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Mars Sample Return Architecture Assessment Study

Simone Centuori[a]*, Pablo Hermosín[a], Javier Martín[a], Gabriele De Zaiacomo[a], Colin Stroud[b], Alexander Godfrey[b]. Myles T. Johnson[b], Holly Johnson[c], Tej Sachdev[c], Rohaan Ahmed[c]

[a] *Deimos Space S.L.U., Ronda de Poniente 19, Edificio Fiteni VI, P2P1, Tres Cantos, Madrid, Spain,* Simone.centuori@deimos-space.com

[b] *Lockheed Martin UK Ampthill, Reddings Wood, Ampthill, Bedford MK45 2HD, United Kingdom,* colin.a.stroud@lmco.com

[a] *MDA Corporations, 9445 Airport Road, Brampton, Ontario L6S 0B6, Canada,* Holly.Johnson@mdacorporation.com

* Corresponding Author

The current paper aims to present the results of the ESA funded activity "Mars Sample Return Architecture Assessment Study": its objective has been to identify the critical parameters of the MSR mission, perform the relevant trade-offs at both mission and system level, evaluate different mission scenarios and select the best candidates to be then furtherly studied.

The study has been carried-out by an industrial consortium composed by DEIMOS Space S.L.U. as prime contractor and responsible for the mission design, Lockheed Martin UK Ampthill for the system design, mass budget and risk analysis and MDA Corporation for the payload mechanisms.

Mars Sample Return is a joint collaborative project of ESA and NASA aimed at bringing to Earth several surface and atmosphere samples from the Red Planet. The mission is considered a major milestone to enable Mars human exploration, because it will allow scientists to better understand the characteristics of Mars and, based on this information, to design the infrastructure that will receive the first astronauts travelling to the Red Planet.

Such complex objective envisages several mission phases, from the Earth-Mars transfer to the Mars orbital phase, the descent and landing on the Martian surface, the ascent from the Red Planet, the inbound leg towards Earth and the entry in the terrestrial atmosphere followed by the landing on our planet of the capsule containing the astrobiological sample.

Several trade-offs of all the mission design parameters have been studied during the course of the activity in order to evaluate the mission feasibility and its sensitivity to the most critical design drivers: the combination of candidate propulsion systems (1 chemical and 4 electrical) together with the consideration of all possible staging scenarios and the possibility of performing a full or partial aero-braking for Mars orbit acquisition, led to the analysis of more than 500 mission scenarios.

An extensive analysis has then been conducted at both mission and system level to verify the fulfilment of the mission goals including: the full definition of the transfer trajectories and rendezvous operations, the entry corridor in the Earth return together with the dispersion at Earth landing site, the mission risk assessment and the payload mechanisms definition.

The most promising options foresaw full chemical propulsion or a mixed system joining the chemical thrusters with the powerful ARM electrical engine. In both cases the mission will rely on the use of aero-braking and staging to reduce the spacecraft wet mass.

1. Introduction

Mars Sample Return (MSR) is a joint collaborative project of ESA and NASA aimed at bringing to Earth several surface and atmosphere samples from the Red Planet. The mission is considered a major milestone to enable Mars human exploration, because it will allow scientists to better understand the characteristics of Mars and, based on this information, to design the infrastructure that will receive the first astronauts travelling to the Red Planet.

Such complex objective envisages several mission phases, from the Earth-Mars transfer to the Mars orbital phase, the descent and landing on the Martian surface, the ascent from the Red Planet, the inbound leg towards Earth and the entry in the terrestrial atmosphere followed by the landing on our planet of the capsule containing the astrobiological sample.

Different architecture options have been evaluated along time, and the currently selected one envisages the coordination of three spacecraft launched between 2020 and 2026 (see Figure 1).

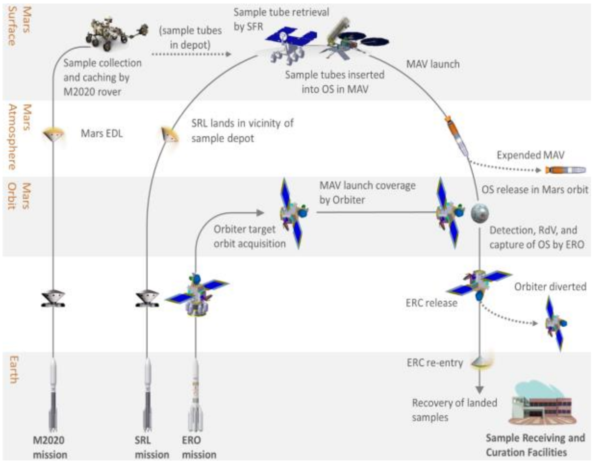

Fig. 1: MSR overall architecture

- <u>M2020</u> will deliver a rover on the Red Planet surface to sample Mars terrain and atmosphere and to then cache the samples tubes in dedicated depots
- <u>The Sample Return Lander (SRL)</u> is comprised of three elements; a surface platform, a Sample Fetch Rover (SFR), and a Mars Ascent Vehicle (MAV). Once deployed to the surface, the SFR will traverse toward the depot(s), collect the sample tubes, and return to the lander platform. The sample tubes will then be transferred into an Orbiting Sample (OS) containment element, and loaded on-board the MAV. The MAV launches the OS into Mars orbit, where it is to be captured by the ERO mission.
- <u>The Earth Return Orbiter (ERO)</u> will detect, rendezvous with, and capture the OS, before biosealing it and transferring it safely to the Earth Re-entry Capsule (ERC) that will be released by ERO on an Earth entry trajectory

During the MSR Architecture Assessment Study (MSRAAS) several mission configurations have been analysed at architectural level, to take into account the possibility of using different propulsion systems for different manoeuvres and to evaluate the impact of various sequences of staging event. The goal was to minimise the spacecraft wet mass at launch while meeting the tight timeline requirements to return the samples back to Earth before the end of 2029 and matching the OS availability for rendezvous (RdV) in Low Mars Orbit (LMO) after May the 1st 2028. During the course of the study another important constraint was raised at satellite system level, where the spacecraft power capability was limited to 20kW to match feasible solar panels technologies[1]: the impact of this constraint is considerable since the available power strongly affects the electric propulsion thrust capabilities and, therefore, the duration of the associated transfers.

