MarcoPolo-R External Final Presentation of the CDF study

Science and Robotic Exploration Directorate, Advanced Studies and Technology Preparation Division

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ESTEC
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European Space Agency

Requirements

- Target asteroid: 175706 / 1996 FG3
- Launch between 2020 and 2024
  - ✓ Similar baseline and backup scenarii are desirable
- Duration < 10 years, ideally much less
- Mission design consistent with low-cost solution

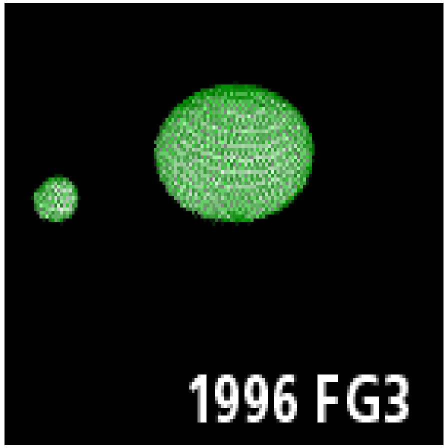
Target Overview

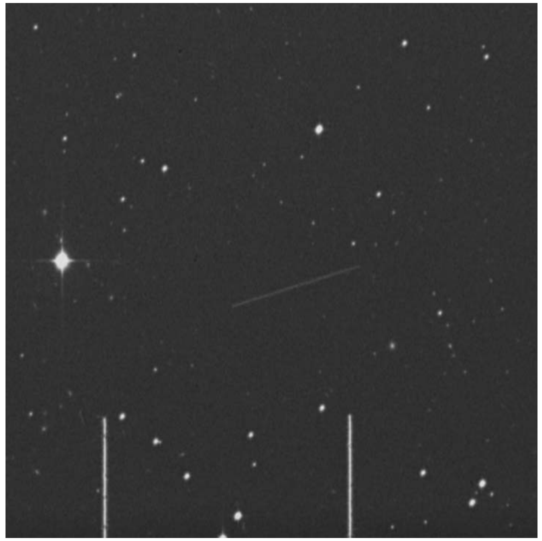
- Name: 1996 FG3
- Class: C-type binary
- Orbit: 0.69 to 1.42 AU from Sun
- Inclination: 1.99 deg
- Eccentricity: 0.3498
- Orbital period: 1.08 years

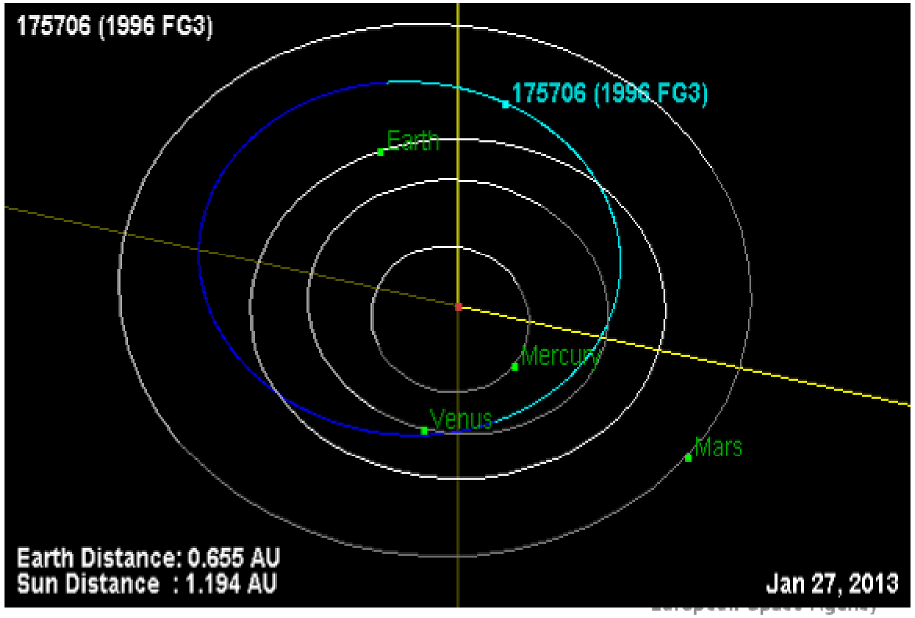
- Primary
  - ✓ Diameter: 1.4±0.2 km
  - ✓ Mass: 2.1 E12 kg
  - ✓ Geometric albedo: 0.035
  - ✓ Spin period: 3.595±0.002 hrs
  - ✓ Density: 1.4±0.3 g.cm^-3
  - ✓ Taxonomic type: C

- Secondary
  - ✓ Secondary to primary diameter ratio: 0.28±0.02
  - ✓ Orbital semimajor axis: 3.1±0.5 km
  - ✓ Orbital eccentricity: 0.1±0.1
  - ✓ Orbital period around primary: 16.14±0.01 hrs

- Surface compressive strength: ≤2 Mpa







Assumptions

- Launch with Soyuz-2.1b/Fregat M from CSG, Direct escape
- Launch between 2020 and 2024
    - ✓ Find similar baseline and backup scenarii
- Chemical and electric propulsion systems (SEP= Solar Electric Propulsion) were traded → use off-the-shelf solutions

Analysis of CP Transfers

1. Direct escape with CP found to lead to excessive transfer durations and insufficient dry mass for launch within regarded period → discarded
2. GTO launch with CP:
   a. Transfer durations 9 – 11 years in 2022-2024 launch timeframe
   b. Dry mass < 1100 kg (minus all margins, navigation, attitude control propellant)

Direct Escape with SEP

- Allows reduced mission durations
  a. High Delta-v budget but typical of SEP transfers
  b. Reduced number of swingbys allows shorter transfers
- Assumptions:
  a. S/C based on existing platform
  b. Solar array size such that power avl. for SEP 3 kW@1AU
  c. Target dry mass: 1210 kg
  d. T6 GIT from Bepi Colombo
  e. This led to missions consistent with launcher capability and dry mass target

Summary of Trajectory Characteristics

Case	Sep01b	Sep01c	Sep04de	Sep04d
Launch	2021/4/11	2021/4/11	2022/2/4	2023/2/4
Venus s/b	2	2	1	1
Earth s/b	1	0	1	0
Ast. Arrival	2025/3/29	2024/9/12	2027/5/29	2027/5/29
Stay duration [d]	265	180	190	190
Earth arrival	2029/1/8	2027/12/14	2030/3/8	2030/3/8
Total duration [y]	7.8	6.7	8.1	7.1
Launch mass [kg]	1584	1584	1696	1696
Xe mass w/10% [kg]	208	241	267	267
SEP delta-v [m/s]	4955	5899	6167	6167
Hy.mass [kg] (nav+att)	96+15	87+15	87+15	77+15
Dry mass [kg]	1265	1241	1327	1337

May require updates

Asteroid Approach

- Due to the use of SEP, the asteroid approach is slow and tangential
- Example Sep04d:
  - ✓ Arrival – 1 month: Distance ~30,000 km
  - ✓ Arrival – 6 months: Distance ~ 2E6 km
- Approach navigation performed entirely with SEP
  - ✓ No hydrazine consumed
- First characterization of target can be performed during arrival
  - ✓ This leaves more time for other science activities while in orbit around primary

In-Orbit Phase

- Numerical analysis of orbit around primary body was performed
  - Primary modeled as ellipsoid with near-spherical shape
  - Gravitational perturbations of SRP, solar gravity and secondary body taken into account
  - For low orbit, effect of second body is negligible
    - Avoid coplanar and commensurate orbits

Implications of Low, Controlled Orbit

- Orbit assumed <u>for analysis purposes</u> (not baseline!!):
  - ✓ As low as safely possible (~200 m above mean radius?)
    - Orbital period ~4 h
    - Orbital velocity ~40 cm/s
  - ✓ 90 deg inclination wrt asteroid's heliocentric orbit plane
  - ✓ Node placed such that 9/21 or 15/3 SSO is obtained
  - ✓ Radius, inclination and node controlled by small RCS manoeuvres
    - Cost < 2 m/s/month
    - Individual manoeuvres small (x mm/s), but accuracy is important
    - Provide thrusters that are so small that manoeuvres are non-impulsive and steady-state thrust is reached

To be consolidated

System Trade-Offs

➢ System level Trade-Offs

a. Transfer - Electrical Propulsion vs. Chemical Propulsion

b. Custom platform vs. “standard” platform reuse

c. Sampling approach

System Trade-Offs
- EP vs. CP -

➢ Performance analysis results:

Scenarios					
	Total Dv (m/s)	Launch mass (kg)	Total wet mass (w/o adapter) (kg)	Dry mass (kg)	Total Propellant (kg)
EP only 2021	4860.45	**1615.00**	1505.00	**554.49**	950.51
EP only 2023	5843.25	**1715.70**	1605.70	**638.07**	967.63
EP + CPSK 2021	4860.45	**1615.00**	1505.00	**1142.46**	362.54
EP + CPSK 2023	5843.25	**1715.70**	1605.70	**1176.08**	429.62
CP Mono-Prop 2021	3106.95	**2916.50**	2806.50	**618.39**	2188.11
CP Mono-Prop 2023	3455.55	**2916.50**	2806.50	**526.15**	2280.35
CP Mono-Prop 2024	3400.95	**2916.50**	2806.50	**539.63**	2266.87
CP Bi-Prop 2021	3106.95	**2916.50**	2806.50	**978.44**	1828.06
CP Bi-Prop 2023	3455.55	**2916.50**	2806.50	**875.00**	1931.50
CP Bi-Prop 2024	3400.95	**2916.50**	2806.50	**872.83**	1933.67

Based on the previous estimates, the following can be concluded:

- EP Only: Discarded due to both limited performance and controllability issues related to the maximum thrust achievable by EP Resistojets (<1N)
- CP Mono-Prop: Discarded due to its very low dry mass performance.

- EP+CPSK performances are high enough for such a mission
- CP Bi-Prop from GTO is barely feasible

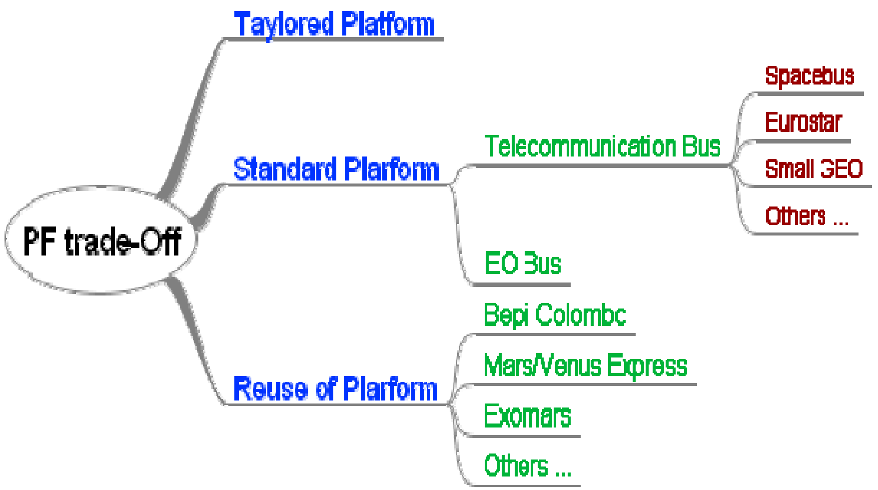
System Trade-Offs
- EP vs. CP -

- Selected Option: **EP+CPSK**
- The following trade table presents the selection's rationale using:
  - ✓ Original (trade) keys as defined previously
  - ✓ "Derived keys" derived from the previous "Performance" analysis and propulsion architectures
  - ✓ The "Flexibility" fields add additional consideration for such trade
    - "to Dry mass" translates the design sensitivity to mass
    - "un-optimized design" translates the potential use of PF not tailor made (to link with std vs. tailored PF trade)
    - "departure date change" translate potential MA flexibility (e.g. contingency case)

Trade Keys	Propulsion system options			
	Electrical Propulsion		Chemical Propulsion	
	Value	Rating	Value	Rating
Original Keys				
Re-entry Speed	12.69	0	12 to 13.3	0
System Cost		0		0
Risk		0		0
Mission Duration [yr]	7.1 to 7.6	+	7.8 to 10	0
Stay time @ Asteroid [days]	244 to 263	++	192 to 204	0
Derived Keys				
Propellant to Dry mass ratio	0.3 to 0.36	+++	2 to 2.3	- -
Quantity of propulsion systems	2	- -	1	++
Flexibility to ...				
Dry mass increase		++		-
Un-optimized design		++		-
Return Date		+		- - -

System Trade-Offs
- Custom vs. standard platform -

- The following options where considered:



- In the attempt to reduce cost, the **reuse of a SmallGEO platform** has been selected. This is supported by the following aspects:
  a. ESA's detailed knowledge of the platform
  b. SmallGEO has virtually most of the required equipments by default (e.g. sensors, ...)
- Note: SmallGEO is to be considered more as a benchmark than a formal selection. <u>If feasible, the reuse of any other small telecom platform could be envisaged</u>

System Trade-Offs
- Sampling Approach -

- Selected Option: **Touch & Go**

- Selection rationale:
  a. "Full landing with long stay" & "Full landing with a short stay"
     Both full landing option have been discarded as they have been judged to be the most expensive in terms of development and impose additional requirement on the GNC (need for high landing accuracy)
  b. "Hover & Go"
     This option has been discarded due to its GNC implication
  c. "Touch & Go"
     Selected as a baseline primarily because of it's simplicity and therefore "lower" cost/GNC implications
     Note: The team is fully aware of the Marco Polo internal review board report not retaining this architecture as a higher risk option.

