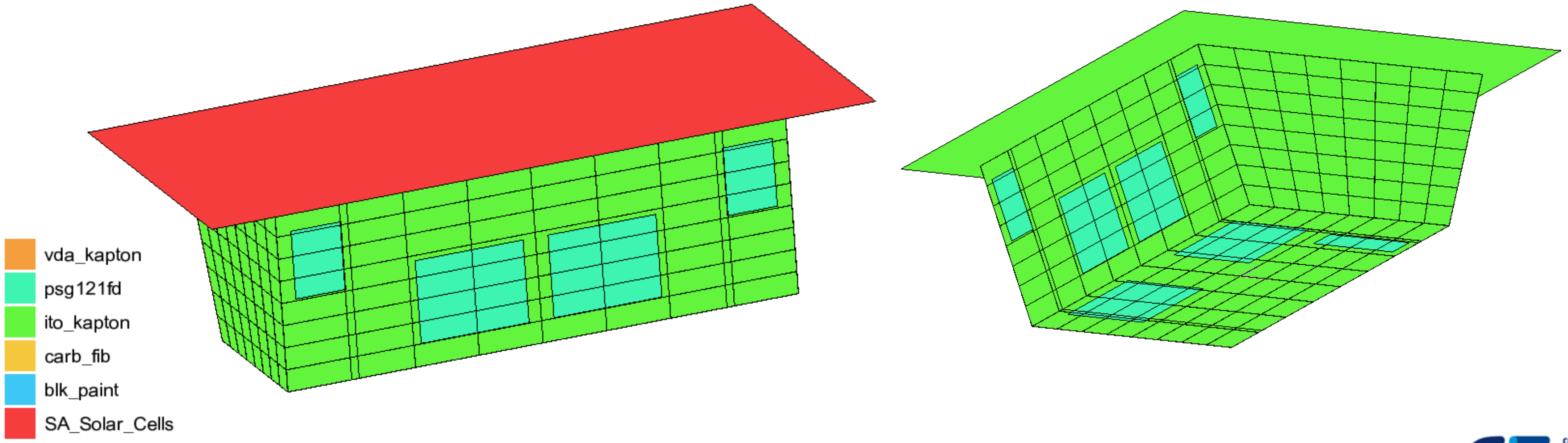
	Temperature Limits (°C)				Thermal Stability
	Min OPT	Max OPT	Min NOPT	Max NOPT	[K/root Hz @ 0.1 mHz]
Telescope	-100	30	-100	50	?
Optical Bench	10	30	0	40	1E-05
Gravitational Reference Sensor	10	30	-10	30	1E-04
GRS Front-End Electronics	10	30	-20	50	?
Phasemeter	10	30	0	40	1E-03
Frequency Distribution System	10	30	0	40	?
Laser (4, 2=OFF)	23	29	-10	30	1E-03
Laser Frequency Stabilisation (2, 1=OFF)	10	30	0	40	1E-04
Charge Management System	10	30	0 -> -10	40->50	-
Caging Control Unit	10	30	0 ->-10	40->50	-
Diagnostics	10	30	0 ->-10	40->50	-
Payload Processing Unit	10	30	0 ->-10	40->50	-

Assumptions (1)

- Thermal design can is based on two sizing cases
  - Transfer phase (TFM), EP operating, P/L non-operational
  - On-station phase (SciM), P/L operating, EP non-operational
  - Following Session 11, the detumbling operation is reduced to less than 10 minutes, with no need for specific thermal control measures
- The EP thrusters are isolated on external panels and conductively decoupled,
  - thermal control independent of platform, not taken into account in model
- For the transfer phase, LISA is constrained to a 40° SSA for pitch and a 30° SSA for roll along the trajectory
- For the on-orbit phase, LISA is constrained to a 30° SSA in all directions
- Constraints for TCS hardware:
  - No classical cycling heaters due to temperature stability requirements
  - No heat pipes due to possible gravity and/or microvibration effects
  - No mechanical coolers (e.g. for low temperature acquisition detector)

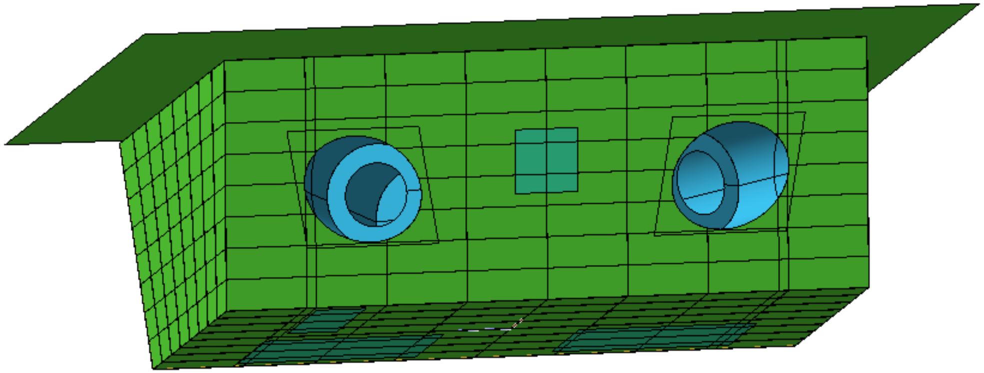
Baseline Design (1)

- Similar approach to LISA Pathfinder
- White-painted radiators are used to enable efficient rejection of the heat generated by the units
- Where not required as radiators, external surfaces are covered by MLI
- ITO Kapton MLI is used to prevent a charge build-up on the spacecraft



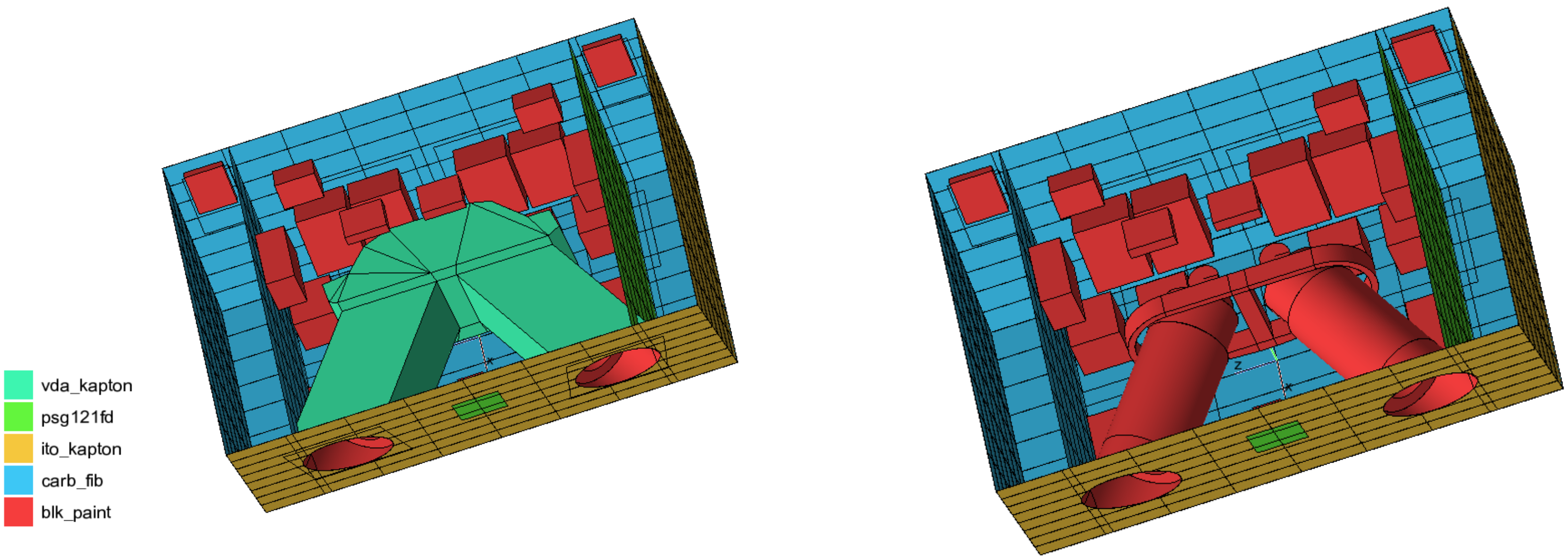
Baseline Design (2)