It is important to comment that during MSR phase-A study, the currently ongoing follow up of MSRAAS where DEIMOS Space is still involved in the mission analysis, the earliest date for the RdV between ERO and the OS has been modified together with the maximum available power for ERO. The proposed solutions have remarkably changed between the two studies.

As any sample return mission[2], MSR encompasses high design complexity derived from the numerous phases to be accomplished. Due to the significant impact on the mass margin, one of the most critical aspects is the propulsion system selection, for which different options could be considered (see Table 1 for the engines detailed performances):

- Chemical propulsion (CP) for all manoeuvres
- Electrical propulsion (EP) for all manoeuvres considering different EP engines
- A mixed propulsion system making use of different engines for the different manoeuvres

This yields a total of 66 spacecraft scenarios to assess. Moreover for each scenario eight different staging options are considered to yield a total of 528 spacecraft permutations to study. These configurations will be analysed primarily at mass budget and timeline level to evaluate their feasibility.

2. Mission Analysis

The mission analysis has been of capital importance during the MSRAAS activity to define the detailed mission timelines, highly influenced by the propulsion system capabilities, and to evaluate the mission feasibility in the proposed time frame. Its main objective is to optimise the main mission manoeuvres:

- <u>MOI</u>: Mars Orbit Insertion into a 4-sol high elliptic orbit after the interplanetary cruise
- <u>TOA</u>: Target Orbit Acquisition to reach LMO
- <u>DOA</u>: Departure Orbit Acquisition to raise LMO
- <u>TEI</u>: Trans-Earth Injection to leave Mars orbit

2.1 Launchers

The launcher families considered in this study are Ariane 6[3], Atlas V[4], Falcon[5], and Delta IV (Figure 2; for the sake of completeness, it has been highlighted in light orange and light green the range of the required C3 for chemical and electric outbound, respectively).

Engine	Type	Thrust	Isp
HTAE	Chemical	1100 N	320 sec
ARM	Electric	500 mN	2400 sec
PPS-5000	Electric	310 mN	1900 sec
NEXT-C	Electric	280 mN	4000 sec
T6	Electric	145 mN	4010 sec

Table 1: Engines considered for MSRAAS

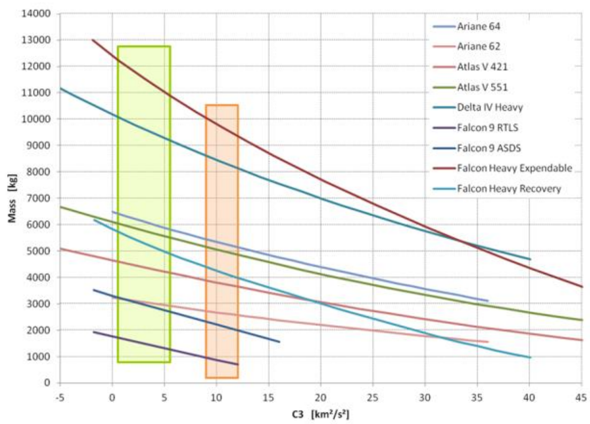

Fig. 2: MSRAAS launchers performance summary

2.2 Interplanetary trajectories

One of the critical aspects of any interplanetary mission is the trajectories selection to fulfil its goals while minimising the spacecraft propellant mass: of course trajectories and DV are strictly dependent on the selected propulsion system and this has to be properly chosen according to mission requirements and goals. All the optimisations have been conducted with LOTNAV[6], a tool developed for ESA to allow systematic analyses of trajectory optimisation, navigation and guidance of interplanetary probes utilising both chemical and low-thrust propulsion. Electrical transfers have been calculated with the assumption of zero relative velocity at Mars arrival and departure; the impact of using the chemical propulsion to reduce a residual relative velocity at Mars arrival or to give an initial v-infinity at Mars departure has also been evaluated.

The outbound details for the chemical propulsion case are detailed in Table 2, while the electric propulsion solutions are presented for four different engines configurations in Table 3 and Table 4.

LAUNCH PERIOD	OPENING	CLOSURE
Launch epoch:	19/10/2026	08/11/2026
Hyp. escape vel. (km/s):	3.17	3.17
Escape declination (deg):	22.8	35.4
Launcher perf. (kg):	5343	5343
Wet mass at escape (kg):	5233	5233
Date of DSM:	-	-
Delta-v DSM (m/s):	0	0
Arrival epoch:	03/09/2027	08/09/2027
Hyp. arrival vel. (km/s):	2.5987	2.5651
Days launch to MOI	318.9	304.1
Delta-v MOI (m/s):	836.7	818.8
Delta-v TOA-1 (m/s):	1322.9	1322.9
Delta-v TOA-2 (m/s):	2159.6	2141.7
Total delta-v (m/s)	2613.4	2588.3
Mass at target body (kg):	2629.6	2644.7

Table 2: CP baseline launch window detail

SELECTED ENGINES	2xT6	2xARM
Escape date	07/09/2026	05/10/2026
Hyp. escape vel. (km/s)	1.66	1.1214
Esc. declination (deg)	5.88	6.336
Engine Isp (s)	4000	2400
Escape Mass (kg)	2535.6	2889.17
Thrust switch on	28/09/2026	05/10/2026
		24/06/2027
Arrival date	24/03/2028	23/10/2027
Hyp. arrival vel. (km/s)	0.100	0.100
Arrival declination (deg)	-55.3	11.13
Propellant mass (kg)	233.64	597.81
Duration (day)	563.1	383
Delta-V (m/s)	3788.9	5452.5
Arrival Mass (kg)	2302.0	2291.36

Table 3: EP baseline launch window detail (1/2)

SELECTED ENGINES	2xPPS5000	2xNEXT-C
Escape date	11/10/2026	02/10/2026
Hyp. escape vel. (km/s)	1.9	2.39
Esc. declination (deg)	8.24	8.12
Engine Isp (s)	1900	4000
Escape Mass (kg)	2757.66	2642.3
Thrust switch on	11/10/2026	09/09/2026
	26/06/2027	08/04/2027
Arrival date	23/11/2027	25/11/2027
Hyp. arrival vel. (km/s)	0.100	0.100
Arrival declination (deg)	-0.89	-10.5
Propellant mass (kg)	656.86	336.5
Duration (day)	408	419
Delta-V (m/s)	5065.84	5300.90
Arrival Mass (kg)	2100.8	2305.8

Table 4: EP baseline launch window detail (2/2)

An example of the trajectories obtained for electric propulsion systems is then depicted in Figure 3, where the transfer afforded with two T6 engines is graphically illustrated.