Overall Mission Architecture

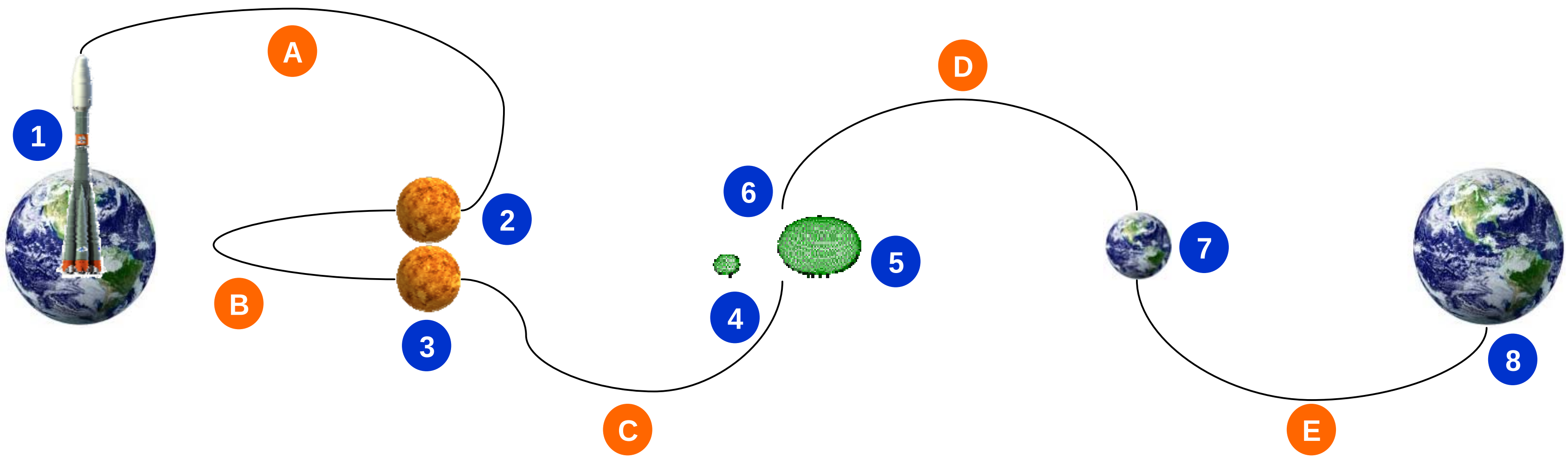
- Based on the EP transfer selection and data from MA, 2 mission scenarios have been selected:

  a. Baseline: Transfer option 01b (launch in 2021)
  b. Backup: Transfer option 04d (launch in 2023, also possible one year earlier in 2022, 1 added year)

Note: Might be updated based on MA refinement

Overall Mission Architecture

1. Baseline Architecture (Option 1b)

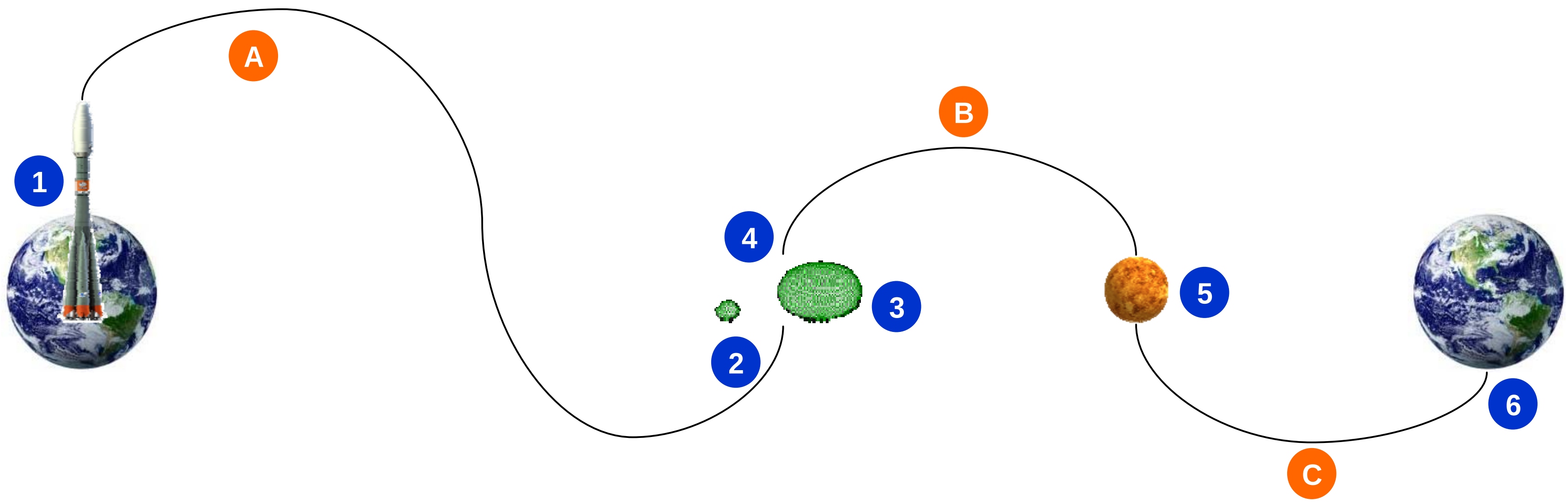


1 Soyuz ST launch from Kourou (FR)
2 First Venus Swing-by
3 Second Venus Swing-by
4 Arrival to 1996 FG3
5 Observation & sample collection
6 S/C Departure (Return to Earth)
7 Earth Swing-by
8 ERC return to Earth

EP thrust and coast arcs

Overall Mission Architecture

2. Backup Architecture (Option 4d)



1. Soyuz ST launch from Kourou (FR)
2. Arrival to 1996 FG3
3. Observation & sample collection
4. S/C Departure (Return to Earth)
5. Earth Swing-by
6. ERC return to Earth

EP thrust and coast arcs

System Baseline
- Overview -

Mission Description		
	Baseline	**Backup**
Launch	Soyuz ST w.Fregat MT from Kourou	
Launch Date	11/04/2021	04/02/2023
Launch injection	Direct escape	
Outbound Transfer	EP & swing-bys	
	2 Venus Swing-by	No swing-by
Inbound Transfer	EP & swing-bys	
	Earth swing-by	Venus swing-by
Overall System Characteristics		
Mass (inc. Margin)	Dry Mass: 1170 kg	
	P/L Mass: 25 kg	
	Max EP Propellant Mass: 254 kg of Xenon	
	Max CP Propellant Mass: 172 kg of Hydrazine	
S/C Main Components	- Main S/C with sampling system and P/L - ERC	
S/C Overall Dimensions	S/C body dimensions: 1.9 x 2 x 3 m^3 (TBC)	
	S/C wingspan: 12 m (TBC)	

May require updates

Sized for the worst cases (highest delta-V, lowest launch mass, highest distances to Earth/Sun, highest re-entry velocity, etc.)

System Baseline
- Main Spacecraft -

esa

Main S/C Description		
Structure	S/C body similar to SmallGEO's	
GNC	Sensors	2 x IMU
		1 x Star tracker (redundant)
		2 x Coarse sun sensor
		1 x Radar Altimeter
		2 x WAC
	Actuator	4 x Reaction wheels
		8 x 1N monopropellant thrusters
		4 x 20N monopropellant thrusters
Robotics & Mechanisms	Sampling	Single arm for sampling and sample transfer
		3 sample capability
		Designed for 5 cm/s horizontal speed sampling
		Designed for 15 cm/s vertical speed sampling
	Support	~~MGA pointing~~ Fixed ERC hold down, ejection and spin SA HDR, deployment and orientation

System Baseline
- Main Spacecraft (cont.)

Main S/C Description (cont.)		
Propulsion	Monopropellant system (Hydrazine)	
	Electric propulsion system w. T6 thruster (+ redundant)	
Power	SA	2 x Deployable single panel wings
		Solar cells: 28% 3J GaAs
		___kW (EOL @ 1AU)
		___kW (EOL @ _AU)
	Battery	1 x Lithium Ion
		Capacity: ___Ah
Communications	X-Band system	
	1 x Fixed HGA	
	1 x Fixed MGA	
	2 x Fixed LGA for 4π coverage	
Thermal	MLI, heating lines, Black Paint, SSM ...	
DHS	Bus	TBC (CAN or MIL-STD-1553B)
	OBC	SPARC family (E.g. LEON series)
	Memory	TBC (512 Gbit SDRAM or 1024Gbit Flash)

System Baseline - Main Spacecraft -

Spacecraft

Dry mass contributions	Without Margin	Margin %	Margin kg	Total kg	% of Dry
Structure	265.00 kg	20.00	53.00	318.00	34.31
Thermal Control	41.25 kg	5.00	2.06	43.31	4.67
Mechanisms	49.00 kg	11.24	5.51	54.51	5.88
Communications	32.00 kg	10.31	3.30	35.30	3.81
Data Handling	29.00 kg	5.00	1.45	30.45	3.29
GNC	33.10 kg	8.17	2.71	35.81	3.86
Propulsion	148.79 kg	6.12	9.11	157.90	17.03
Power	143.00 kg	0.00	0.00	143.00	15.43
Harness	70.00 kg	20.00	14.00	84.00	9.06
Instruments	20.52 kg	20.00	4.10	24.62	2.66
Total Dry(excl.adapter and ERC)	831.66			926.89	kg
System margin (excl.adapter)		20.00 %		185.38	kg
Total Dry with margin (excl.adapter and ERC)				1112.27	kg
ERC	~~40.00~~ kg 54 kg !!	0.00	0.00	~~40.00~~ 54 kg !!	
Total Dry with margin incl. ERC (excl.adapter)				1152.27	kg
Propellant - Xenon	254.00 kg	0.00	0.00	254.00	
Propellant - Hydrazine	172.00 kg	0.00	0.00	172.00	
Adapter mass (including sep. mech.), kg	110.00 kg			110.00	
Total wet mass (excl.adapter)				1578.27	kg
Launch mass (including adapter)				1688.27	kg

Previous Marco Polo study

MP Reference AST	
143.70	kg
27.30	kg
49.30	kg
28.70	kg
23.30	kg
43.10	kg
119.70	kg
87.90	kg
50.00	kg
22.90	kg
595.90	kg
119.18	kg
715.08	kg
0.00	kg
646.00	kg
57.00	kg