- Underside of solar array is covered with MLI
- For insulation between the solar panel and the bus structure, the same type of titanium blades as on LPF are foreseen



Baseline Design (3)

- The required isolation of the telescope assembly is achieved with low-conductance mounts and an internal MLI tent
- Equipment mounted directly on panels, black painted internally
  - redundant, co-located items, doublers may be needed



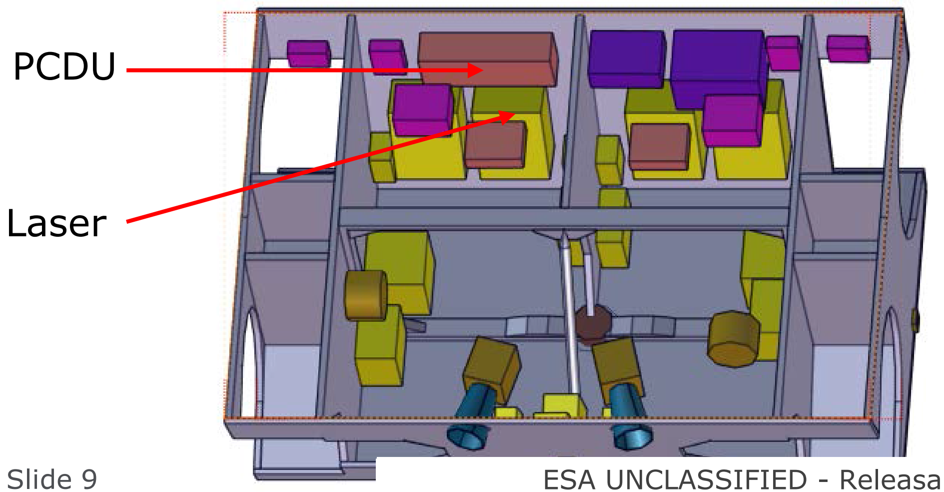
Baseline Design (5): Results of modelling

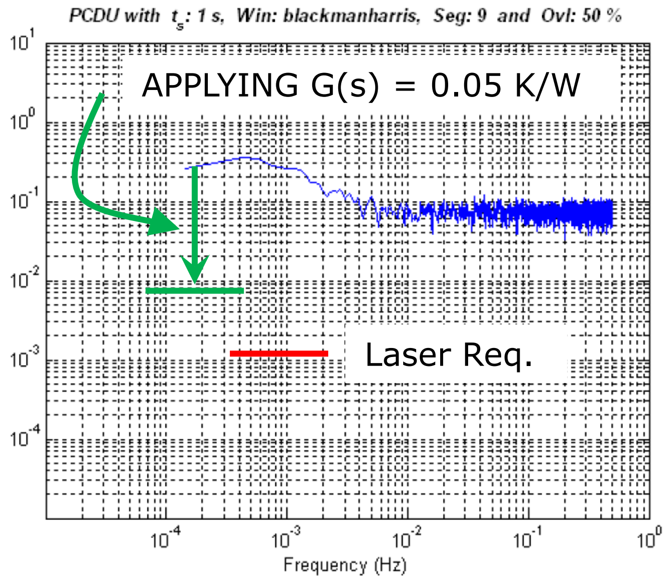
- Standard requirements for science missions have been used for uncertainty:
  - +/- 15K uncertainty to be applied to calculation
  - For heater controlled items uncertainty can be reduced, but quoting from science requirements:
    - *Increased heating power assuming units need to be 15K warmer than actually required.*
    - *Increased radiator area to dissipate heat assuming it is 15K colder than actually predicted.*
- Using current model and assumptions **should be feasible to keep all equipment within temperature limits** for both transfer and science (see additional slides)
- Heater power is computed as:
  - Transfer: **445W**
  - Science: **147W**

Thermal stability

Thermal stability

- Equipment shall operate continuously with constant dissipation (as far as possible)
- For LPF thermal noise sources were considered to be: solar flux variation, PCDU, OBC, FEE, cold gas equipment dissipation
- Guiding principle should be to thermally decouple “noisy” equipment from sensitive items, current layout can be optimised





Equipment list

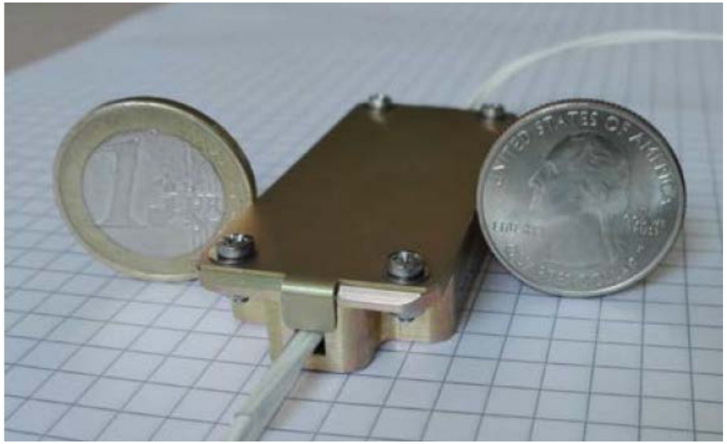
- Heaters (445W transfer, 147W science)
- MLI → 16kg

Options and technical development (1)

Heaters

- The on-station phase temperature stability was successfully achieved for LPF with a set of multiple trim heaters, operated constantly in various combinations.
- For LISA active control will be necessary, at least more flexibility with trimming will be required
- **Linear** heater control (open loop) may be needed for LISA, similar to GRACE follow-on

2016 Conference paper: "THE GRACE FOLLOW-ON QUIET ELECTRICAL POWER SYSTEM" Manfred Amann, Mike Gross, Hauke Thamm

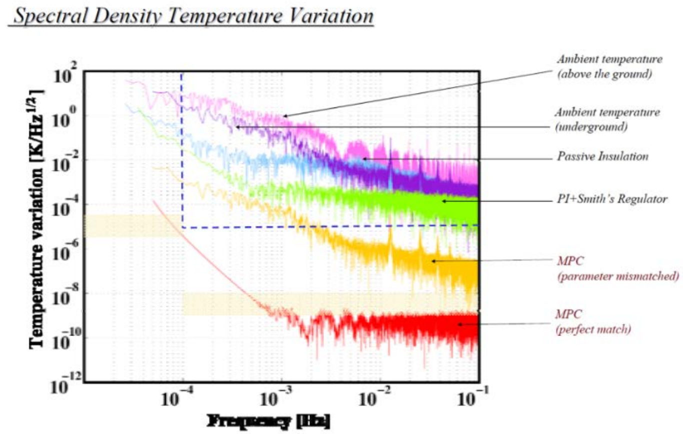


Options and technical development (2)

Control and Temperature measurement

- For the optical bench thermal control, work by Stanford University has investigated Model Predictive Control (MPC)
- Study identifies that temperature measurement resolution needs to be improved by factor 100
- Also for **thermal testing** new techniques may be required; new sensors, IR, lock-in thermography etc.