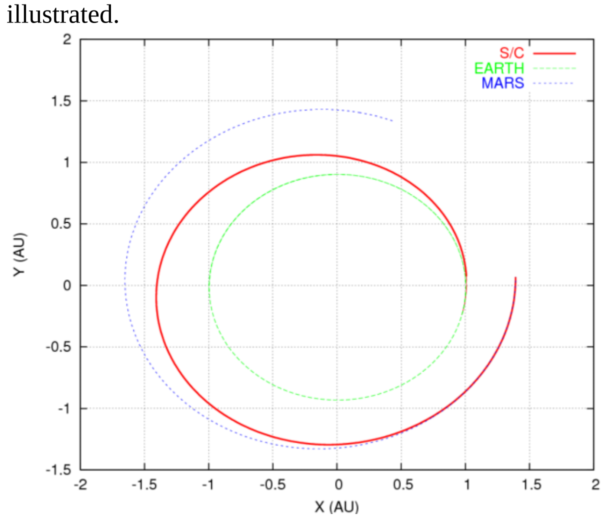

Fig. 3: Example of outbound EP trajectory (2xT6)

The inbound trajectories shall target two landing sites (Utah and Woomera) and are constrained by the return date in 2029: considering the Mars operations duration, ERO shall leave Mars during the second half of 2028. Thanks to the high thrust available, this does not turn into a problem for CP solutions (Table 5), but the return journey accommodation could be critical for EP (see Table 6 and Table 7). This is especially true due to the longer time needed to acquire the departure conditions from LMO, requiring long spiral trajectories around Mars. The departure mass is fixed at 1900 kg for the two T6 engines[7] and at 1250 kg in the other cases[8].

LANDING SITE	Utah	Woomera
Escape year	2028	2028
DSM	No	No
Arrival constraint	No	Yes
Mars escape date	11/09/2028	23/09/2028
Hyp. escape V (km/s)	2.54	3.231
TEI (w/o GL) (m/s)	716.1	1129.88
Earth arrival date	03/06/2029	22/06/2029
Hyp. arrival V (km/s)	5.587	5.459
Arrival declination (deg)	-3.774	-30.000
Delta-V w/o GL (m/s)	716.1	1129.9
Duration (day)	265.1	271.7

Table 5: CP inbound trajectories detail

SELECTED ENGINES	2xT6	2xARM
Mars escape date	23/07/2028	07/11/2028
Escape Mass (kg)	1900.0	1250
Engine Isp (s)	4000	2400
Thrust switch on date	23/07/2028	07/11/2028
Earth arrival date	24/04/2029	14/07/2029
Hyp. arrival V (km/s)	5.5	4.5
Arrival declination (deg)	15.53	-13.26
Arrival Mass (kg)	1727.62	1052.13
Delta-V (m/s)	3737.6	4053.13
Propellant mass (kg)	172.38	197.87
Duration (day)	428.0	249

Table 6: EP inbound trajectories detail (1/2)

SELECTED ENGINES	2xPPS5000	2xNEXT-C
Mars escape date	03/10/2028	01/09/2028
Escape Mass (kg)	1250	1250
Engine Isp (s)	1900	4000
Thrust switch on date	03/10/2028	01/09/2028
Earth arrival date	13/07/2029	09/08/2029
Hyp. arrival V (km/s)	4.23	4.5
Arrival declination (deg)	-20.14	16.41
Arrival Mass (kg)	1025.69	1132.12
Delta-V (m/s)	3682.63	3882.82
Propellant mass (kg)	224.31	117.88
Duration (day)	283	342

Table 7: EP inbound trajectories detail (2/2)

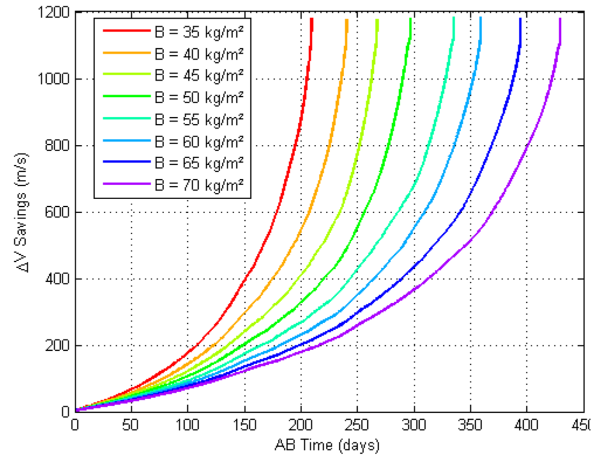

Fig. 4: Aero-braking parametric analysis

2.3 The Mars phase

The Mars mission phase has the main objective to capture the OS after its launch from Mars surface. It then requires selecting a LMO and to define the RdV operations. Moreover, to reduce the amount of needed propellant, the possibility of performing aero-braking in Mars atmosphere has been explored.

Aero-braking, in fact, allows saving part of the chemical burn for TOA by reducing the orbit apocentre thanks to the air drag effect of an atmospheric passage. A parametric analysis for different values of the ballistic coefficient has been performed (Figure 4): with enough time, more than 1.1 km/s could be saved with respect to a chemical TOA of 1.32 km/s. Additionally, the impact of partial aero-braking has been analysed, calculating the DV saving as a function of the amount of available time to perform the TOA manoeuvre. The calculations have been performed with ABMAT[9], a dedicated tool to assess the aero-braking performance and robustness: ABMAT reproduces faithfully real mission operations timelines by simulating sequentially the three main aero-braking phases (Walk-In, Main Phase and Walk-Out) with precise orbit GNC architecture, in terms of both ground and on-board operations.

The criteria to trade-off the operational LMO are summarised in Table 8: a polar orbit offers better performances in terms of observation, illumination and communications, but also a slower RAAN drift (longer waiting time for departure plane acquisition after RdV). A preliminary inclination of 45º is then proposed.