System Baseline
- Earth Re-entry Capsule -

Earth Re-entry Capsule Description		
Trajectory	Landing location	Woomera, Australia
	Entry velocity	12.69 m/s
	FPA	11 deg
Shape	Main Diameter	900 mm
	Nose Diameter	___mm
TPS	DEAM (European PICA-like), TRL4-5	
	Heat load	Max: ¬370MJ/m^2
	Heat Flux	Max: ¬15.3 MW/m^2
Thermal	TPS/Structure insulation by Aerogel	
EDLS	None (no parachute)	
Structure	Load bearing	
Mechanisms	Sample container	
	Back shell opening/closing system	
GNC	None (passive re-entry)	
Communications	Recovery Beacon	
Power	Battery	For 2 days (TBC) Beacon operation
		___ x ___
		Capacity: __Ah
Propulsion	None	
DHS	None	

System Baseline Composite

Baseline Mission

1. S/C & ERC as presented previously (1169kg dry mass)
2. Propellant mass based on the Δv budget presented previously:
   a. Hydrazine: ~104kg
   b. Xenon: ~208kg

   Updated MA values

3. Resulting wet mass: 1481kg *(ex. 110kg LVA)*
4. Option Max launch Mass: 1584kg (ex. LVA)
5. **Resulting launch margin: 6%**

Backup Mission

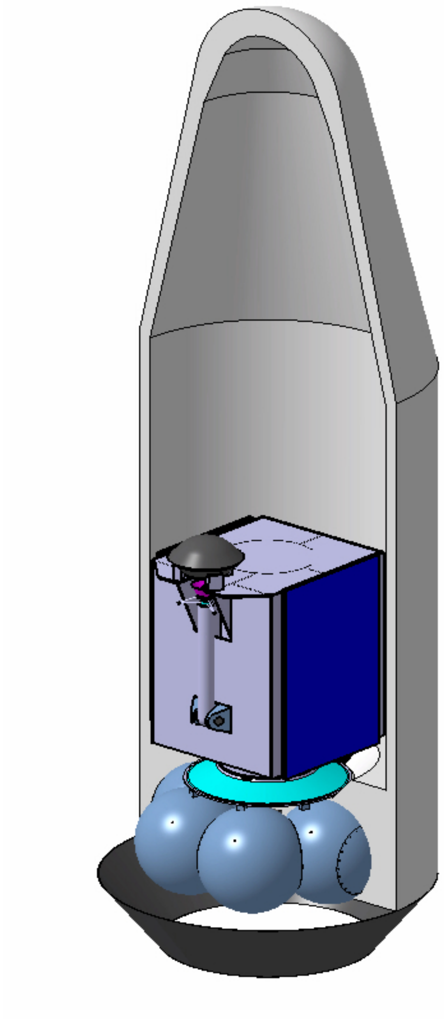
1. S/C & ERC as presented previously (1169kg dry mass)
2. Propellant mass based on the Δv budget presented previously:
   a. Hydrazine: ~146kg
   b. Xenon: ~280kg

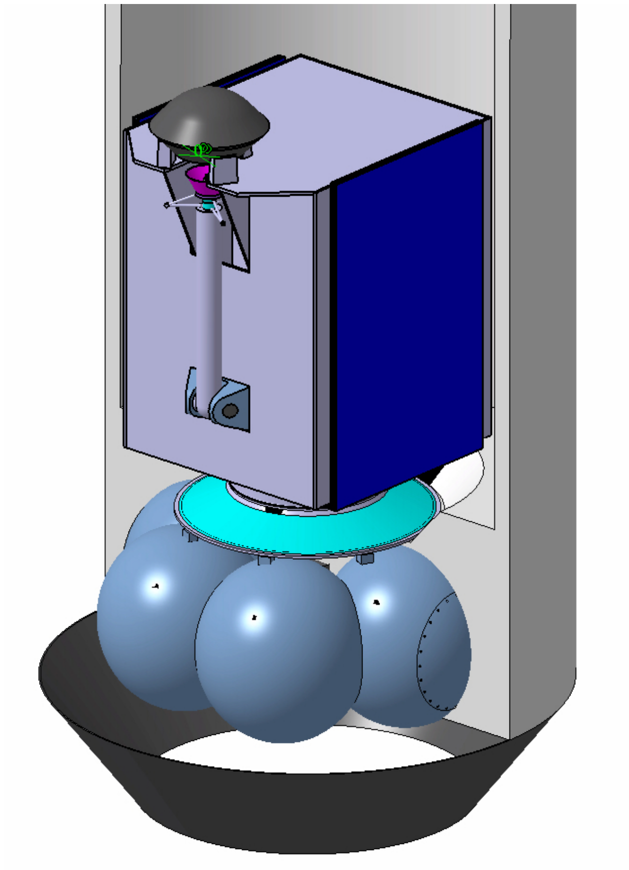
   "Old" MA values

3. Resulting wet mass: 1596kg *(ex. 110kg LVA)*
4. Option Max launch Mass: 1696kg (ex. LVA)
5. Resulting launch margin: 5.5%

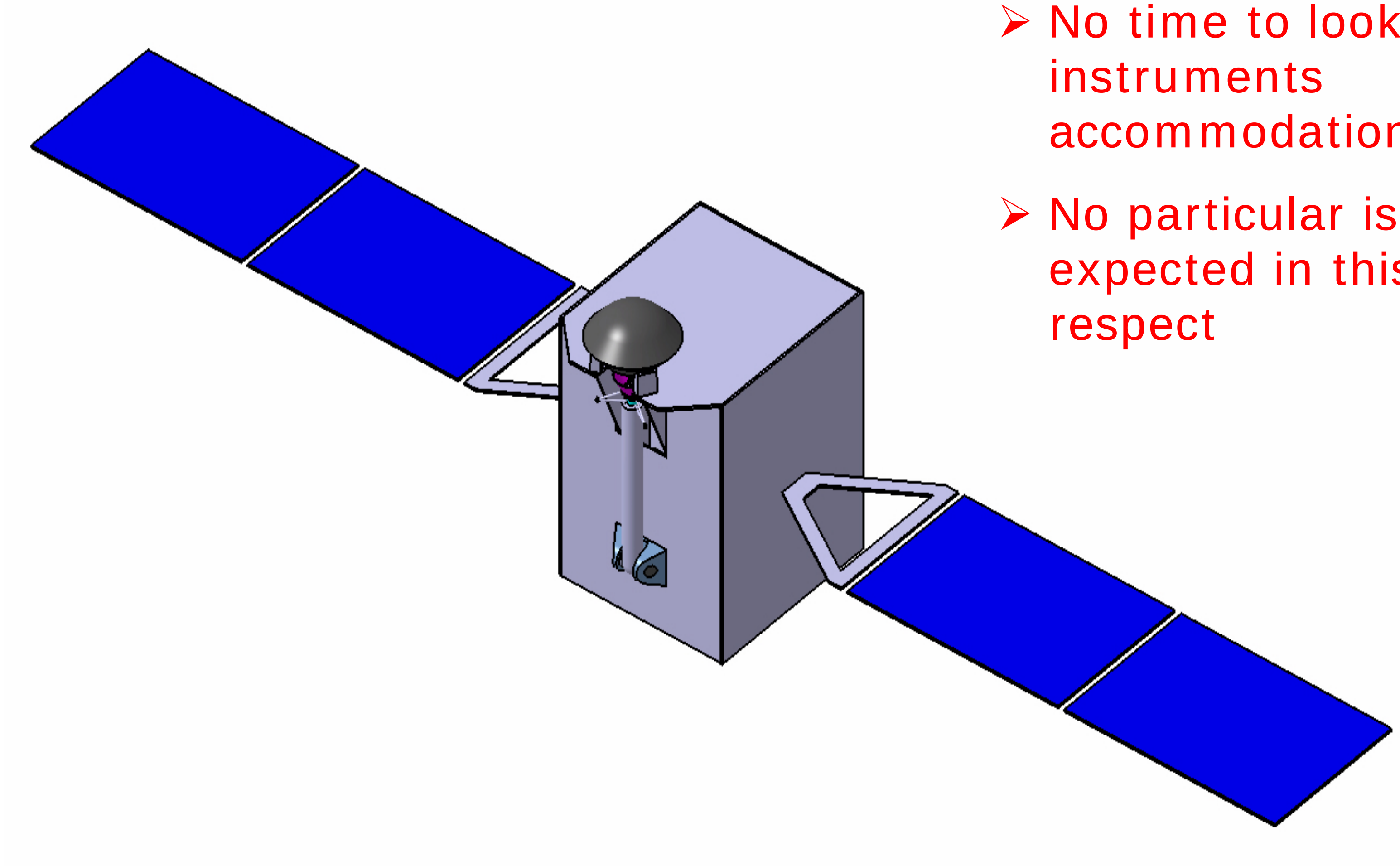
Note: Based on updated MA data
→ Wet mass ~1528kg for launch mass of 1696kg
→ Launch margin ~10%

S/C Design in Soyuz



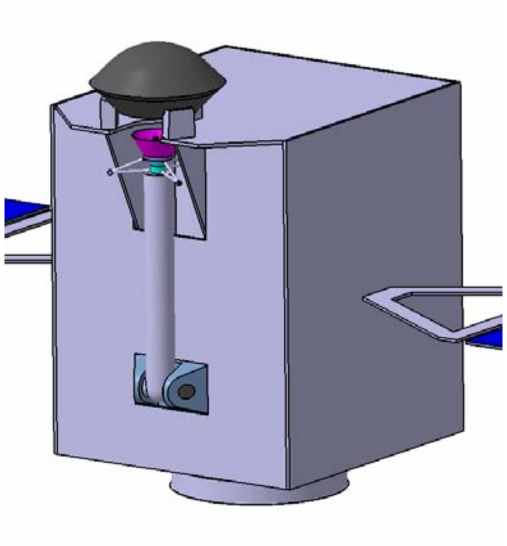


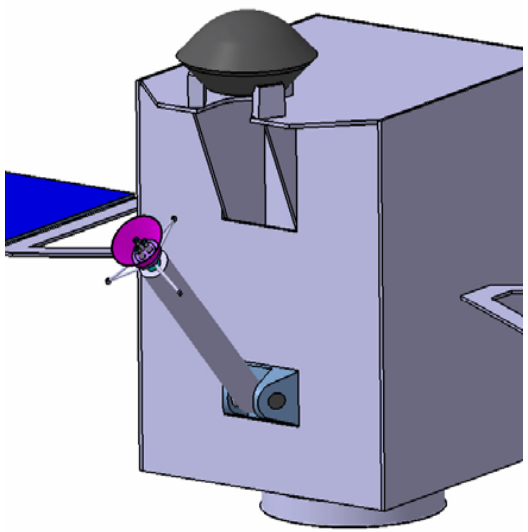
S/C Deployed



- No time to look at the instruments accommodation,
- No particular issue expected in this respect