(presentation at 50th anniversary of Stanford University Department of Aeronautics and Astronautics in 2008, Higuchi et al)



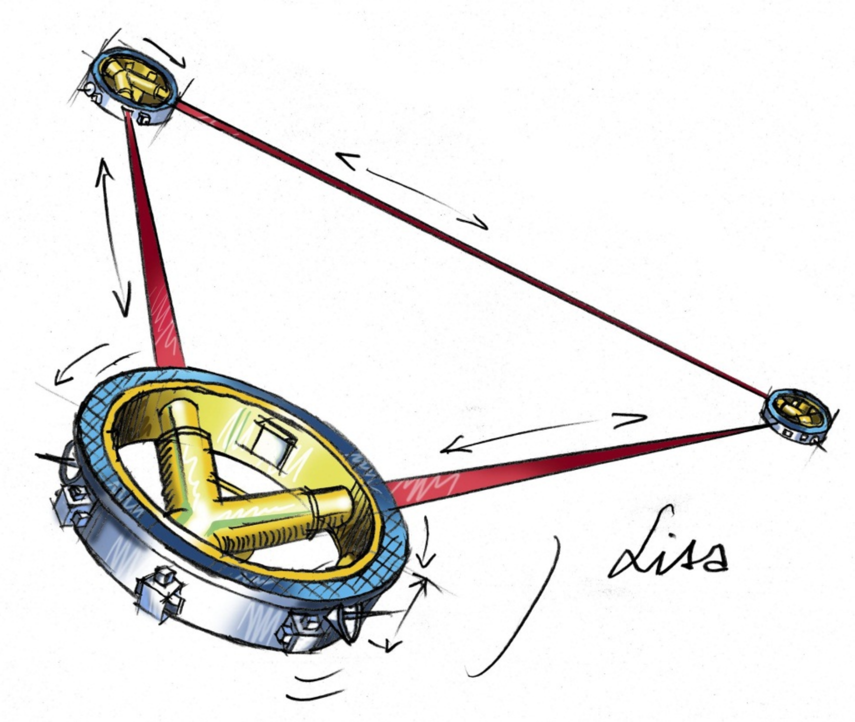
 ESA UNCLASSIFIED - Releasable to the Public Thermal

LISA study

Risk

Internal Final Presentation
ESTEC, 05-05-2017

Prepared by the CDF* Team



(*) ESTEC Concurrent Design Facility

Safety and Dependability Requirements (part 1)

Reliability and Fault Management Requirements in MRD		Covered area of Saf.&Dep.
REQ-02	The lifetime of S/C shall be compatible with the mission requirement.	*Life time*
REQ-03	Single-point failures with a severity of catastrophic or critical for S/C and mission (as defined in ECSS-Q-ST-30C/40C) shall be eliminated or prevented by design of the S/C and mission units.	*Failure tolerance/avoidance*
REQ-04	Single-point failures in the S/C and mission (other than catastrophic or critical) shall be avoided in the design of the S/C and mission units. Retention of single-point failures in the design shall be declared with rationale and is subject to formal approval by ESA.	*(SPF severity of consequences needs to be specified, e.g. catasrophic or critical according to ECSS-Q-ST-30C/40C)*
REQ-05	Retention in the design of single-point failures of any severity rating is subject to formal approval by ESA on a case-by-case basis with a detailed retention rationale.	*Failure tolerance/avoidance*
REQ-06	A failure of one component (unit level) shall not cause failure of, or damage to, another component or subsystem within and between S/C and mission units.	*Failure propagation*
REQ-07	The failure of an instrument shall not lead to any 'Safe Mode'* of the S/C and mission units. *Remark:* ** relaxiation of requirement like: ...shall not lead to an 'Ultimate Safe Mode' to be clarified*	*Failure propagation*
REQ-08	The design shall allow the identification of on-board failures and their recovery by autonomously switching to a redundant functional path. Where this can be accomplished without risk to spacecraft and instrument safety, such switching shall enable the continuity of the mission timeline and performance.	*FDIR Level*
REQ-09	Where redundancy is employed, the design shall allow operation and verification of the redundant item/function, independent of nominal use.	*Redundancy verification*
REQ-10	ESA/ADMIN/IPOL Space Debris Mitigation for Agency Projects) if applicable* *Remark:* *The applicability of this requirement will be defined by the responsible LEOP operation/ launch authority*	*Space Debris Mitigation*
REQ-11	The S/C design shall be compliant with applicable safety related launch requirements *(e.g. CSG Safety Regulations)*	*launch safety*

Safety & Dependability Requirements (part 2)

Reliability and Fault Management Requirements in MRD		Covered area of Saf.&Dep.
REQ-01.1	The overall reliability of the mission* shall be ≥ 85% at end of life. *Remark:* *'mission' is here understood as the deployment and operation of the LISA constallation over the specified Life time starting with the separtaion of 1st of 3 S/Cs.*	***Reliability***
REQ-01.2	The availability of the constellation for science observation during nominal operational phase shall be ≥ ..TBD..% over a period of ...TBD..hour. Whereby the number of corrective maintenance shall be ≤ .. TBD.. over a period of .. TBD..hours. The MTTRS for a planned maintenance shall be ≤ ..TBD..hours. The MTTRS for a corrective maintenance shall be: - ≤ ..TBD..hours in case of 'Intermediate Safe mode' and < ..TBD..hours in 'Ultimate Safe Mode'* - < ..TBD..hours in case of 'Intermediate Safe mode' and < ..TBD..hours in 'Intermediate Safe Mode'* including activities in the ground segment** reasonably needed for recovery of full science observation. *Remark:* *MTTRS - Mean Time To Recover (full science) service* *TBD - To Be Done* ** 'Ultimate/ IntermediateSafe Modehas to be defined in a separate technical requirements* *** activities in the ground segment'has to be specified in a separate operational requirement*	***Availability***

Mission Success Criteria's

Program	**PRO1**: Laser Interferometer Space Antenna (LISA) in frame of Cosmic Vision L3 **PRO2:** detect and observe low-frequency Gravitational Waves (sensing methodology is laser interferometry between free flying test masses) **PRO3:** A constellation of three spacecrafts (S/C) is required, flying in a triangle ('mission')
SRE/EOP/ HSF/ Technical	**TEC1:** The mission operates successfully over the designated mission lifetime 10 ...12.25years (max.) (app. 2 years mission transfere & Commissioning + 4 years nominal operation + 4 years extended operation). **TEC2:** A reliability of >85% at the end of mission/ program success **PER1:** Availability* of constellation for science performance (major science objectives) ** depending from science needs;* *defined on e.g. periodical basis and with uper limits for MTTRS (Mean Time To Recover [major science] performance)*
Protection / Safety	**SAF1:** Catastrophic hazard* (2 Failure/Error Tolerance), critical hazard* (1 Failure/Error Tolerance) incl. undesired human performance (human error/ failure) **SAF2**: No SPF can lead to catastrophic hazards* on mission level No performance degradation owing to SPF, and no failure propagation. ** in terms of safety/ protection on S/C level, in terms on science services on mission level* **PRO1:** Mission shall be compliant with requirements applicable for space debris mitigation if requested
Schedule	**SCH1**: All architecture elements are available and their FRR successful for the launch (NLT 2034) **SCH2:** The contributions from international partners are available at the relevant milestones of the development schedule **SCH3**: TRL 5/6 for all critical subsystems at the time of mission adoption end Phase A/B1 (est. 2024) **SCH4**: Low development risk during Phase B2/C/D
Cost	**COS1:** CaC for ESA ≤ **1050M€** (2014 e.c.) -> A Class Mission(2034 e.c.)

Risk Policy – Severity definition (part 1)

Score Severity name Level (ECSS)	Risk domains			
	Dependability #1 [Performance / Technical]	Safety[health] #1/#2+ Protection[property+environment+...]#3	Schedule #1	Cost #1
5 Catastrophic 1	Performance (e.g. science): * Failure leading to the impossibility of fulfilling the objectives of the mission performance, e.g.: loss of mission or failure propagation: * form lower system level to highest system level * from S/C to constellation * leading to loss of safety-related barriers	Safety: * Loss of life, life-threatening or permanently disabling injury or occupational illness; Protection * severe detrimental environmental effects * Loss of launch site facilities.	Delay resulting in project cancellation	Cost increase resulting in project cancellation
4 Critical 2	Performance (e.g. science): * Failure resulting in a major reduction in mission/ campaign performance (e.g 70-90% of overall science return) Technical: * Critical degradation of the mission (system functionalities critical for performance can not be replaced or recovered)	Safety: * temporarily disabling but not life-threatening injury, or temporary occupational illness; Protection: * Major damage #4 to flight systems or ground facilities or to public or private property * Major detrimental #4 environmental effects * Major damage to ground facilities.	Critical launch delay by 24-48 months	Critical increase in estimated cost by 100-200 M€ (20 .. 50%)
3 Major 3	Performance (e.g. science): * Failure resulting in a major reduction in mission/ campaign performance (e.g 30-70% of overall science return) Technical: * Major degradation of the mission (some system functionalities can not be replaced or recovered)	Safety: * Minor injury, minor disability, minor occupational illness. Protection: * Minor damage #4 to flight systems or ground facilities or to public or private property * Minor detrimental #4 environmental effects * Minor damage to ground facilities.	Major launch delay by 6-24 months	Major increase in estimated cost by 40-100 M€ (10 .. 20%)