Criteria	Low inc	Mid inc	High inc
Arr/Dep Declination	-	+	++
RAAN drift	++	-	--
Surface coverage	-	+	++
Revisit frequency	/	/	/
Eclipse duration	-	++	++

Table 8: Low Mars orbit inclination trade-off

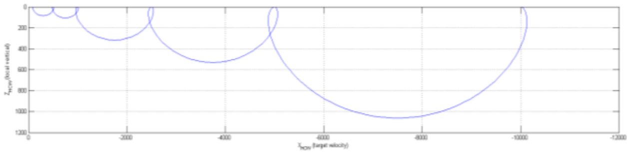

Fig. 5: Approach analysis from 10 to 0.1 km (in m)

A rendezvous analysis has also been performed to optimise this mission phase duration and the associated propellant consumption (the resulting trajectory of the final phase can be seen in Figure 5). The RdV operations can be performed in less than 21 days and require a reduced DV cost below 40 m/s.

2.4 Local Entry corridor and EDL performances

The Earth Re-entry analysis is a multi-step process[10] that has the final objective to define the reference trajectory for the Entry Descent and Landing (EDL) phase of the ERC, and to support the capsule design. At first, a 2-D local entry corridor (LEC) analysis was performed, with proper worst cases combinations of the variability expected along the trajectory.

The resulting 2D LEC plot for Utah landing site are shown in Figure 6 for the CP case and in Figure 7 for the EP case: each curve represents a constraint and the grey region is the feasible area in terms of FPA and β.

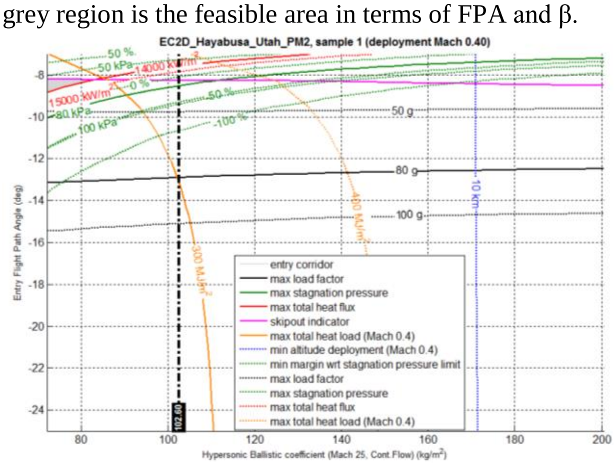

Fig. 6: Resulting 2D LEC for Utah CP scenario

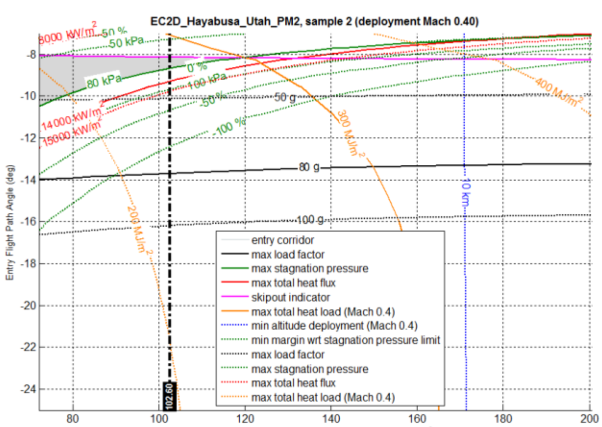

Fig. 7: Resulting 2D LEC for Utah EP scenario

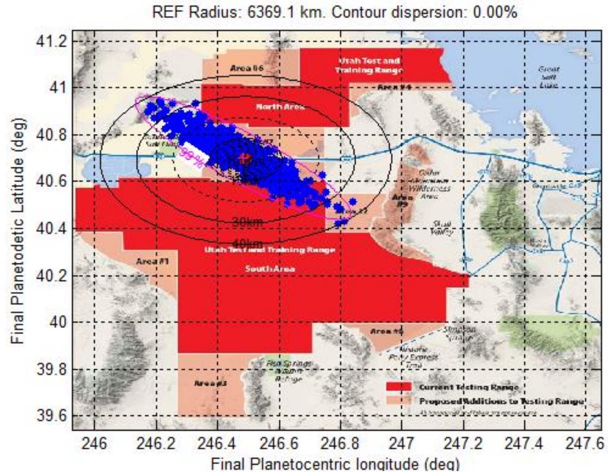

Fig.8: Dispersion at touchdown, Utah CP scenario

The main conclusions are that an entry corridor always exists for Woomera (both for EP and CP inbound) as well as for Utah EP return; a reduction of the ballistic coefficient or an increase of the capsule diameter is needed for Utah CP return.

A 3 DOF Monte Carlo Analysis has been carried out for the EDL assessment; the performances in terms of position and velocity dispersion at touchdown are in line with the requirements for all scenarios (see Figure 8).

3. The Spacecraft Design

The MSR spacecraft design is highly coupled with the mission analysis results, especially to reflect the propulsion system selection and the staging strategy implementation. When considering the four principal manoeuvres (MOI, TOA, DOA and TEI), with:

- two possible propulsion systems (CP or EP);
- a different number of EP engines (one or two);
- the possibility of including the aero-braking;
- different points in time to perform the staging;

it is clear that the number of possible solutions is very high. An extensive analysis has then been conducted at timeline and system level to verify the fulfilment of the mission goals for more than 500 configurations.

A first filter is represented by the timeline feasibility: given the trajectories analysis presented in the previous sections, the different mission phases durations are defined by the employed propulsion system, and it is possible to calculate the LMO acquisition date and the latest departure from Mars to fulfil the goal of returning the sample back to Earth by 2029. A mission scenario is then considered feasible only if it allows sufficient time in LMO to perform all the RdV operations with the associated margins.

The solutions which have passed this timeline filter are detailed in Figure 9 and will undergo the mass budget selection described in the next sessions. But before that, the payload needs to be fully characterised.