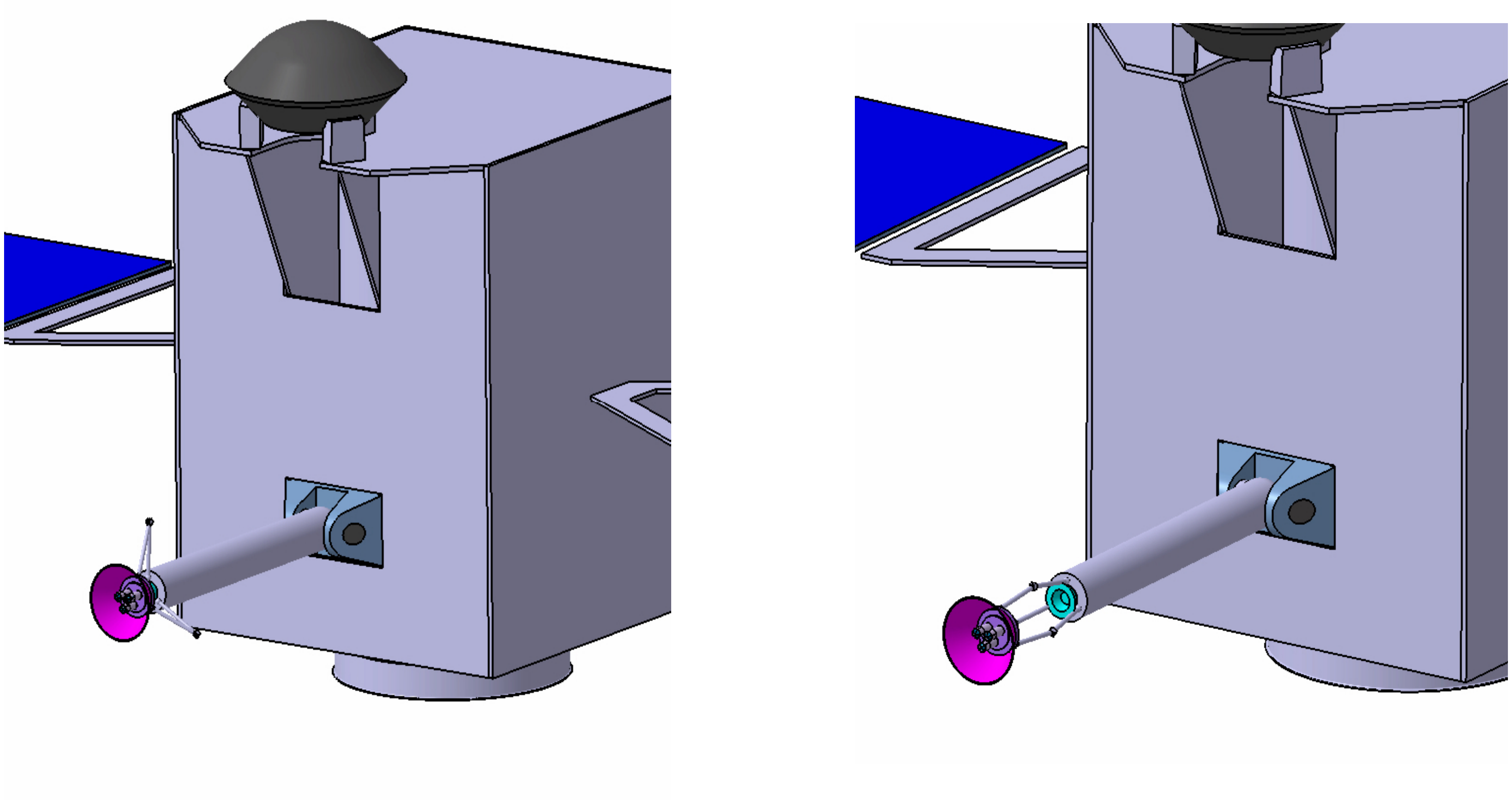
S/C Operation sequence 1





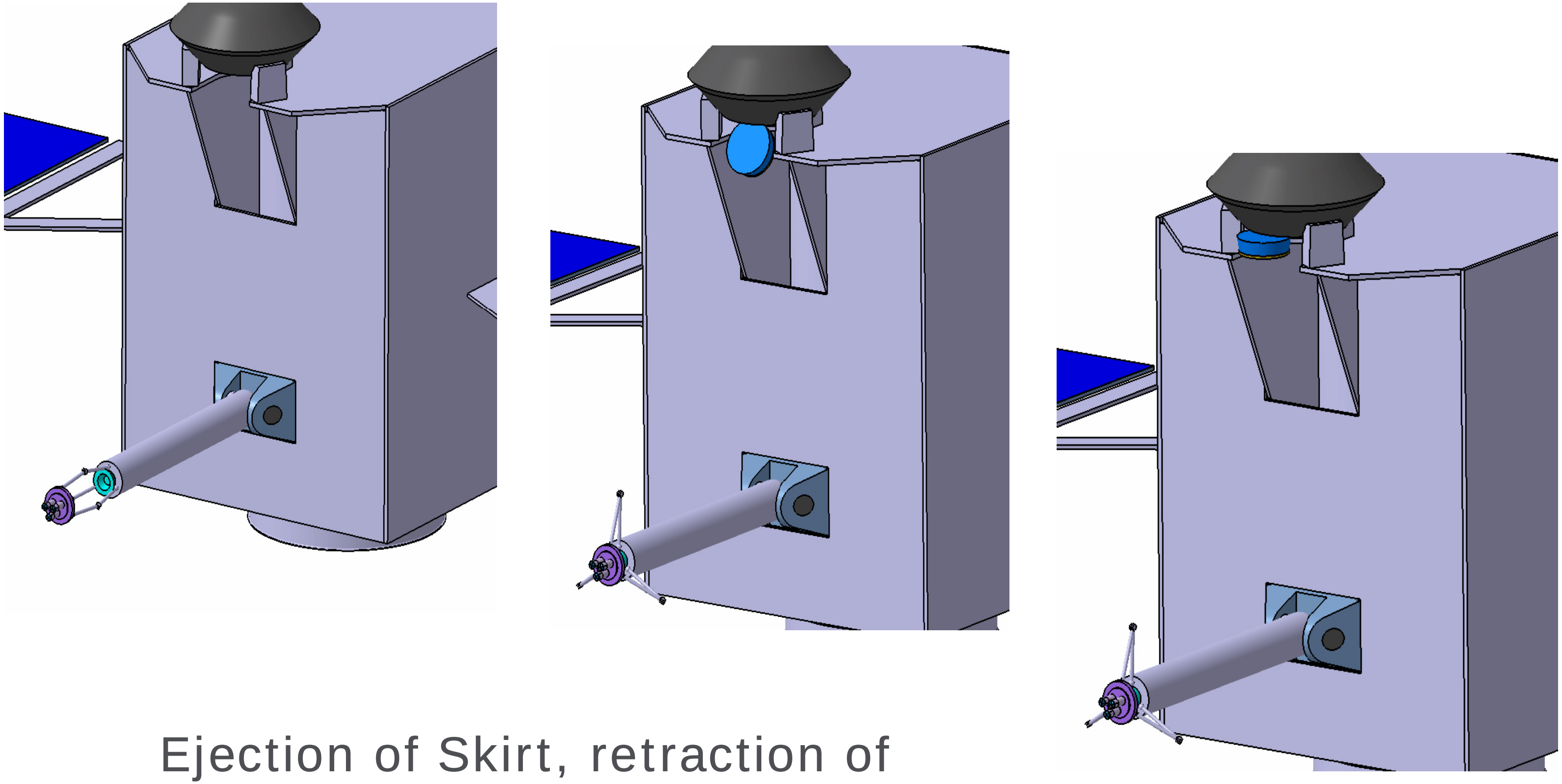
Deployment of the Sampling Boom 1

S/C Operation sequence 2



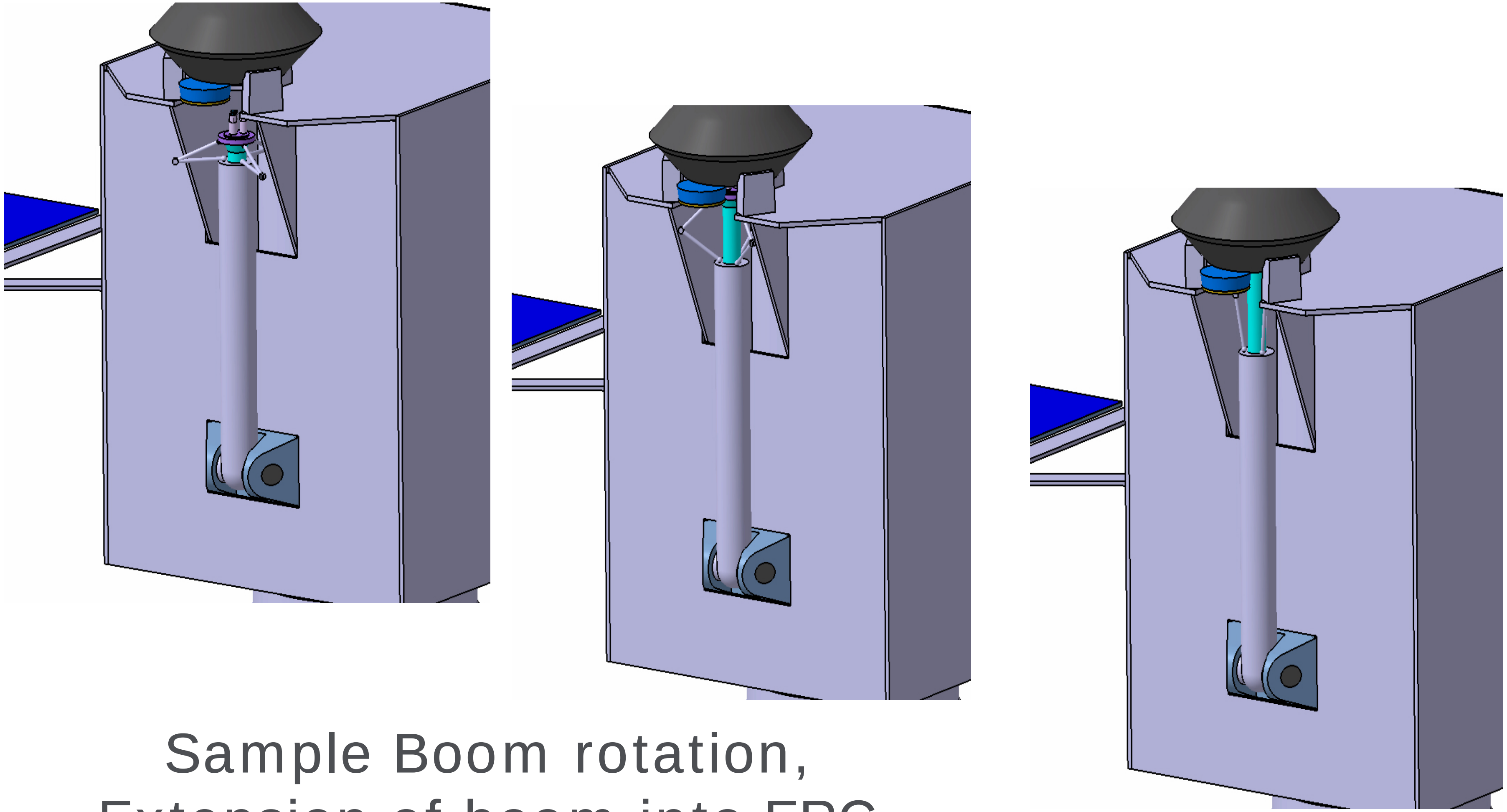
Deployment of the Sampling Boom 2

S/C Operation sequence 3



Ejection of Skirt, retraction of
Sample Boom, Opening of ERC Lid

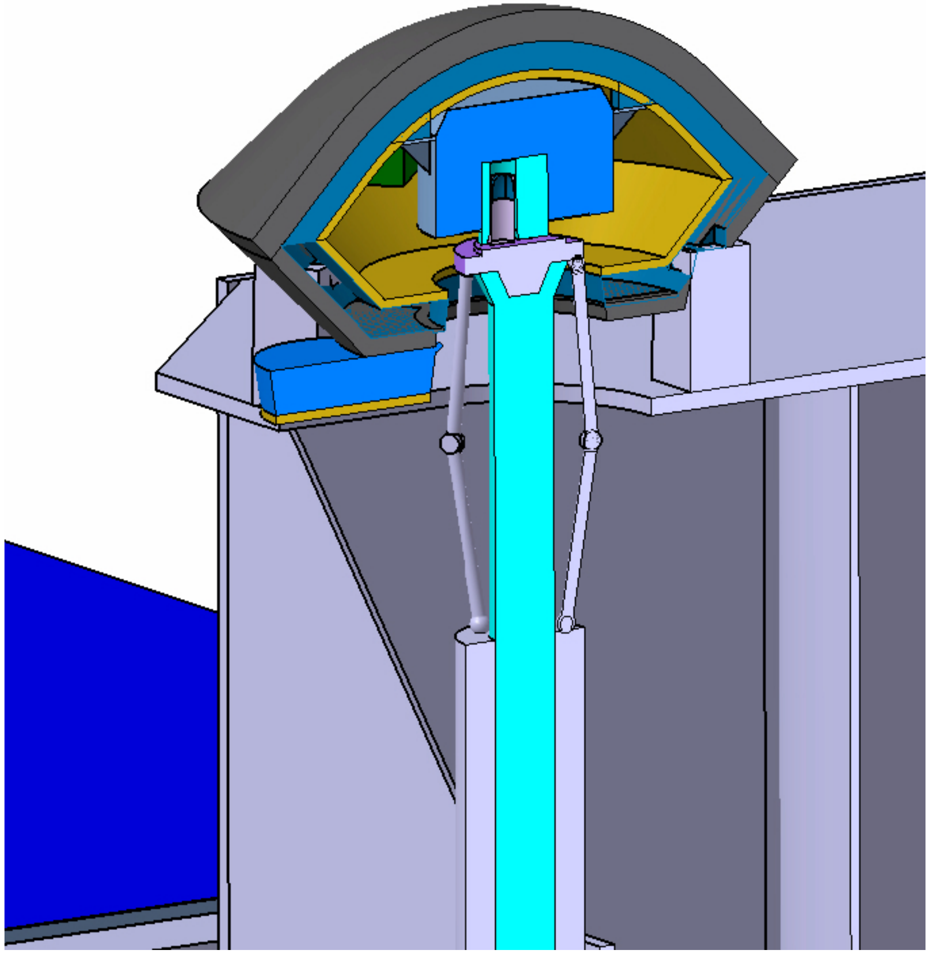
S/C Operation sequence 4



Sample Boom rotation,
Extension of boom into ERC

S/C Operation sequence 5

Storing Sample Canisters in ERC



Further sequence:

Retraction Sampling Boom

Closing ERC Lid

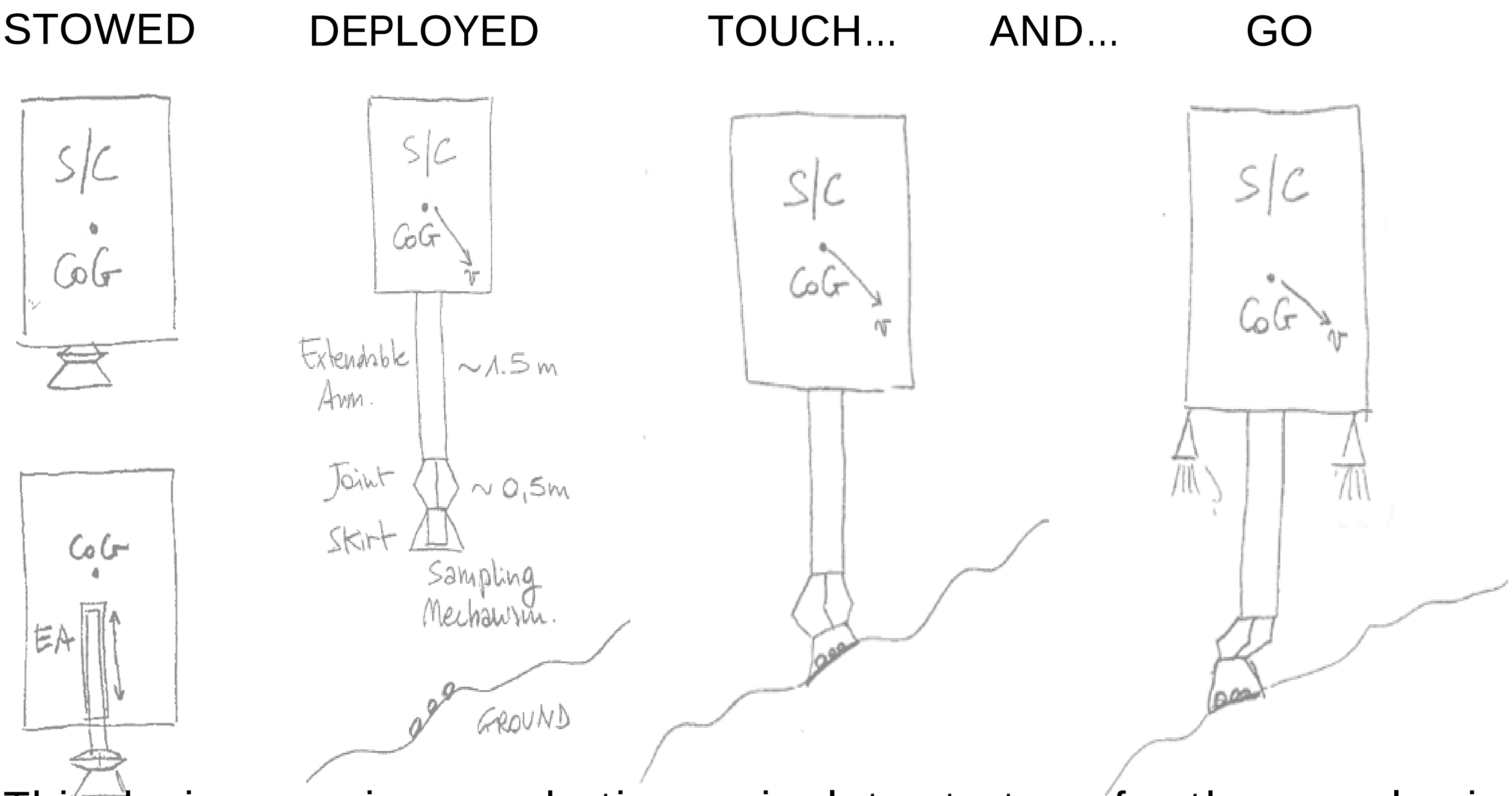
Ejection and Spin-up of ERC

Retrieval of ERC !!!

Assumptions

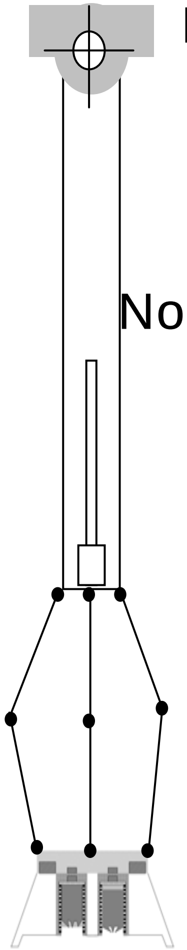
- Sampling strategy: touch and go
- Duration of sampling operation: <2 sec
- S/C distance from ground: 2 m