Risk Policy – Severity definition (part 2)

Score Severity name Level (ECSS)	Risk domains			
	Dependability #1 [Performance / Technical]	**Safety[health] #1/#2+** **Protection**[property+environment+...]#3	**Schedule #1**	**Cost #1**
2 Significant ./.	Performance (e.g. science): * Failure resulting in a substantial reduction in mission/ campaign performance (e.g. 10-30% of overall science return) Technical: * Minor degradation of mission (e.g.: system is still able to control the consequences	Safety/ Protection: * severity of consequences are less than catastrophic, critical and major severity but higher than minor severity	Significant launch delay by 3-6 months	Significant increase in estimated cost by 10-40 M€ (5 .. 10%)
0/1 no/ Minor or Negligible 4	Performance (e.g. science): * no/minimal reduction for mission/ campaign performance (e.g. 0 - 10% of overall science return) Performance: * No/ minimal consequences for system system functionality can be replaced or recovered with operational constrains	Safety: * No/ minimal consequences * casualty risk <10E-4 (controlled/ uncontrolled re-entry) * collision risk with manned systems <10-4 Protection: * No/ minimal consequence * lifetime in LEO <25years (re-entry, grave yarding) * avoidance of generation of space debris (sat. dis-	No/ minimal consequences - delay by 1-3 months	No/ minimal consequences (in-crease in estimated cost by 0-10 M€) (<5%)

#1 Reduction of 'Performance', 'Delay', 'Cost overrun' coming from insufficient TRL status('Technological Risk')
'Programmatic Risk' has to be considered in risk domains (Dependability, Safety, Schedule, Cost) effected by mission objectives
#2 'Safety'- stands for all consequences related to human health and well being
#3 'Protection' stands for consequences to be expect out-side of safety, mission/ campaign and project
#4 has to be specified based on national and international lows and regulations
applicable regulation of entities involved in project, mission/ campaign,

Risk Index – Severity vs. Likelihood

"*" safety related (comp./ funct./ SW/ Human Performanced)

Severity Score	A	B	C	D	E
5	A5* (<10^4) / A5* (<10^4)	B5*	C5*	D5*	E5*
	A5	B5	C5	D5	E5
4	A4	B4	C4	D4	E4
3	A3	B3	C3	D3	E3
2	A2	B2	C2	D2	E2
1	A1	B1	C1	D1	E1
0	no risk				
	A	**B**	**C**	**D**	**E**

Likelihood

Risk Index	Risk Magnitude	Proposed Actions (during assessment phase)
B5*, C5*, D5*, E5* D5, E5, E4	Very High Risk	**Unacceptable risk:** implement mitigation action(s) - either likelihood reduction or severity reduction through new baseline with appropriate party
C5, D4, E3	High Risk	**Unacceptable risk:** *see above*
A5*, B5, C4, D3, E2	Medium Risk	**Acceptable risk for study however unacceptable for project:** therefore implement further reduction action(s) with responsible party/ project partners
A5, A4, B4, B3, C3, C2, D2, D1, E1	Low Risk	**Acceptable risk:** control, monitor; during project seek responsible work package management attention.
A1-3, B2, B1, C1, 0	Very Low ('0' - no) Risk	**Acceptable risk/ no risk:** *see above; '0' - no actions to be taken e.g. in case the risk is eliminated*

"*" safety related

Risk assessment based on the **'Worst case'** approach!

Major Risks – sorted by..

Design/ TRL & realisation
Launch (preparation) & IOT including 'Space Debris Mitigation'
Cruise
Mission performance including 'Planetary Protection'
Overall Cost/ Schedule + Programmatic
Other

Major Risks – Design & realization **(part 1)**

- **DIV –> dependability risk - Mechanism failure**
- **DV –> dependability risk - S/C reliability in constellation (loss of mission)**
- **DVI -> dependability risk - Science availability of constellation (anomalies)**
- **DVII1/2/3-> prog./sched./cost risk – options for propulsion system (mass, TRL,..)**
- **DIXa,b -> dependability risk - SPF S/C antenna (SPF, micro vibration)**
- **DX -> dependability risk – Laser links/ ranging (robustness) -> DVI**
- **DXI -> dependability risk – design information science instruments (completeness)**

Major Risks – Launch, Cruise, Mission(perform.) (part 2)

- **LI/II –> safety risk – propulsion system, dangerous media/ high pressure**
- **LIIIa/b –> dependability risk - S/C deployment /collision**
- **CI -> dependability risk – trajectory anomaly**
- **MI -> dependability risk - robustness of constellation acquisition**
- **MIIa/b -> dependability risk – Micro-meteoroids (loss/ science impact)**
- **PI –> programmatic risk – draw back of consortium members**
- **OCI -> cost risk – cost overrun**

Major Design Risks (part 1)

DIV -> dependability risk – mechanism failure

Risk scenario: several mechanism (PL release/ optical bench/ telescope opening/ Point Ahead Angle Mechanism/ Payload Mech.(OATM or IFPM)/ …) could lead to loss of mission/ critical reduction of science return .. due … failure in any parts of the mechanism

Initial risk => likel.: med.(10-2)/ sev.: catast.(science) => **high risk**

Mitigation: **- adequate redundancies if possible** -> decrease of likel.

- adequate reliability targets; verhigh TRL

- intensive PA approach (especially testing)

Final risk => likel.: low(10-3)/ sev.: catast.(science) => **med. risk**

Resid./add. risk*: negligible contribution to cost

Remark: resid

Major Design Risks (part 2)

DV –> dependability risk - S/C reliability in constellation

Risk scenario: mission success reliability is usually 85% (for usually 1 S/C!); however constellation based on 3 fully functional S/Cs (85% S/C rel. leads to mission rel. of 62%)

Initial risk => likel.: high(10-1)/ sev.: crit.(loss of mission) => **very high risk**

Mitigation: **- suitable reliability requirements for crit. functions/ subsys.**-> decrease of likel. (keep an eye in Common Cause failure for effective use of redundancies!)

- **use of highly reliable subsystems (high TRL) incl. functional redundancies on constellation level;** -> decrease of likel./sev.
- **intensive PA approach incl. full dependability assessment** -> decrease of likel.

Final risk => likel.: med.(10-2)/ sev.: major(science) => **med. risk**

Res./abb. risk: .. no..

Major Design Risks (part 3)

DVI -> dependability risk - Science availability of constellation*

Risk scenario: anomalies of 3 S/C can contribute to science unavailability

Initial risk => likel.: high.(10-1)*/ sev.: major(science) => **med. risk**

Mitigation: - suitable availability & reliability requir. for crit. function / subsystems reliability / constellation & anomaly recovery (MTTRS) -> decr. of likel./sev.

- **advanced FDIR capacity; high S/C autonomy due to advanced OBSW;** -> decr. of sev.
- **intermediate safe mode** (recovery-time relevant subsystems stay alive)

Final risk => likel.: med.(10-2)/ sev.: signif.(science) => **low risk**

Res. / abb. risk: .. no..