Engine		Option	Maneuver MOI	TOA	DOA	TEI
HTAE	x1	1	Chemical	Chemical	Chemical	Chemical
HTAE	x1	17	Chemical	Aerobraking	Chemical	Chemical
T6	x2	2	Chemical	Chemical	Chemical	Electric
T6	x2	9	Electric	Chemical	Chemical	Chemical
T6	x2	10	Electric	Chemical	Chemical	Electric
T6	x2	18	Chemical	Aerobraking	Chemical	Electric
ARM	x1	2	Chemical	Chemical	Chemical	Electric
ARM	x1	3	Chemical	Chemical	Electric	Chemical
ARM	x1	5	Chemical	Electric	Chemical	Chemical
ARM	x1	6	Chemical	Electric	Chemical	Electric
ARM	x1	9	Electric	Chemical	Chemical	Chemical
ARM	x1	10	Electric	Chemical	Chemical	Electric
ARM	x1	11	Electric	Chemical	Electric	Chemical
ARM	x1	13	Electric	Electric	Chemical	Chemical
ARM	x1	14	Electric	Electric	Chemical	Electric
ARM	x1	18	Chemical	Aerobraking	Chemical	Electric
ARM	x1	19	Chemical	Aerobraking	Electric	Chemical
ARM	x1	21	Electric	Aerobraking	Chemical	Chemical
ARM	x1	22	Electric	Aerobraking	Chemical	Electric
ARM	x2	2	Chemical	Chemical	Chemical	Electric
ARM	x2	3	Chemical	Chemical	Electric	Chemical
ARM	x2	4	Chemical	Chemical	Electric	Electric
ARM	x2	5	Chemical	Electric	Chemical	Chemical
ARM	x2	6	Chemical	Electric	Chemical	Electric
ARM	x2	7	Chemical	Electric	Electric	Chemical
ARM	x2	8	Chemical	Electric	Electric	Electric
ARM	x2	9	Electric	Chemical	Chemical	Chemical
ARM	x2	10	Electric	Chemical	Chemical	Electric
ARM	x2	11	Electric	Chemical	Electric	Chemical
ARM	x2	12	Electric	Chemical	Electric	Electric
ARM	x2	13	Electric	Electric	Chemical	Chemical
ARM	x2	14	Electric	Electric	Chemical	Electric
ARM	x2	15	Electric	Electric	Electric	Chemical
ARM	x2	16	Electric	Electric	Electric	Electric
ARM	x2	18	Chemical	Aerobraking	Chemical	Electric
ARM	x2	19	Chemical	Aerobraking	Electric	Chemical
ARM	x2	20	Chemical	Aerobraking	Electric	Electric
ARM	x2	21	Electric	Aerobraking	Chemical	Chemical
ARM	x2	22	Electric	Aerobraking	Chemical	Electric
ARM	x2	23	Electric	Aerobraking	Electric	Chemical
ARM	x2	24	Electric	Aerobraking	Electric	Electric
NEXT-C	x2	2	Chemical	Chemical	Chemical	Electric
NEXT-C	x2	5	Chemical	Electric	Chemical	Chemical
NEXT-C	x2	6	Chemical	Electric	Chemical	Electric
NEXT-C	x2	9	Electric	Chemical	Chemical	Chemical
NEXT-C	x2	10	Electric	Chemical	Chemical	Electric
NEXT-C	x2	18	Chemical	Aerobraking	Chemical	Electric
NEXT-C	x2	21	Electric	Aerobraking	Chemical	Chemical
NEXT-C	x2	22	Electric	Aerobraking	Chemical	Electric
PPS5000	x2	2	Chemical	Chemical	Chemical	Electric
PPS5000	x2	3	Chemical	Chemical	Electric	Chemical
PPS5000	x2	4	Chemical	Chemical	Electric	Electric
PPS5000	x2	5	Chemical	Electric	Chemical	Chemical
PPS5000	x2	6	Chemical	Electric	Chemical	Electric
PPS5000	x2	9	Electric	Chemical	Chemical	Chemical
PPS5000	x2	10	Electric	Chemical	Chemical	Electric
PPS5000	x2	11	Electric	Chemical	Electric	Chemical
PPS5000	x2	12	Electric	Chemical	Electric	Electric
PPS5000	x2	13	Electric	Electric	Chemical	Chemical
PPS5000	x2	14	Electric	Electric	Chemical	Electric
PPS5000	x2	18	Chemical	Aerobraking	Chemical	Electric
PPS5000	x2	19	Chemical	Aerobraking	Electric	Chemical
PPS5000	x2	20	Chemical	Aerobraking	Electric	Electric
PPS5000	x2	21	Electric	Aerobraking	Chemical	Chemical
PPS5000	x2	22	Electric	Aerobraking	Chemical	Electric

Fig. 9: Considered solutions after timeline selection

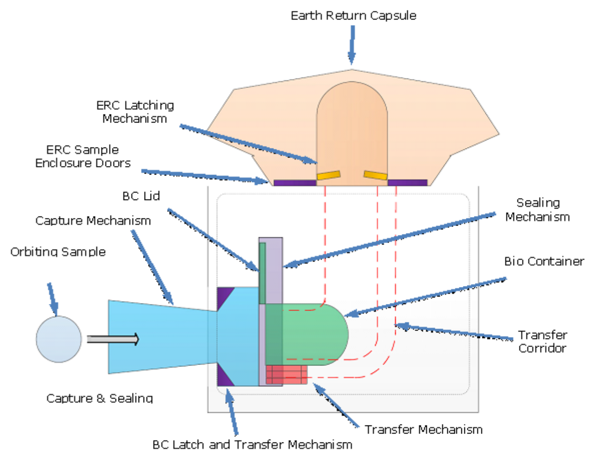

Fig.10: System Level Mission Payload Overview

3.1 Payload

The concept of operations for the payload elements has been identified to provide the required input to the mass budget analysis described in the next section: Figure 10 show the pre-capture system level configuration of the payload and Table 9 the mass detail of its elements.

3.2 Mass budget analysis

A full set of results is then tabulated in Table 10 and colour coded thusly:

- **>9000kg**
- **<9000kg** (performance of Delta IV Heavy)
- **<5500kg** (predicted performance of Ariane 64)
- **<5000kg** (performance of Atlas 551)

Staging options are considered for MOI, TOA and DOI and are abbreviated with a “Y” or a “N” whether the staging is applied to each manoeuvre or not (i.e. “YYY” is 3 staging, “NNN” is no staging). Six promising candidates (<5500kg) are identified, when the CSS is ejected:

1. Option 17 CP NYN 4768kg
2. Option 7 1ARM NNY 5066kg
3. Option 7 1ARM YNY 4923kg
4. Option 19 2ARM NNY 5381Kg
5. Option 20 2ARM NNN 5493kg
6. Option 20 2ARM NNY 5136kg

An example of spacecraft configuration for option 17 (full CP with aero-braking) is shown in Figure 11.