- S/C relative velocity wrt soil:
  - vertical: 30 cm/s (now 15 cm/s)
  - lateral: 15 cm/s (now 5 cm/s)
- Direction of lateral velocity: unknown
- Slope of the soil: unknown
- Soil density up to 1.8 g/cm3
- Soil temperature up to 200 degrees
- Soil compression strength 2 MPa

Concept design 1 / 6



This design requires a robotic manipulator to transfer the samples in the ERC → **Deemed too costly**

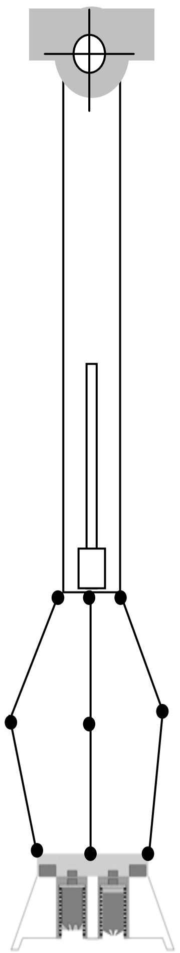
Concept design 2 / 6



Concept design 3 / 6

S/C

Hinge



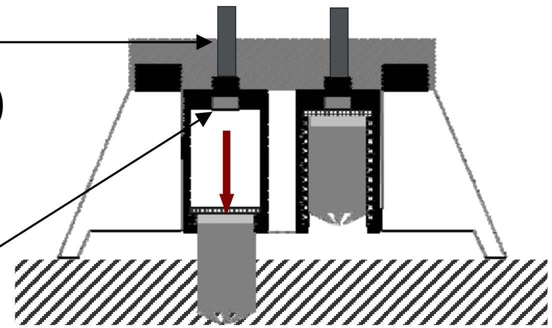
Boom

Passive/Active Joint (very soft to reduce counterforces on the S/C)

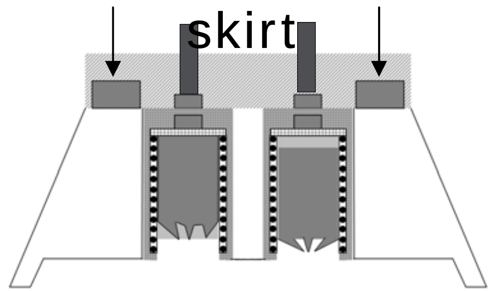
Skirt

Non-explosive Actuator (NEA)

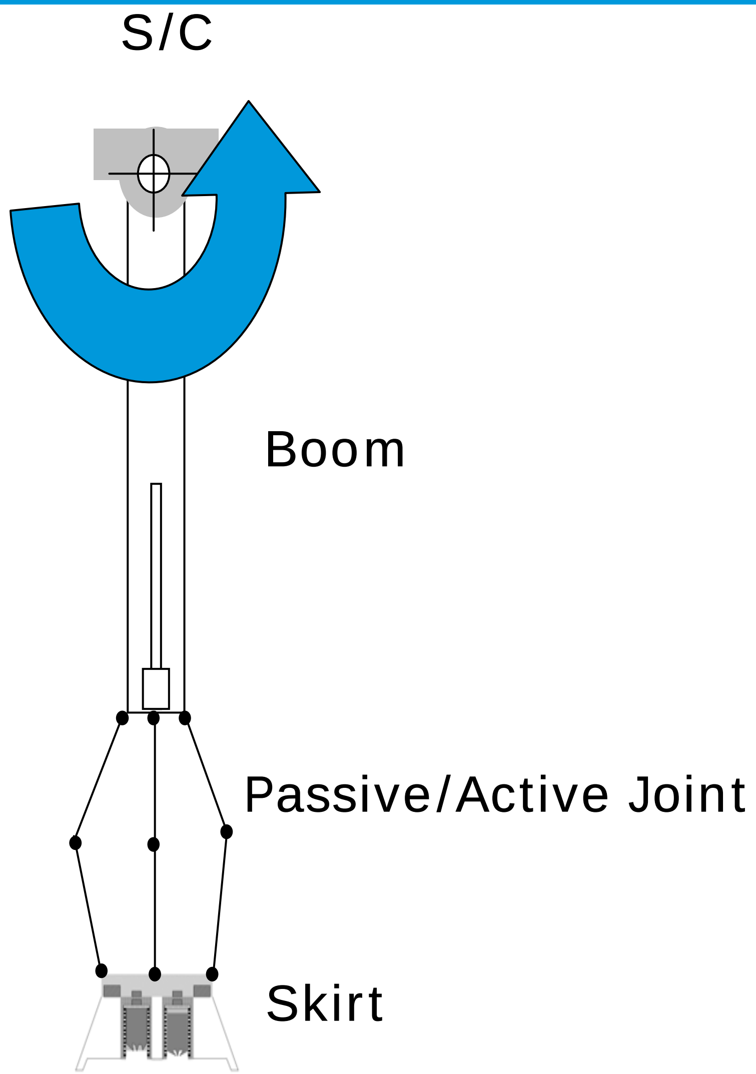
Pyro gas generator cartridge (triggered by sensor on the skirt pad)