Remark:

** very much depending from science needs*

Major Design Risks (part 4.1)
3 propulsion options

DVII1-> programmatic risk – chemical propulsion (mass)
(opt.1) Risk scenario: - 300..750kg/ S/C over mass budget could lead to cancellation of project
- add. cold gas micro propulsion* for science operation (see residual risk)

Initial risk =>: likel.: max.(10-0)/ sev.: catast.(prog.)=> **very high risk**

Mitigation: **use of other propulsion options** -> risk eliminated

Final risk => likel.: -/ sev.: -(-) => **no risk**

Res./add. risks: * limited life time experiences for requested operation period of 4.. 6 years
* *see also* LI, LII, MIIa (safety during launch preparation, micro meteoroids)

DVII2 –> any risk - EP propulsion
(opt.2) Risk scenario: - well established electrical propulsion systems for trajectory
- add. cold gas micro propulsion* for science operation needed (see residual risk)

Initial risk => likel.: n/a / sev.: n/a => **no risk**

Mitigation: ***..not needed..***

Res./add. risks: * limited life time experiences for requested operation period of 4.. 6 years
* development risk for Helium tanks (not available in needed size)
* *see also* LII, MIIa (safety during launch preparation, micro meteoroids)

Major Design Risks (part 4.1)
3 propulsion options

DVII3 –> schedule / depend. risk - EP+EMP propulsion

(opt.3) Risk scenario: - risk due to performance and qualification issues related to low TRL of relatively new electric micro propulsion system*,** for science operation (low trust)

- addition equipment in comparison to other opt. (reliability/ availab. impact);
- cold gas micro propulsion only for de-tumbling (LEOP)

Initial risk => likel.: high(10-1)/ sev.: crit.(schedule, depend.) => **high risk**

Mitigation: - fast integration of EMP into ESA development program -> decrease of sev./ like

- intensive testing program.

Final risk => likel.: med.(10-2)/ sev.: major(schedule, depend.) => **low risk****

Res. / add. risk: * limited life time experiences for requested operation period of 4.. 6 years

** procurement might be an risk issue due to development monopole

Major Design Risks (part 5)

DIXa -> dependability risk - S/C antenna (SPF)

Risk scenario: several single point failure sources in antenna systems

Initial risk => likel.: med.(10-2)/ sev.:catast.(loss of mission) => **high risk**

Mitigation: - **intensive testing of mech. for movab. antenna** -> decrease of likel.

- **functional redundancy via different ways of. S/C-S/C & S/C-ground communication** -> decrease of likel.

Final risk => likel.: low(10-3) / sev.: catast.(loss of mission) => **med. risk**

Res./add. risk: impact on science reliability availability

DIXb -> dependability risk – antenna mechanism (micro vibration)

Risk scenario: micro vibration of antenna mech. might has an impact on science availab.

Initial risk => likel.: high.(10-1)/ sev.: maj.(science) => **med. risk**

Mitigation: - **adequate design requirement** -> decrease of likel./ sev.

- **intensive test program** -> decrease of likel.

Final risk => likel.: med.(10-2)/ sev.: signif.(science) => **low risk**

Res./add. risk: negligible cost impact

Major Design Risks (part 6)
not considered in Risk Index

DX -> dependability risk – Laser links / ranging (robustness) TbC

Risk scenario: instability of laser link/ ranging has an impact on science availab.

Initial risk => likel.: ….(10-..)/ sev.: ….(science) => **…. risk**

Mitigation: - **..TbC..** -> decrease of likel.

Final risk => likel.: ….(10-..)/ sev.: ….(science) => **…. risk**

Res. / add. risk: ..no..

DXI -> programmatic risk – design information science instruments (incomplet.)

Risk scenario: no full set of design information available during study
(e.g. risk assessment)

Initial risk => likel.: max.(10-0)/ sev.: signif.(prog.) => **med. risk**

Mitigation: - **delta study for science instruments** -> decrease of likel.

Final risk => eliminated => **no risk**

Res. / add. risk: possible but neg. impact expected on several design details

Major Launch (prep.) & Deploy./ IOT Risks (part2)

LI -> safety risk - ground personal (CP)

(DVII1) Risk scenario: toxic chemical propulsion e.g.MON/MMH 4*198/ S/C or Hydrazine or ..) and high energy release (He 1..2kg)

Initial risk => (>10-4)/ sev.: catast.(life threat) => **very high risk**

Mitigation: Safety+Launch regulations (design & handling)-> decrease likel./sev.

Final risk => likel.: min.(10-4)/ sev.: catastr.(life threat) => **low risk**

Res./add. risk: ..no..

LII -> safety risk - ground personal (CP, EP)

(DVII1, DVII2) Risk scenario: health issues due to high pressure comp. (cold gas tank up to 310bar)

Initial risk => (>10-4)/ sev.: catastr.(life threat) => **very high risk**

Mitigation: Safety+Launch regulations (design & handling)-> decr. of likel./sev.

Final risk => likel.: min.(10-4)/ sev.: catastr.(life threat) => **low risk**

Res./add. risk: ..no..

Major Launch (prep.) & Deploy./ IOT Risks (part 2)

LIII/IV-> dependability risk - mission deployment/ collision risk

Risk scenario: loss of mission due to ..

* tumbling of S/C after separation + limited battery capacity before complete S/C deployment
* collision possibility with other S/Cs after release from dispenser

Initial risk => likel.: med.(10-2)/ sev.: catast.(loss of mission)=> **high risk**

Mitigation: - minimizing tumbling rate -> decrease of likel.

- adequate operation procedures

- including detailed contingency procedures

Final risk => likel.: low (10-3) / sev.: catast. (loss of mission) => **med. risk**

Res./add. risk: ..no..

Major Cruise Risks

CI -> dependability risk – trajectory anomaly

Risk scenario: loss of mission due to deviation in trajectory (late discovery anomaly and insufficient time for recovery from any kind of critical PF failure)

Initial risk => likel.: med.(10-2)/ sev.: catast.(loss of mission) => **high risk**

Mitigation: - **frequently control of TM + appl. operation proc.** -> decr. of likel.

- **bacon signal from S/C s**

Final risk => likel.: min(10-4)/ sev.: catast.(loss of mission) => **low risk**

Res. / add. risk: ..no..

Major Mission performance Risks (part 1)

MI -> dependability risk - robustness of constellation acquisition

(see also DX 'laser link/ ranging')

Risk scenario: impact on science availability

Initial risk => likel.: high(10-1)/ sev.: crit.(science) => **high risk**

Mitigation: - Scanning of laser beam is required (~90 min for 175 uRad)

- absolute sensing of the incoming laser angular position;

- gyro mode for short term. attitude stabil. -> decrease likel.

Final risk => likel.: low(10-2)/ sev.: crit.(science) => **low risk**

Resid. Risk: increased number of SM (see DVI)

MIIa/b -> dependability risk – Micro-meteoroids (loss of mission/ science impact)

(DVII1, DVII2)

Risk scenario: in 6.25/12.25a .. 8 to 15 penetrations (e.g. for CP/ EP cold gas tanks) could lead to loss of mission(w.c.)/ impact on science availability

Initial risk => a. likel.: low(10-3)/ sev.: catast.(dep.) => **med. risk**

b. likel.: med.(10-2)/sev.: signif.(dep.) => **low risk**

Mitigation: a. **adequate shielding requirements** -> decrease likel. **espec. for tanks (CP, EP)**

Final risk => a. likel.: min(10-4)/ sev.: catast.(science) => **low risk**

Major Mission performance Risks (part 2)

MIII -> dependability risk – Radiation

Risk scenario: during life time (6.25 .. 12.25a)

radiation effects (SEE*) sensitive equipment

… due … TNID** (*hard to shield)*

Initial risk => likel.: med.(10-2)/ sev.: catast.(loss of mission) => **high risk**

Mitigation: **- early identification of TNID sens. equipment** -> decr. of likel.

(e.g. some integrated circuits, transistor, diodes, ..)

- adequate design to enforce shielding of such equipment

- replacement of such equipment

Final risk => likel.: min(10-4)/ sev.: catast.(loss of mission) => **low risk**

Resid. / add. risk: ..*no*..

*Remark: * Single Event Effect*

*** Total non-ionizing dose (irradiation by heavy ion/ protons)*

Overall Cost / Schedule Risks

OCI -> cost risk – overall cost overrun

Risk scenario: 1050bill EUR* budget excided by …??..EUR

..due to … ?? or several risk mitigations

Initial risk => likel.: …/ sev.: …(…) => **… risk**

Mitigation: **…** -> decrease of sev./ likel.