Element	MASS [kg]
Orbiting Sample (OS)	12.0
Capture and Sealing System (CSS)	177.8
Bio-Container (BC)	52.0
Transfer Mechanism (TM)	/
Earth Return Capture (ERC)	191.4
Docking Mechanism (DM)	/
Total	**433.2**

Table 9: Payload Element Mass with 30% margins

Option	1 Stage	2 Stage	3 Stage	4 Stage
1-CH	14355-NNN	8966-YNN		
17-CH	5702-NNN	4768-NYN		
2-T6	15302-NNN	9645-NYN	9416-YYN	
9-T6	14181-NNN	8163-NYN	7886-YYN	
10-T6	9511-NNN	8011-NYN	7734-YYN	
2-ARM	21162-NNN	13137-YNN		
3-ARM	10122-NNN	7889-NYN	7386-YNY	
5-ARM	10550-NNN	7088-NYN		
6-ARM	9195-NNN	7706-NYN	7526-NYY	
7-ARM	5677-NNN	5066-NNY	4923-YNY	
9-ARM	19943-NNN	11649-YNN		
10-ARM	16194-NNN	10332-NYN	9708-YYN	
11-ARM	8491-NNN	6975-NNY	6901-YNY	
13-ARM	8801-NNN	6191-NYN		
14-ARM	7762-NNN	6774-NYN	6678-NYY	
18-ARM	7679-NNN	6554-NYN	6436-YNY	
2-2ARM	26187-NNN	16075-YNN		
3-2ARM	12301-NNN	9514-NYN	8817-YNY	8712-YYY
4-2ARM	10871-NNN	8618-NYN	8403-YYN	
5-2ARM	12871-NNN	8313-NYN	8260-NYY	
6-2ARM	11307-NNN	9400-YNN	9302-NYY	9236-YYY
7-2ARM	6900-NNN	6044-NNY	5861-YNY	
8-2ARM	6292-NNN	5792-YNN		
9-2ARM	34202-NNN	17796-YNN		
10-2ARM	26603-NNN	16432-YNN		
11-2ARM	12273-NNN	9470-NNY	9146-YNY	9039-YYY
12-2ARM	10826-NNN	9433-YNN	8739-YYN	
13-2ARM	12855-NNN	8095-NYN	8044-NYY	
14-2ARM	11270-NNN	9294-NYN	9075-NYY	
15-2ARM	6838-NNN	5835-NNY		
16-2ARM	6059-NNN			
18-2ARM	9381-NNN	7942-NYN	7753-NYY	
20-2ARM	5493-NNN	5136-NYN		
22-2ARM	9326-NNN	8411-NYN	8108-YNY	
2-NEXT-C	18771-NNN	11019-NYN	10702-YYN	
5-NEXT-C	9660-NNN	6767-NYN		
6-NEXT-C	7600-NNN	6770-YNN		
9-NEXT-C	18021-NNN	9364-NYN	8815-YYN	
10-NEXT-C	12775-NNN	9830-NYN	9300-YYN	
13-NEXT-C	7356-NNN	5630-NYN		
14-NEXT-C	5954-NNN			
18-NEXT-C	7512-NNN	6560-NYN		
2-PPS5000	24643-NNN	14499-YNN		
3-PPS5000	11478-NNN	8520-NYN	8021-YNY	7844-YYY
4-PPS5000	10748-NNN	7948-NYN	7748-YYN	
5-PPS5000	12060-NNN	7561-NYN		
6-PPS5000	11253-NNN	8708-NYN	8512-NYY	8458-YYY
7-PPS5000	6783-NNN	5836-NNY	5572-YNY	
8-PPS5000	6456-NNN	5713-YNN		
9-PPS5000	37259-NNN			
10-PPS5000	31784-NNN	19051-YNN		
11-PPS5000	12977-NNN	9524-NNY	8698-YNY	8504-YYY
12-PPS5000	12069-NNN	9190-NYN	8415-YYN	
13-PPS5000	13735-NNN	7830-NYN		
14-PPS5000	12718-NNN	9022-NYN	8759-NYY	
18-PPS5000	8171-NNN	6901-NYN	6705-NYY	
18-T6	6463-NNN	5768-NYN		
21-ARM	9299-NNN	6554-NYN		
22-ARM	9764-NNN	7988-NYN	7793-YNY	7693-YYY
19-2ARM	5980-NNN	5381-NNY		
21-2ARM	10461-NNN	7365-NYN	7179-YNY	
21-NEXT-C	9055-NNN	6534-NYN		
24-2ARM	7801-NNN	7503-YNN	7188-YYN	
22-NEXT-C	7523-NNN	7095-NYN	7043-YNY	
21-PPS5000	12666-NNN	7710-NYN	7464-YNY	
22-PPS5000	12649-NNN	9184-NYN	8732-YNY	

Table 10: Mass budget results (with CSS ejection)