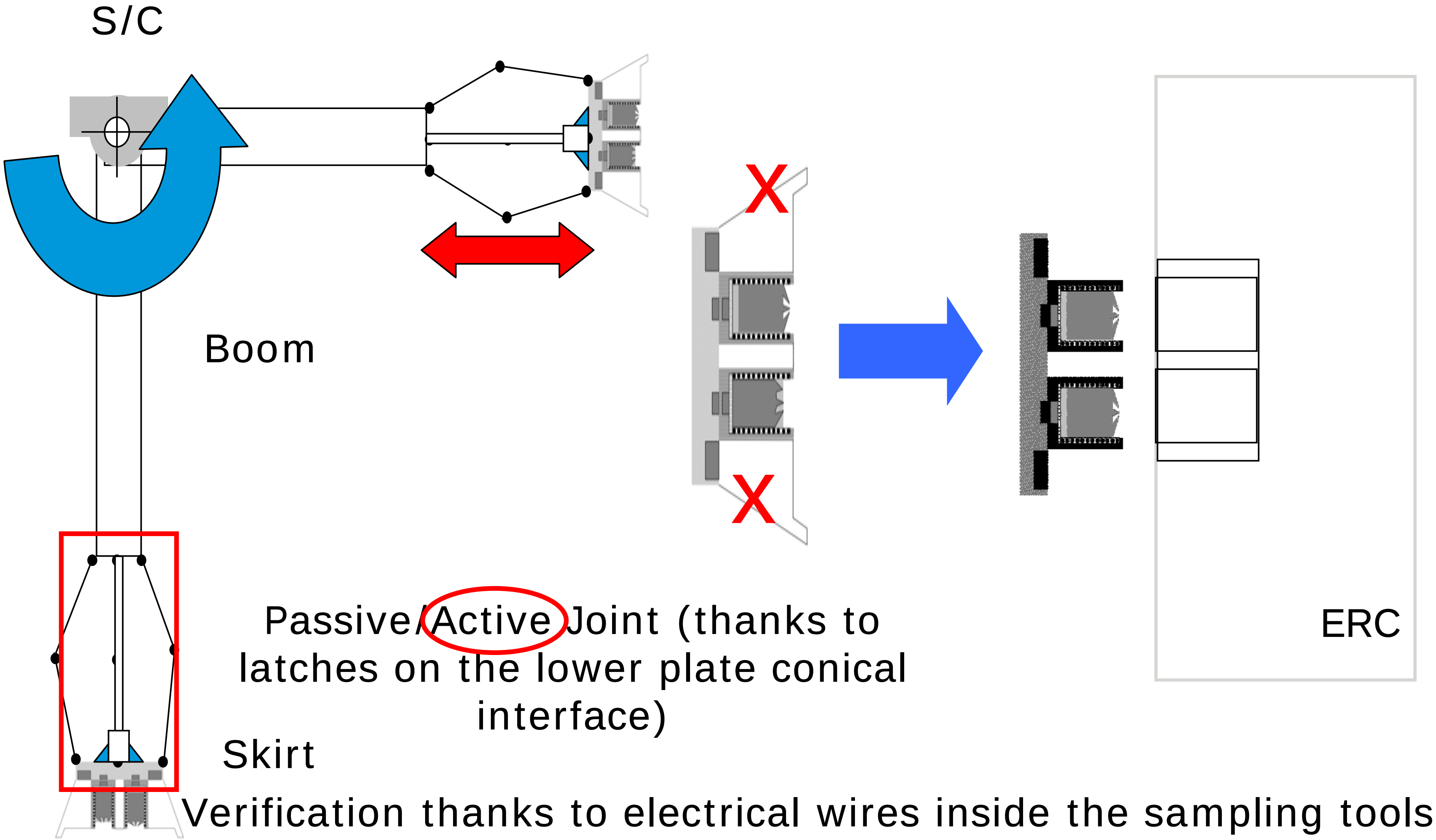
Spring guided pyro devices to eject the skirt



Concept design 4 / 6



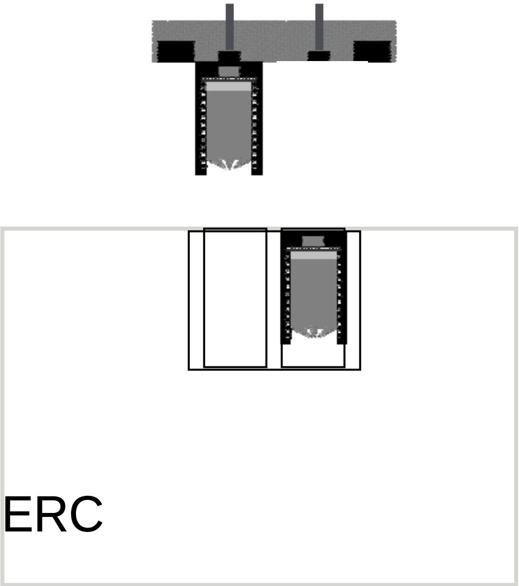
Concept design 5 / 6



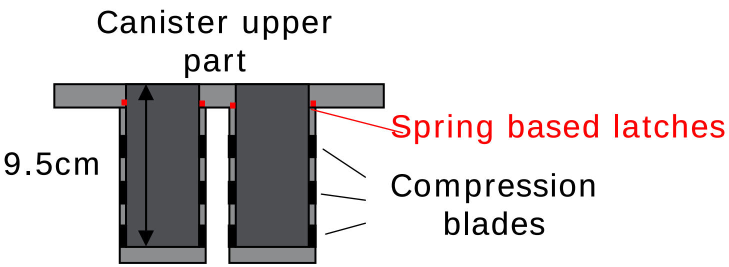
Passive/Active Joint (thanks to latches on the lower plate conical interface)

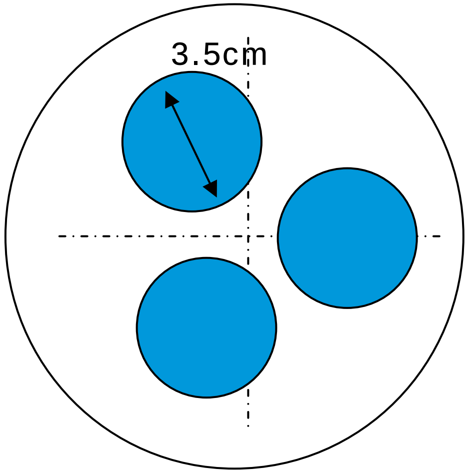
Verification thanks to electrical wires inside the sampling tools

Concept design 6 / 6



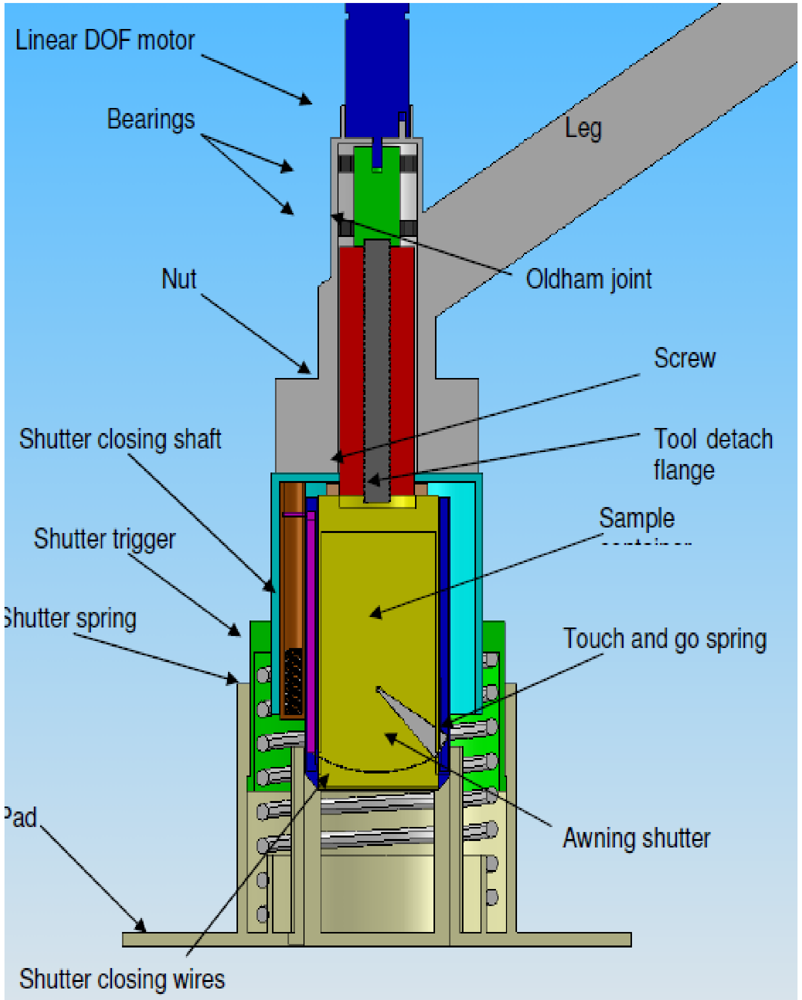
- Skirt released with 3 pyros (spring guided)
- Non-explosive actuators to release sample containers when inserted in the ERC canister
- cylindrical canister to allocate 3 samples (OD6)
- conical latches system to fix the sample containers in the canister (OD20)