Final risk => likel.: …/ sev.: …(…) => **low risk**

*Remark: * all inclusive for LISA (Realization, launcher, mission& science operation)*

other project costs:

- Hubble telescope app. 4.5bill USD (build, launched, comm. 1993)

- James Webb space telescope estimated costs 8/ 8.8bill USD (blc/LCC(5a))

- European ELT estimated costs 1.5bill EUR (bc)

OSI -> schedule risk – ..*see DVI13*.. (delay due to development of low TRL equipment)

Risk scenario: delay by ..??.. due to …

Initial risk => likel.: …/ sev.: …(…) => **… risk**

Mitigation: **…** -> decrease of sev./ likel.

Final risk => likel.: …/ sev.: …(…) => **low risk**

Preliminary risk assessment

Severity	A	B	C	D	E
5 (catastr.)		LI-s[CP], LII-s[CP,EP]			
		MIIa-dt	DIV-dt, DIXa-dt, LIII/IV-dt, CI-dt, MIII-dt	DV-dt	(DVII1[CP]-pr), PI-pr
4 (critical)				(DVII3-sh/dt), MI-dp	
3 (major)				DVI-dp, DIXb-dt	
2 (signif.)			MIIb-dp		
1 (minor)					
0	no risk (elimin.)				
	A (min.) < | ≤ 1/10000 (10E-4) .. almost never	B (low) ≤ 1/1000 (10E-3) .. seldom	C	D	E

pr - programmatic/ dt - dep.(tech.) / dp - dep.(perform.)/ p - protection / s -safety/ sh - s

	Design & realisation
DIV	mechanism
DV	mission reliability
DVI	mission availability
DVII1	CP (cold gas)
DVII2	EP (cold gas)
DVII3	EP+ (EMP)
DIXa	antenna mechanic (SPF)
DIXb	antenna mechanic (micro vibration)
	Launch preparation & Launch & IOT + SDM
LI	toxic propulsion
LIII	S/C separation
LIV	de-tumbing/ collision
	Cruise
CI	Transfer phase anomaly
	Mission performance + PP
MI	constellation acquisition
MIIa	Micro meteoroids (loss of mission)
MIIb	Micro meteoroids (availability)
MIII	Radiation
	Overall Cost/ Schedule + Programmatic
CoI	Cost overrun
PrI	Consortium

Risk assessment shows that the LISA mission has an acceptable risk level

Severity	A	B	C	D	E
5 (catastr.)	LI-s[CP], LII-s[CP,EP])				
	CI-dt, MIIa[CP,EP], MIII-dp	DIV-dt, DIXa-dt, LIII/IV-dt			(DVII1[CP]-pr)
4 (critical)		MI-dp	DV-dt		
3 (major)			(DVII3[EP+]-sh/dt)		
2 (signif.)			DVI-dp, DIXb-dt, MII-dp		PI-pr
1 (minor)					
0	no risk.. (DVII2[EP], ~~DVII1~~)				
	A (min.) < | ≤ 1/10000 (10E-4) .. almost never	B (low) ≤ 1/1000 (10E-3) .. seldom	C (medi.) ≤ 1/100 (10E-2).. sometimes	D (high) ≤ 1/10 (10E-1) .. frequently	E (max.) ≤ 1.. certain
					Likelihood

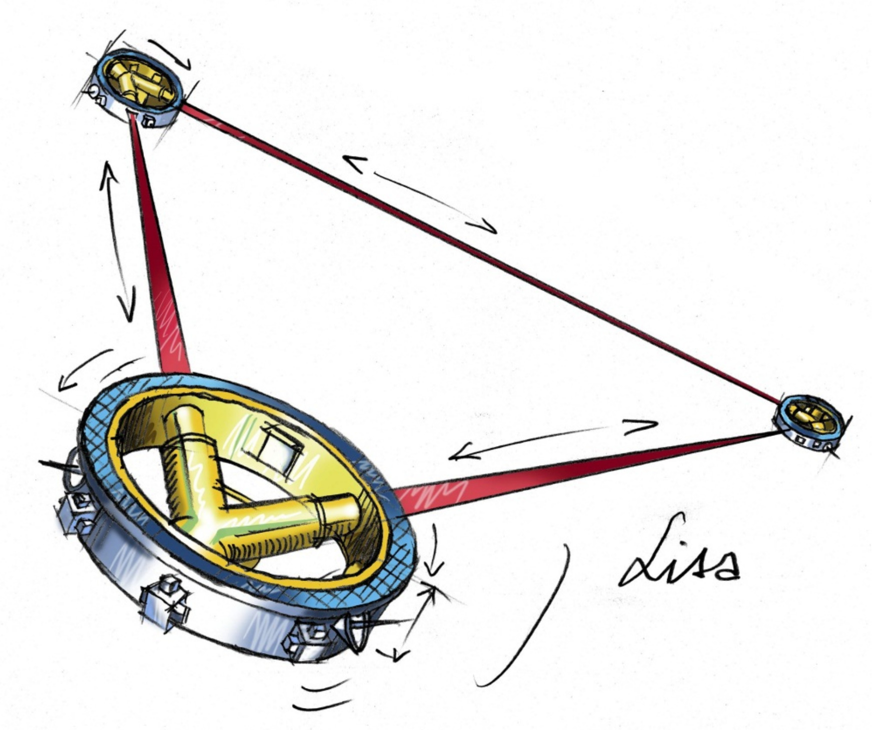
pr - programmatic/ dt - dep.(tech.) / dp - dep.(perform.)/ p - protection / s -safety/ sh - schedule/ c - cost

LISA

Programmatics/AIV

Internal Final Presentation
ESTEC, 05-05-2017

Prepared by the CDF* Team



(*) ESTEC Concurrent Design Facility

Requirements and Design Drivers

Req. ID	Requirement
CONS-020	The mission shall be launched before 2034 TBC
CONS-030	TRL 6 shall be achieved by all elements at the end of phase B1 (2024)
CONS-040	The mission shall be compatible with a launch on Ariane 6.4 from Kourou
CONS-050	Back up launcher shall be identified (not restricted to European launchers)
MIS-010	The mission shall consist of three identical spacecraft
MIS-030	The mission shall be designed for a lifetime of 6.5 years
MIS-040	The mission should be designed for an orbit lifetime of 10 TBC years
SYS-020	No interface shall require the presence of more than one instrument, i.e. no routing of interfaces and no common use
SYS-030	The accommodation of the payload shall be designed such that any instrument can be tested individually and removed or added to the spacecraft for these tests.
SYS-050	Mechanisms operation shall not disturb the science data collection (implication: micro-vibration characterisation)
PAY-010	The payload shall be identical in all three spacecraft
PAY-020	The payload shall consist of: Telescope, Laser and Science Instrument Assembly with Optical Bench, Gravity Reference Sensor, Phase Meter, Diagnostics Package and Data Processing unit
PAY-030	The total mass of the payload shall be lower than 360 kg, including margin
PAY-060	The overall dimensions of the payload shall be under 2150,1500,900 mm
PAY-070	The payload shall be thermally isolated for the service module

Assumptions (1/2)

- TRL 6 shall be achieved before the start of the Implementation Phase.
- The 3 S/C shall be launched together.
- Modular S/C configuration (PM, SVM, PLM) similar to Lisa Pathfinder is preferred but not a requirement.
- While truly modular configuration might not be feasible, it is assumed, that the **payload (subsystem) integration will be done by the prime contractor** while the S/C with all other subsystems might be integrated by a separate contractor.
- The **Main Core Assembly** will be one module including telescope, 2 instruments and optical benches etc., which is part of the payload subsystem and will be integrated on or removed from the spacecraft in one piece.
- All interfaces (mechanical, thermal, data handling, power, etc.) of the **Main Core Assembly** must be very well defined to **allow as much as possible stand-alone verification** with the use of MGSE and EGSE/simulators.

Assumptions (2/2)

- Thermal stability of payload equipment, Main Core Assembly, instrument, optical benches etc. is important and must be controlled.
- Gravity balancing is very important, taking also into account telescope and antenna pointing and consumables use.
- Micro-vibration characterization is very important. Measurements cannot done over the complete frequency range and need to be extrapolated (frequency range 0.1 mHz-1Hz).
- Solar Arrays are fixed mounted (for above reasons).