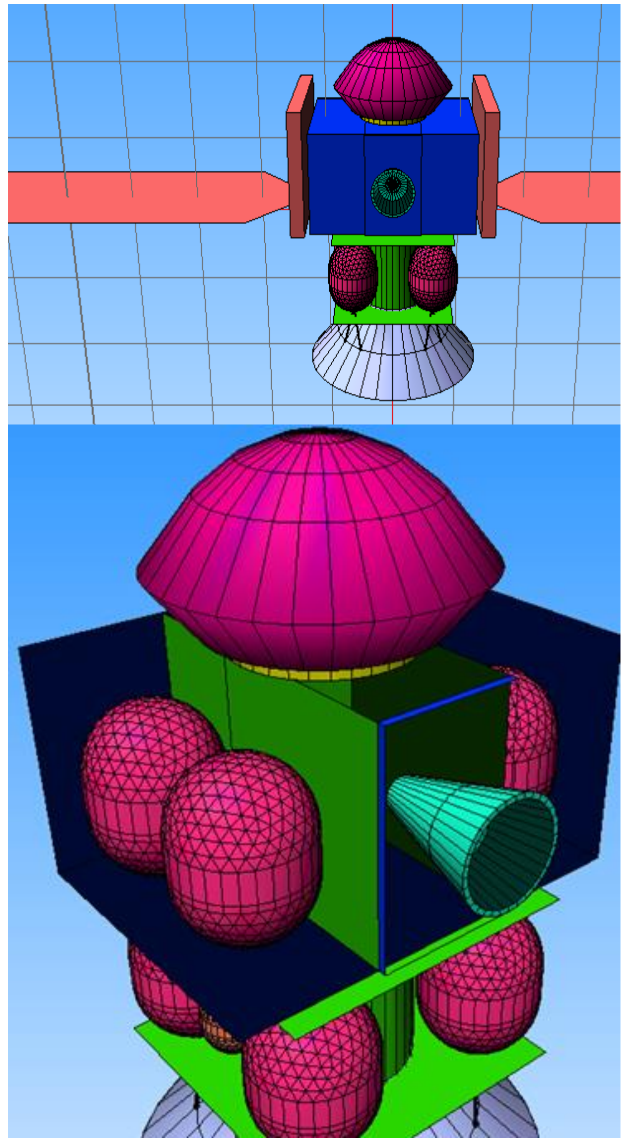

Fig. 11: Preliminary configuration for Option 17 full CP mission with aero-braking

4. Mission Development Plan And Risk & Opportunity Analysis

The mission development schedule has been analysed in detail at mission, subsystem and technology level. On the base of this investigation, the following Mission Development Plan (MDP) is proposed in Fig 12. The plan includes an early phase for technology development, and covers Phase A through to the phase of operations, Phase E, as described in ECSS-M-30A[11]. Also included are the major review milestones and the 2026 launch window epoch. It can be seen that for the baseline schedule, the Launch Readiness Review (LRR) is just ahead of the 2026 launch window providing a margin of ~5 months. Critical path analysis would be required in future studies to assess whether this level of schedule margin is sufficient.

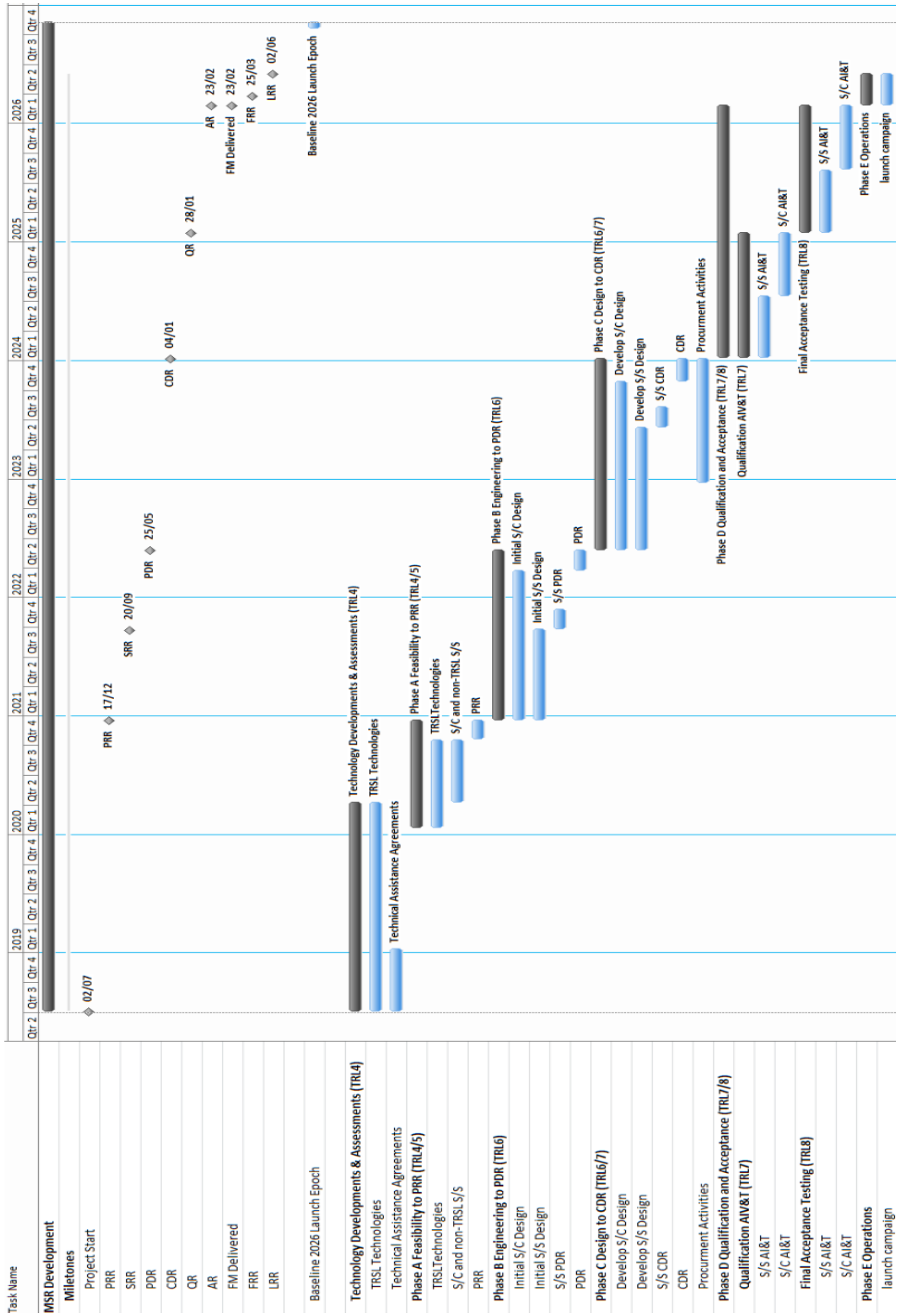

Fig. 12: Proposed Development Schedule

A Relative Risk and Opportunity Assessment (RROA) has been conducted to understand the relative risk and opportunity merits associated with different architecture options. During this assessment, two main categories of major risks/opportunities were considered:

1. Major specific risks and opportunities for the MSR mission (i.e. not normal satellite risks), which must be highlighted to focus the development of relevant elements as required.
2. Discriminator risks and opportunities between architecture options (i.e. risks whose severity and/or likelihood are dependent on what type of engine is used, whether aero-braking is specified, the level of staging).