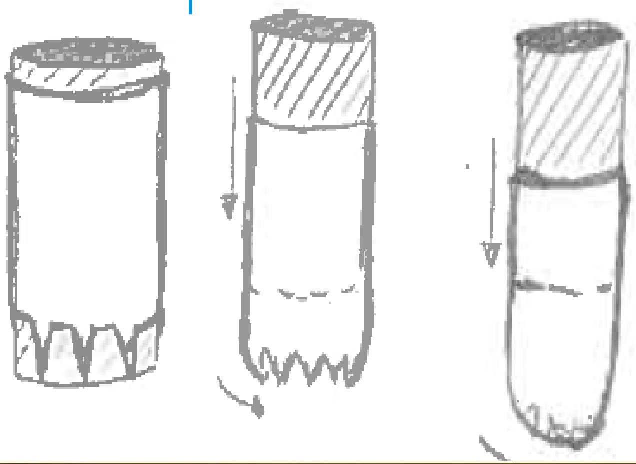




Sample tool

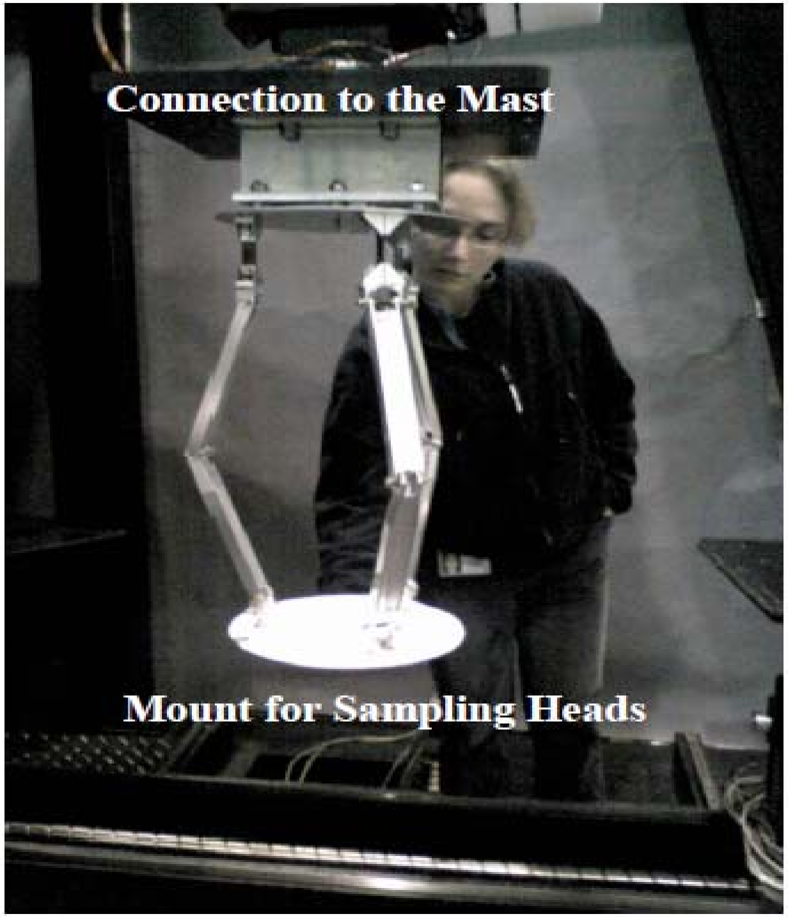
TAS industrial study

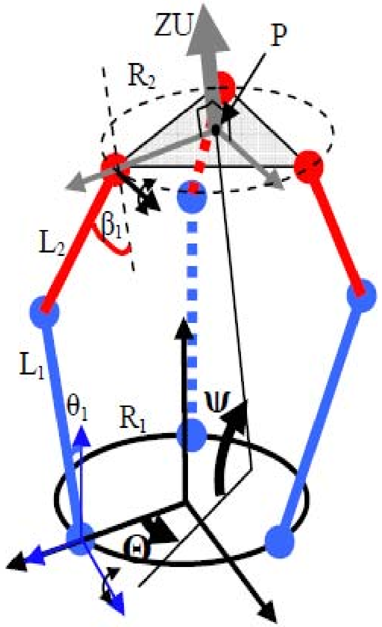




ASTRIUM internal development

Passive / active joint



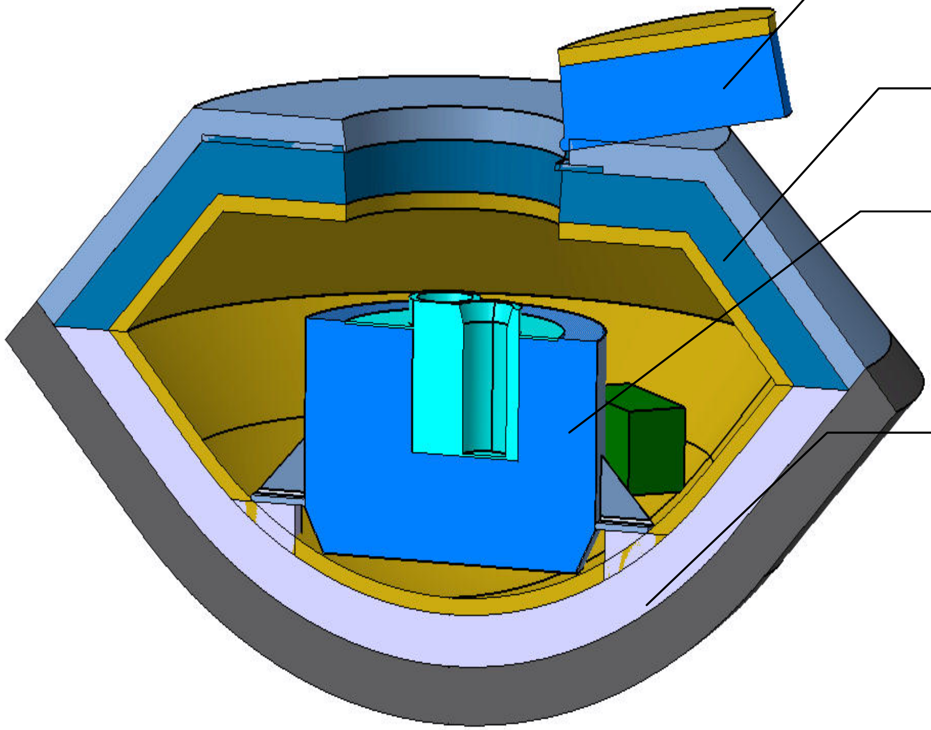


Main comments on the baseline design

- Higher risk to perform two main functions (sampling and transfer) with the same device (articulated joint) compared to landing option
- Counterforces not in line with the S/C CoG
- Thruster action sized to accelerate upwards the S/C enough to prevent it to crash
- Uncertainty in the soil compression strength requires decoupling of descending and sampling functions (best option is low velocity approach and pre-determined sampling forces)
- ERC canister sized to allocate 3 samples instead of 1 (One-shot sampling tool!)
- Closing and latching mechanism mass on the ERC
- Advantages:
  - ✓ No need of further mechanisms to perform sampling and transfer
  - ✓ All-in-one solution → expected lower cost than other solutions

ERC design

Capsule
Diam=0.92 m
Mass=54 kg
FPA =-9 deg!
75 mm crushable material



LID

BC-cold structure

1mm CFRP skin; 38 mm AL honeycomb core

Crushable material

AL honeycomb 1/8 – 5052 – .001:

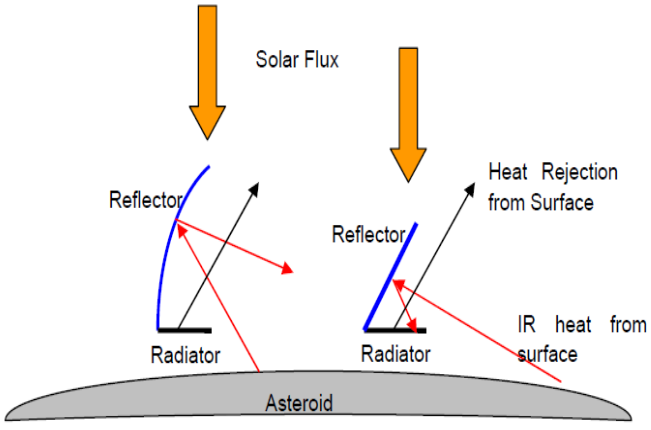
Crush strenght 1.8MPa. Denisty = 72 kg/m3

FS – cold structure

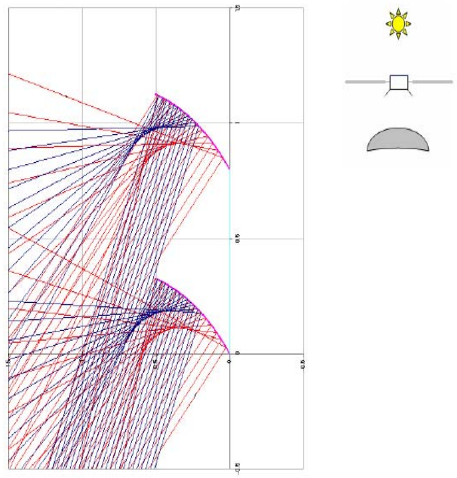
1mm CFRP skin; 38 mm AL honeycomb core

Thermal Design Description

- The platform is Small-GEO=>Radiative surfaces are fixed on the +/-Y axis; → avoids direct solar impingement during transfer.
- During descent, ascent and local characterization radiators see high heat fluxes coming from the asteroid => they must be shielded → OSRs and parabolic fins which reflect away the fluxes

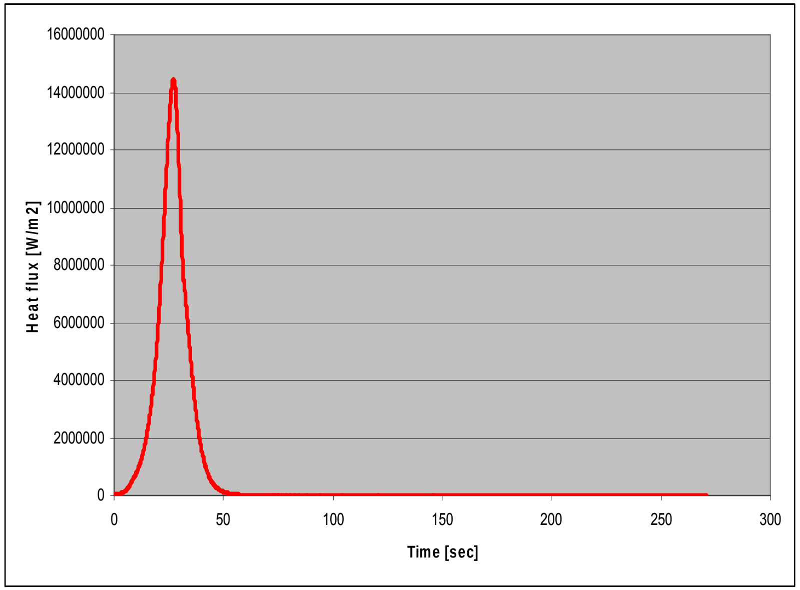


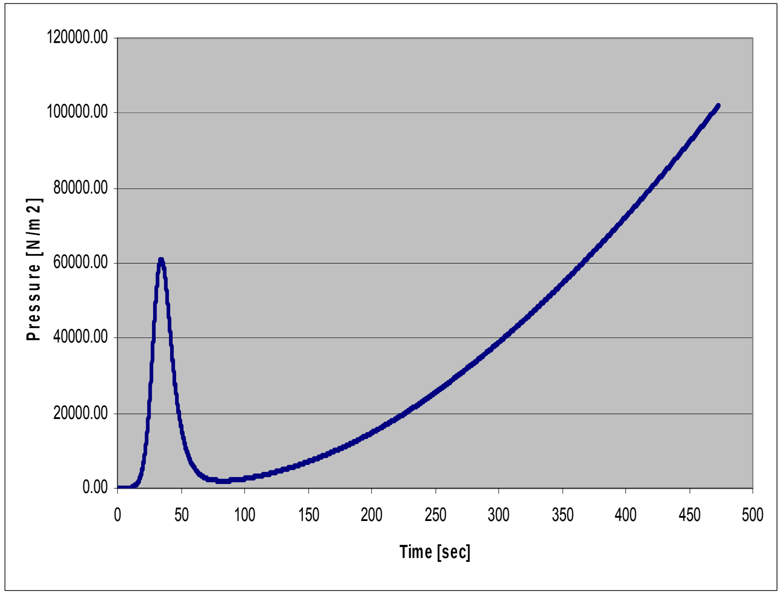
Shape of parabolic reflector ensures IR flux is directed away from radiator surface



- External surfaces not used as radiators are covered with high temperature MLI blankets due to the Venus Fly-by.
- Internal units painted black except Battery, EP tank and pipes which require a very stable temperature control. These units are wrapped in 10 layer MLI blankets to be decoupled from the rest of the internal environment.
- Heaters lines are used for temperature regulation.

Aerothermodynamics data (baseline design)





- On the left the total heat flux acting on the stagnation point of the front shied; (sum of radiative and convective and including margins)
  - ✓ The total flux acting on the back shield is assumed being 10% of the one on the front.
- On the right the stagnation pressure.
- 9 mm char + 1 mm recession + the rest for thermal insulation

ERC Summary and Mass Budget

Earth Reentry Capsule						
	Without Margin		Margin		Total	% of mass w all margin
Dry mass contributions			%	kg	kg	
Structure	**10.90**	**kg**	**20.00**	2.18	**13.08**	28.8
Thermal Control	**2.00**	**kg**	**10.00**	0.20	**2.20**	4.9
Mechanisms	**7.10**	**kg**	**17.25**	1.23	**8.33**	18.4
Communications	**3.50**	**kg**	**20.00**	0.70	**4.20**	9.3
Power	**0.80**	**kg**	**20.00**	0.16	**0.96**	2.1
Balast Mass	**2.00**	**kg**	**0.00**	0.00	**2.00**	4.4
Total Dry	**26.30**				**30.76**	**kg**
System margin			**20.00**	**%**	**6.15**	**kg**
Total Dry with margin					**36.91**	**kg**
TPS (Front & Back Shells)	**14.60**	**kg**	**20.00**	**2.92**	**17.52**	32.2
Launch mass					**54.43**	**kg**

Main GNC Requirements

- Main critical phase= Descent & Landing

- Main critical function: Navigation

- Key technical requirements
  - ✓ Touch-Down velocities (vertical: 15 cm/s, horizontal: 5 cm/s)
  - ✓ Attitude wrt terrain normal: 10 deg.
  - ✓ Landing accuracy: a few 10s meters (e.g. 30-60 m, TBC)
- Programmatic and System-level:
  - ✓ TRL 5 in 2012
  - ✓ Design-to-cost approach