Trade offs

- Propulsion system configuration
  - Heritage (CP module + cold gas system)
  - **Electric Propulsion, EP (integrated electric propulsion + cold gas system)**
  - Electric Propulsion +, EP+ (integrated electric propulsion + miniRIT/FEEPs)
- Launch configuration
  - Cylindrical satellites on top of each other
  - **Trapezoidal satellites next to each other (base mounted similar to SWARM satellites)**
- Telescope (3 options):
  - mechanical movement of the telescope – favourite from testing point of view
  - optical movement of the telescope
  - hybrid (mechanical + optical) movement
- Laser (low_power-amplifier-modulation versus **low_power–modulation-amplifier**)
- Payload subsystem optimization (number of individual units)

Product Tree

- The product tree taken from the OCDT model has been evaluated. It summarizes the equipment and instruments per subsystem and the associated Technology Readiness Level (TRL), if available, but without identifying subassemblies.
- The number of units in each subsystem, and their mass, can be used to estimate the integration effort.
- The TRL status allows to estimate the time needed before the required TRL can be reached that allows the start of the implementation phase of the project.
- For any item with a TRL below 6 development plan should identify the resources and time needed to reach TRL 6.
- The product tree clarifies also the difference between options of the OCDT model.
- *The complete OCDT product trees can bee seen in file “prog-experimental-LISA 20170502.xlsx” in the session 13 presentation directory and in the report.*

TRL scale

TRL	ISO Definition	Associated Model
1	Basic principles observed and reported	Not applicable
2	Technology concept and/or application formulated	Synoptic, block diagram
3	Analytical and experimental critical function and/or characteristic proof-of concept	Proof of concept model, such as mathematical models, simulations, supported by experimental data or characteristics
4	Component and/or breadboard validation in laboratory environment	Breadboard of the element (integration of functionally representative breadboard).
5	Component and/or breadboard critical function verification in a relevant environment	Breadboard, also referred to as sub-scaled EM for the critical functions
6	Model demonstrating the critical functions of the element in a relevant environment	One or more of the following: Full scale EM(s), SM, STM, TM, DM(s), representative for critical functions in form fit and function.
7	Model demonstrating the element performance for the operational environment	QM
8	Actual system completed and "flight qualified" through test and demonstration	FM acceptance tested, integrated in the final system
9	Actual system completed and accepted for flight ("flight qualified")	FM, flight proven

Source: ECSS-E-HB-11A, 1 March 2017, Technology readiness level (TRL) guidelines

PLM product tree from OCDT

The TRL for several technologies for the payload are identified at low values (TRL 3 and 4).

The OCDT product trees show for the s/c without payload only a few items at TRL 5, but none lower:

- Option _CP 0
- Option_EP 1 (mechanism)
- Option_EP_plus 1 (mech) +5 (eprop)

No TRL is identified for structural and thermal parts because these items are build according to specifications and no new developments are needed for them.

Owner	Name	n_items	TRL
INS	Acquisition CCD Electronics	1	4
INS	Caging Control Unit	2	7
INS	Charge Management System	2	4
INS	Diagnostics	1	7
INS	Electronics_GRS_SAU	2	7
INS	Gravitational Reference Sensor	2	7
INS	GRS_FEE_PCU_HARN	1	7
INS	Laser Control Unit	4	6
INS	Laser Frequency Stabilisation	2	9
INS	Laser_option1	4	6
INS	Optical Bench	2	6
INS	Payload Harness	1	-
INS	Payload Processor Unit	1	4
INS	Payload Structure	1	-
INS	Phasemeter	2	6
INS	Telescope_LISA_option1	2	3

OCDT payload product tree items

Technology readiness

Low TRL items (incomplete, source: LISA Proposal January 2017)

Subsystem	Technology	TRL
GRS	UV source : LEDs	4
DFACS	Colloidal Micropopulsion	5 (feed system)
DFACS	miniRIT & HEMP Micropropulsion	4 / 3
Laser	Fibre Amplifier TESAT	5
Laser	Fibre Amplifier	4
Laser	Master Oscillator -ELC	4
Optical Bench	Fibre injectors	5
Optical Bench	Manufacturing	4
Optical Bench	Photoreceivers - US	4 / 5
Optical Bench	Photoreceivers - DLR/AdlershofInterferometric phase reference	4
Optical Bench	Interferometric phase refernce	4
Optical Bench	Pointing mechanism	4
Telescope	Optomechanical Stability	4
Telescope	Optical Truss	4
Telescope	Pointing - Articulated Telescope	4
Telescope	Pointing - In field Guiding	3
Phase Measurement System Technologies	Complete functionality	4
Phase Measurement System Technologies	LISA - specific functions	4
Diagnostics	Diagnositc Items	4

Technology developments (1/2)

Reference	Activity Title	Prog	Status	Duration	Start
C207-009PW	GRS Front End Electronics characterization for LISA	CTP			
C207-010EE	Compact low noise magnetic gradiometer	CTP			
C207-011PW	Charge Management System for LISA	CTP			
C207-012PW	Opto-mechanical stability characterization for LISA	CTP			
C207-013PW	Metrology system for LISA	CTP			
C216-113PW	Optical Bench Development for LISA	CTP			
C216-137FM	Optical Bench Manufacturing Industrialisation Study	CTP	Running	12	01-Apr-17
C216-138FM	Metrology Telescope Design for a Gravitational Wave Observatory	CTP	Running	12	28-Nov-16
C216-138FM (B)	Metrology Telescope Design for a Gravitational Wave Observatory	CTP	Running	12	28-Nov-16
C217-030MM	High-power laser system for eLISA	CTP			
C217-045FM	Phase Reference Distribution for Laser Interferometry	CTP	Running	16	Apr-17
C217-046FM	Gravitational Wave Observatory Metrology Laser	CTP	Running	36	Apr-17
C217-046FM-P1	Gravitational Wave Observatory Metrology Laser	CTP	Running	12	
T205-033EC	Assessment and Preliminary Prototyping of a Drag Free Control System for the L3 Gravity Wave Observatory	TRP	In Preparation	12	On hold
T217-064M	Fine Structure of Laser Radiation in the Far Field	TRP	In Preparation	12	On hold
T219-001MP	Electric Micropropulsion System for a Gravitational Wave Observatory Mission	TRP			

Technology developments (2/2)

- A number of urgent developments are
  - Ongoing
  - In preparation
- This list does not cover all technologies with low TRL
  - e.g. the proposal from January 2017 lists in addition Phase Measurement System Technologies
  - The GOAT Final report Rev. 1, May 2016 goes into more detail so it is not obvious that all mentioned items are covered
  - The GOAT Final Report lists additional system issues
- **It will be necessary to consolidate the list and identify the additional technology developments which are needed to be achieved before the Implementation Phase**

Model Philosophy

- Instruments - it is proposed to build at least 8 models (Danzman Proposal):
  - STM, EM, PFM and 5 x FM,
    plus spare kits and possibly one FM spare

Consequently, at higher level following models are proposed:

- Payload Module (Main Core Assembly)
  - STM, EM (using instrument STM and EM), PFM, 2 x FM
- Spacecraft
  - STM, EFM, PFM, 2 x FM
- Simulators/EGSE are needed to
  - Test the instruments stand-alone,
  - To test the Main Core Assembly alone
  - To test the S/C in absence of the Main Core Assembly
- Equipment
  - Standard approach depending on heritage and previous qualification

Integration and verification approach

- **Instrument EM** performance and Laser mounting technology shall be verified before the start of the Implementation Phase.
- An **S/C STM** is foreseen for early qualification of the structure and the thermal model in Phase C. It will make use of a **payload STM**. The payload STM will use an instrument STM, but it might need a second instrument STM. Depending on the build standard and availability the **Main Core Assembly EM** could be used instead of a **Main Core Assembly STM**.
- Instrument EM and STM shall be used to build an **EM** of the **Main Core Assembly** early in the Implementation Phase for environmental tests. EM's of other payload equipment and early versions of S/C simulator might be needed for that purpose. These tests shall be completed before **PFM** procurement.
- Payload EM and eventually **S/C EFM** will be used for functional verification, software and unit testing throughout the project.