Possible G-Nav-C Strategy

Phase 1: Rosetta-like

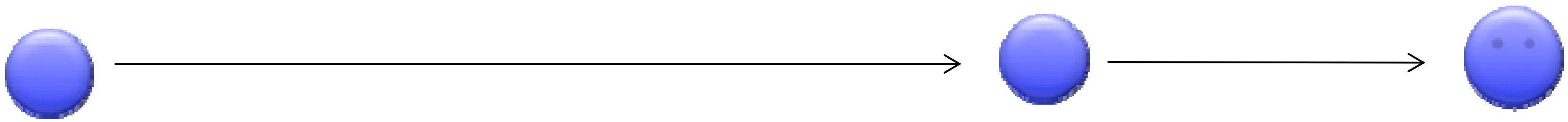
Precise radio-based (+ optical) navigation

Phase 2 (Hayabusa -like)

Safe altitude reduction with ground-in-the-loop

Phase 3 (Simplified M.Polo)

-Mid-Descent (High gate)
-Close-Descent (Low gate)
-Landing (vertical free fall)



Landing site

Rosetta Phases (1-sigma performance)	Drift ~10^5 km	Far Approach	Close A ~25R	Global Mapping	Close Mapping
Target position error	10000 km	150 km	130 km	85 km	78 km
Target velocity error	-	6 cm/s	25 mm/s	21 mm/s	18 mm/s
Position relative accuracy	13000 km	2.3 km	350 m	35 m	4 m
Velocity relative accuracy	0.9 m/s	2.1 cm/s	1.8 mm/s	0.5 mm/s	0.7 mm/s

Proposed GNC Solution

- Relaxation of landing accuracy → no <u>OB</u> absolute nav
  - ✓ No LKs database strictly required
    → Board recommendations from last Marco Polo study fully implemented
- Relative navigation for $V_{lateral}$ control
  - ✓ NPAL-like solution
- Lighter sensors suite
  - ✓ Core: STR/IMU, WAC.
  - ✓ Key T/Os: need of (radar) altimeter : YES (robustness, FDIR)
  - ✓ Dedicated relative terrain sensor for attitude control: NO

GNC D&L Proposal

- Proposed 3-phase approach (reduced autonomy)

- Phase 1: extended Rosetta approach
  - ✓ High perfo radio-navigation (0.6 cm/s, 12 m, 3 sigma)
  - ✓ Same perfo expected at lower altitudes

- Phase 2: Ground-controlled G & N until low gate
  - ✓ **Ground-based** absolute nav during Descent, limited # hoverings

- Phase 3:
  - ✓ Option 2: OB terrain relative navigation ($V_{lateral}$ and attitude control)

GNC Baseline Equipment

Element 1	S/C			MASS [kg]			
Unit	Unit Name Click on button above to **insert new unit**	**Part of custom subsystem**	**Quantity**	Mass per quantity excl. margin	Maturity Level	Margin	Total Mass incl. margin
1	LN200 IMU		2	0.800	Fully developed	5	1.680
2	Hydra STR (3 OH)+(2 EU)		1	4.800	Fully developed	5	5.04
3	RW RSI 12/75-60		4	4.800	Fully developed	5	20.580
4	WAC (2 OH)+(1 EU)		2	2.000	To be developed	20	4.800
5	radar altimeter		1	0.400	Fully developed	20	3.600
6	coarse Sun sensor TNO		2	0.050	Fully developed	5	0.105
-	Click on button below to insert new unit						
SUBSYSTEM TOTAL			**6**	**30.1**		**8.2**	**35.8**

NPAL

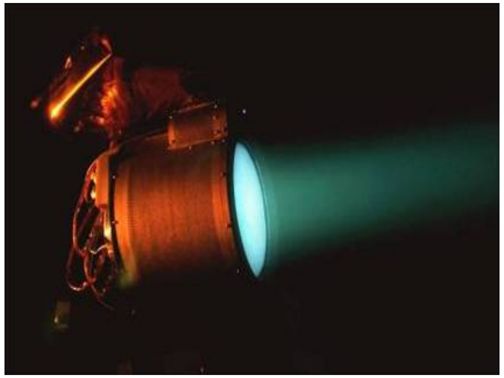
Example: Roke Manor altimeter (Beagle 2)

GNC Conclusions

- A feasible mission, less challenging than before from a GNC viewpoint
  - ✓ Rosetta reuse
  - ✓ R&D effort towards relative navigation ongoing for many years already
  - ✓ Consolidation needed (e.g. comm. delay)
- Proposed cost reduction wrt M. Polo
  - ✓ light radar altimeter
  - ✓ no absolute navigation (perfo. TBC)
  - ✓ no final attitude terrain-relative sensor (perfo. TB checked)
- Options: simplified NPAL design (~MER-DIMES capability only),

Baseline propulsion Design

- Based on T6 Gridded Ion Thruster
- T6 design based on GOCE T5
- The T6 has been developed by QinetiQ (UK) specifically to meet the requirements of both BepiColombo and AlphaBus programmes. Bepicolombo baseline spec are:
  - ✓ Thrust=145mN, ISP=4276sec, Total impulse=20MNs
  - ✓ CDR successfully closed for all equipment (Thruster, PPU, FCU, mechanism..). SS CDR on-going
  - ✓ Any changes to the selected thruster power operating point shall not have a dramatic impact on the existing unit ground qualification.
- The T6 SS is composed by
  - ✓ One nominal thruster and one cold redundant thruster, both equipped with internal cathode and external neutralizer
  - ✓ Other required equipment: Power Processing Unit (PPU), high pressure tank, mechanical pressure regulator, Two Flow Control Units (FCU), Transducers, isolation valves, brackets, pipework, harness….
  - ✓ TBC if thruster orientation mechanisms are required
- Chemical used for planet fly-bys, near-asteroid manoeuvres, attitude control, descent manoeuvres, etc.: ~ 110 kg of propellant, mono-propellant hydrazine system, 16x1N + 4x20N thrusters



Solar Array Sizing

- Several iterations with Mission Analysis and Electrical Propulsion experts
  - ✓ Sized to provide 3 kW to Electrical Propulsion Unit at 1 AU
- 17.0 m2 Array Needed assuming 85 % packing factor
- 68 kg without margin, 4 kg/m2
- EoL Power of 4412 W
- 12m wingspan

Comm Links

* Not the baseline anymore: sec01b case to be computed, even though it is not a standalone sizing case and better then Sec01c for comms

- Sec01c (190 -22 days, 8h/d)*:
  - ✓ 400Gb, 83kbps, X/X, 65W, 1.4m HGA
    - G/S: Ø35m
    - 0.7AU

- Sec04d (244 – 22 days, 8h/d):
  - ✓ 120Gb, 20kbps
    - G/S: Ø35m
    - 1.7AU

1. Selected solution: Fixed HGA, fixed MGA, 2 LGA
2. Compatible with furthest distance to the Sun during cruise (1 kbps) and also safe mode based on MGA (0.4 kbps)

Both solutions with margin and not critical

LGA final Touch-and-go phase

1. Sec01c (0.7AU): carrier recovery + MFSK (MER like)
    - G/S: Ø35m, open loop (post processing) +70m DSN as an option
    - Almost spherical coverage

2. Sec04d (2.1AU):carrier recovery only + MFSK (MER like)
    - G/S: Ø70m mandatory, open loop (post processing)

Mission specific tuning of MER-like design can increase the robustness of the link compared to MER and/or increase the data restitution due to a lower signal dynamics (Doppler, Doppler rate) and better propagation conditions (no atmosphere as on Mars)

Proposed Marco Polo R Timeline

	days	distance	comment
SEP Approach (before stay at asteroid)	120	1 million km - 100 000 km	
	24	100 000 km - 4500 km	Slow approach used for asteroid detection
	6	4500 km – 500km	Min. approach distance similar to Rosetta
Close Approach Trajectory	6	500 km - 100 km	4° full pictures of binary system from ~50 km
Transition to Global Mapping + binary system mapping	15	100 km - 10 km	
Global Mapping + sensors/instrument calibration	28	10 km	

Proposed Marco Polo R Timeline

	days	distance	comment
Global mapping / far global characterization	21	5km	Duration = average Rosetta/Marco Polo
Close observation phase/ global characterization	14	1-2 km TBC	Closer global mapping
Detailed gravity mapping	0	200 m	No dedicated radio science, done together with global observations
Local characterization	35	100 m TBC	As Marco Polo
Landing	35	0	Marco Polo duration (70 days) was sized for 5 landing attempts, to be reduced
Additional science	0		TBD as time available
Asteroid escape preparation	7		As per Marco Polo, SEP impact TBD

Proposed Marco Polo R Timeline

	days	comment
Duration at asteroid (Close Approach trajectory to escape)	161	Only landing/delivery shortened with respect to Marco Polo No shortings compared to Rosetta No 7-day transition phase, can be done much faster Still some margin left (e.g. for radio science or additional science), but only 20 days for instance for trajectory 01c

Study conclusions

The main objective of the study was to reduce the cost of this ESA-led mission concept compared to Marco Polo while maintaining a design feasible and reaching a new more challenging asteroid target !!!

A cost reduction has clearly been achieved but its amplitude is to be assessed during the industrial assessment phase:

- **No landing, simplified lower-accuracy descent and touchdown**
  - ‘Touch and go’ principle, similar to Osiris-REX principle. Sampling mechanism design for 15 cm/s touchdown velocity. Relaxed landing accuracy to tens of meters
- **Highly simplified GNC design + higher confidence in ongoing development**
  - Rosetta-based approach, Inertial navigation until low gate, landing without complex on-board processing
- **Highly simplified sampling mechanism**
  - Only one arm that samples and transfers by extension the samples into the ERC. No robotic arms. Passive articulated joint for terrain roughness compensation, inspired by Hayabusa-2 development.

Study conclusions

- **Short stay time around the asteroid**
  - ✓ Reduces intensive operations time
- **Recurrence**
  - ✓ Structure design based on Small-GEO, AOCS system partly based on Rosetta, altimeter based on enhancements of Beagle-2 H/W, EP system based on commercial GEO platforms/BC, etc.
- **No antenna mechanisms**
  - ✓ No steerable antennas, similar to Osiris-REX design. 170 day operations timeline during 240 day stay-time leads to 30% margin for data downloading (spacecraft Earth pointing) and possible additional science. Possibilities for downloading less critical science data during return trajectory
- **Higher confidence in TPS development**
  - ✓ TPS sizing on heat fluxes is based on ablative material that is under development today in Europe. Samples have already been tested on these heat fluxes at lower pressure
- **Removal of parachutes, fully passive ERC**
  - ✓ Crushable structure designed accordingly

Margin approach and flexibility

- We did <u>not</u> reduce margins. The following margins were still applied:
  a. Launch margin (5%)
     - To overcome launcher performance prediction inaccuracy
  b. System margin (20%)
     - Maturity margin due to phase-0 level
  c. Sub-system margins (5-20%)
     - To overcome changes in the design
  d. Margin on EP thrust time (10%)
     - To compensate for thrusters switching off, S/C off-pointing for comms downloading etc.
  e. Margin on EP thrust & Isp level
     - The Isp & thrust polynomials used apply normally at EOL only; here they were used for all the mission
  f. Propellant margin (3% for chemical, 10% for electrical)
- The EP system allows for flexibility
  a. Non flexibility is typically a potential for cost increase
  b. Compatible with multiple launch years: 2021, 2022, 2023
     - 2022 launch is same as 2023 launch, with added Earth swing-by; this allows for flying the same transfer even with 1 year delay in launch
  c. Change in trajectory has a relatively small impact
     - In case a new target is selected, relatively small impact on Xenon mass
  d. Slow asteroid arrival/departure
     - Allows for flexibility in operations.
  e. Mass still below maximum allowed dry-mass
     - CDF design is NOT mass optimized (re-use of SGEO structure etc., NO propulsion stage) but could be optimized further. Even with 5% launch margin and 20% system margin, the dry mass may still increase
     - Mass overshoots can simply be compensated by a slower trajectory (lower launch energy) using additional swing-bys

This study is based on the CDF Integrated Design Model (IDM), which is copyright.