Test matrix at Spacecraft level

Test Description	STM	PFM	FM 2 + FM 3
Mech. Interface	R, T	R, T	R, T
Mass Property	A, T	A,T	A,T
Electrical Performance		T	T
Functional Test		T	T
Propulsion Test		T	T
Deployment Test (Antenna, Telescope)	A, T	A, T	A, T
Telecommunication Link		T	T
Alignment	A, T	A, T	A, T
Strength / Load	A, T		
Shock / Separation	T		
Sine Vibration	A, T	T	T
Modal Survey (base excitation)	A, T	T	
Acoustic	T	T	T
Outgassing			
Thermal Balance	A, T	T	
Thermal Vacuum		T	T
Micro Vibration		A, T (tbd)	
Grounding / Bonding		R, T	R, T
Radiation Testing			
EMC Conductive Emissions and Suceptibility		T	T
EMC Radiated Emissions and Suceptibility		T	T
DC Magnetic Testing		T	T
RF Testing			
Thermal/Mechanical Stability	T (tbd)		

Abbreviations:
I: Inspection
A: Analysis
R: Review of design
T: Test

STM: Structural Thermal Model
EM: Engineering Model
EFM: Electrical and Functional Model
PFM: Protoflight Model
FM: Flight Model

Instrument and module level tests

- The verification approach for instrument models shall be similar to other equipment
  - STM and EM have to purpose to acquire early verification results
  - PFM and FMs will undergo qualification and acceptance tests respectively
- The verification at module level (Main Core Assembly):
  - The EM shall be geometrically and structurally representative allowing potentially for environmental testing together with the s/c STM
  - As baseline it shall undergo environmental tests separately (vibration and thermal vacuum) to confirm performance, stiffness, load capability and thermal stability.
  - S/c STM testing and Main Core Assembly EM environmental testing shall be completed before s/c and payload PFM procurement.

Schedule – key dates (tentatively)

Event	From	To	Status
L3 Proposal Submission	2016-OCT	2017-JAN	*Done*
L3 Proposal Evaluation	2017-JAN	2017-JUN	*Running*
L3 CDF	2017-MAR	2017-MAY	*Running*
L3 Mission Selection	2017-JUN	2017-JUN	*June SPC (June 21)*
Phase 0 for national contributions	2017-JUL	2017-NOV	
Mission Definition Review (MDR)	2017-NOV	2017-DEC	
Phase A (mission & instruments)	2018-JAN	2020-JAN	Feasibility
Mission Consolidation Review (MCR)	2018-OCT	2018-NOV	*To be confirmed*
Preliminary Requirements Review	2019-NOV	2020-JAN	
Bridging Phase	2020-FEB	2022 FEB	*If needed*
Phase B1	2022-FEB	2024-FEB	Requirements consolidation
Adoption	2024 MAR		Depending on programmatics
Implementation (Phase B2/C/D)	2024	2033	
Launch	**2034**		
Transfer & Commissioning	2034	2036	~18 months + 9 months
Operations	2036	2040	4 years
Extension (TBD)	2040	2044	

Summary and conclusions 1 / 3

- Only a few items with a TRL lower than 6 have been identified for s/c items.
- For the payload between 10 and 20 items are identified with such low TRL.
- A number of payload related developments are identified of which some have been started already.
- The list of necessary pre-development activities to be completed before the start of the Project Implementation Phase needs to be consolidated, budgeted, planned and implemented.
- A baseline model philosophy and integration and verification approach has been presented together with a preliminary s/c level test matrix.
- A schedule has been proposed showing:
  - Start of the Implementation Phase begin June 2024
  - Launch 9 years and 5 month after k.o. of the Implementation Phase.

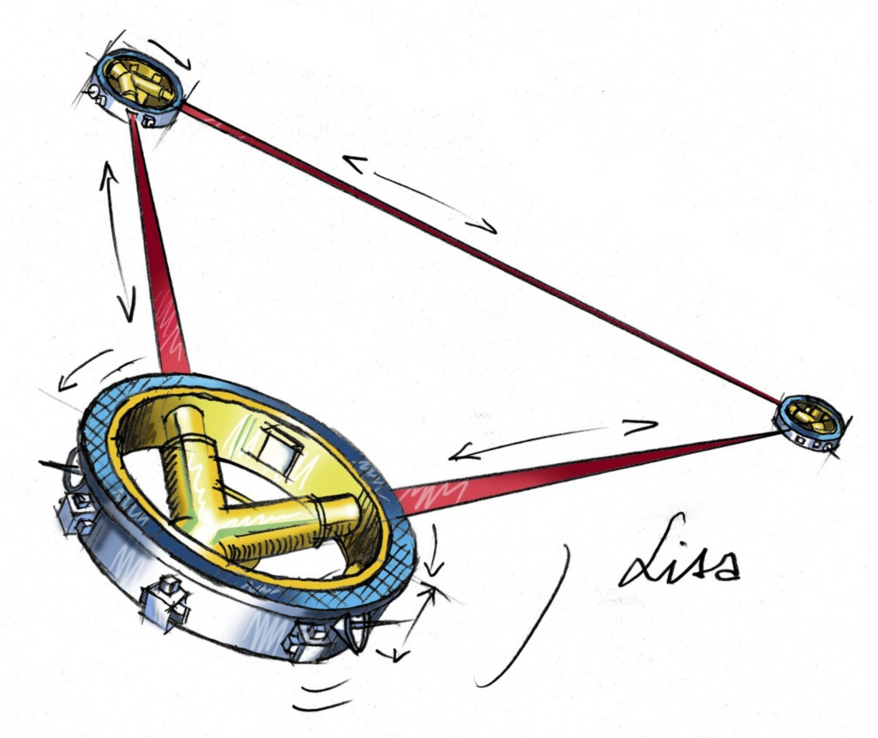
Summary and conclusions 2 / 3

- The schedule is generic and does not take possible differences of the various options into account. Further optimization will be possible.
- Critical is the implementation of all development activities necessary before the Implementation Phase, in particular:
  - The instrument related developments, manufacturing and verification to be funded and organized by the PI and supporting states;
  - The early development of Engineering Models
- Instrument procurement duration is expected to be driven by the manufacturing capabilities for the optical bench. At least 7 units (including EM, but without spares) are needed. With production duration expected to be up to 6 month per unit, the delivery of the flight models should start at least 5 years before launch.

Summary and conclusions 3 / 3

- The necessary development and verification of a spacecraft dispenser is not shown in the schedule, but certainly feasible in the time needed for the design, development and verification of the 3 spacecraft.
- The long duration of the project requires that storage and equipment lifetime need to be taken into account.
- A reduction of the overall schedule appears feasible depending on the status of the payload development:
  - Phases A, Bridging Phase and B1 are rather long, each 2 years, while typical values are about 12 month, 8 month and 14 month
  - The Main Core Assembly EM could be tested together with the s/c STM
  - MAR and MSRR are basically a duplication of effort leading to a Phase B2 duration of 19 month instead of a typical 15 month
- Taking these points into account a launch date advancement by 3 years is imaginable.

LISA



Conclusions

Internal Final Presentation
ESTEC, 05-05-2017

Prepared by the CDF* Team

(*) ESTEC Concurrent Design Facility

CONCLUSIONS

- Definition of the LISA mission has been carried out for three main system options at sub system level, EP options offering the best compromise
- Mission has been sized for 10 years of science operations
- Baseline option has been defined
- Mission compatible with baseline launcher (except for CP option), back up launcher identified
- Payload definition further detailed (architecture, redundancy, budgets)
- Operational scheme has been defined
- Risk assessment for the mission has been carried out
- Programmatic assessment and program schedule has been defined
- Cost assessment for the different system options has been provided

OPEN POINTS

- Consolidation of payload definition and trade offs (Phase 0/A)
- System Performance evaluation

- Define limits for CP option, what will be needed to put it back in the picture
- Electro magnetic compatibility of EP elements
- Gravity balancing
- Technology assessment of different electric micro propulsion systems
- Assessment of micro vibrations due to antenna pointing mechanism
- Propulsion system optimization (AOCS)
- Data compression investigation
- Phase array antenna option
- Spacecraft dispenser
- Micrometeorites evaluation consolidation
- Further investigation into orbit maintenance