Table 11 summarises both schedule impact and risk/opportunity analysis for the most promising options emerged from timelines and mass budget calculations.

From these results, a clear observation is that all-chemical options are seen to be far less risky than electric propulsion ones. This is primarily because of the relative associated risk surrounding the development of EP systems and the large solar arrays needed to supply their power demands at Mars[1]. The architecture options using a 2xARM EP system are seen as slightly riskier than other electric propulsion options because the solar arrays required are considerably larger.

Architecture Option	Wet Mass [kg]	Schedule Impact [months]	RROA Value [-]
#17 - HTAE - NYN - CSS Eject	4768	0	55
#17 - ARM - NYN - CSS Retain	5306	0	51
# 7 - ARM - NNY - CSS Eject	5066	2	88
# 7 – ARM - NNY - CSS Retain	5307	2	84
# 7 - ARM - YNY - CSS Eject	4923	2	88
# 7 - ARM - YNY - CSS Retain	5134	2	83
#19 – 2xARM - NNY - CSS Eject	5381	5	104
#20 - 2xARM - NNN - CSS Eject	5493	5	105
#20 - 2xARM - NYN - CSS Eject	5136	5	103
#20 - 2x ARM - NYN - CSS Retain	5318	5	99
#13 - NEXT-C - NYN – CSS and SA Eject	5074	5	87
# 7 - PPS5000 - YNY – CSS and SA Eject	5462	2	89

Table 11: RROA of promising architecture options

5. Proposed Solutions

Among the feasible options that emerged from the mass budget analysis, option 7 (CP-AB-EP-CP) and option 13 (EP-EP-CP-CP) result to be unfeasible at timeline level: in fact, they both violate some timeline constraint once the EP performances are calculated with the spacecraft detailed wet mass:

- Option 7 shows an overlap between the aero-braking duration to acquire LMO and the spiralling time needed to depart from it towards the interplanetary return journey. The only possibility to allocate both phases would be to postpone the Earth arrival after 2030.
- Option 13 does not offer any feasible outbound transfer, since the high spacecraft wet mass makes unfeasible any short Earth-Mars transfer arriving at the Red Planet in 2027

The only EP system able to accommodate the short transfers imposed by the tight timeline constraints has then to rely on the use of 2 ARM engines, and the three feasible options both at mass budget and timeline level result to be:

- Option17, full CP mission with aero-braking
- Option19 (CP-AB-EP-CP) with 2 ARM engines
- Option20 (CP-AB-EP-EP) with 2 ARM engines

The three of them make use of both staging and aero-braking to reduce the spacecraft wet mass, and they all rely either on chemical propulsion or on a mixed system combining a chemical engine with 2 very powerful electrical thrusters (ARM engines). This result is the consequence of the tight timelines imposed on the mission which do not leave any room to less powerful and more efficient electrical propulsion systems. To accommodate such a demanding plan, the mission design is forced to assume certain risks like associated with staging and aero-braking techniques. Their main mission parameters are summarised in Table 12.

SOLUTION	# 17	# 19	# 20
EP engine	/	2xARM	2xARM
Staging	NYN	NNY	NYN
Wet Mass	4768 kg	5381 kg	5136 kg
Launch date	01/11/2026	01/11/2026	01/11/2026
MOI	CP 03/09/2027	CP 03/09/2027	CP 03/09/2027
TOA	AB 06/08/2028	AB 12/03/2028	AB 21/04/2028
DOA	CP 06/09/2028	EP 12/04/2028	EP 22/05/2028
TEI	CP 11/09/2028	CP 11/09/2028	EP 03/10/2028
Earth arrival	03/06/2029	03/06/2029	22/07/2029

Table 12: Feasible mission options summary

It is important to notice that option 19 could also present some criticality at timeline level: since it relies on a chemical inbound transfer, it shall leave Mars at the beginning of September and, to accommodate the EP departure spiralling, the critical RdV operations are pushed in the middle of the 2028 solar conjunction; on the contrary, the use of EP for the inbound arc of solution 20, allows a Mars departure in October and the RdV could happen after the end of the mentioned solar conjunction. Both options shall in any case reduce the aero-braking duration to 5 months: this value is compatible with the foreseen perigee reduction for Bc around 30 kg/m^2, fully in line with the expected value emerged at system level design.

The three options are then qualitatively compared with respect to four figures of merit:

1. Wet mass
2. Timeline risk.
3. Schedule delays
4. Overall risk

To provide a single overview of the study results, the proposed figures of merit are then graphically presented in Figure 13, where a lower value is a better performance index (lines closer to the centre are then preferable).

6. Summary and Concluding Remarks

This paper presents the architecture assessment study for the proposed mission European Return Orbiter as part of Mars Sample Return mission. The overall programme exhibits a very ambitious goal, such us the return of several Mars soil and atmosphere samples to Earth, which leads to a complex mission design configured in a multi-module solution involving ESA and NASA with three different spacecraft. The current study focused on the architecture definition of the ERO mission, in charge to travel to Mars, rendezvous with the OS launched from Mars surface and bring it back to Earth before 2030.

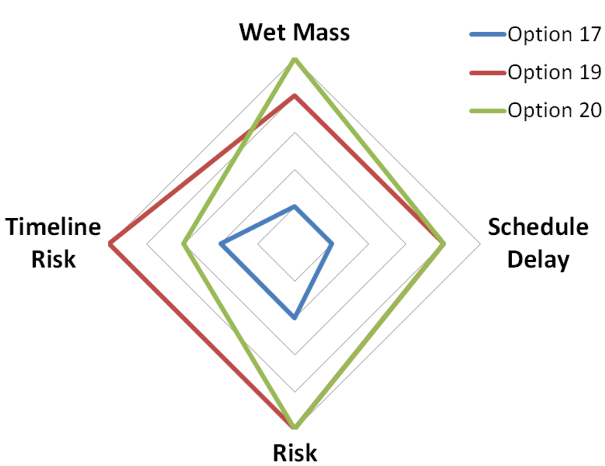

Fig. 13: Comparison of the three feasible solutions

The study has been carried-out by an industrial consortium composed by DEIMOS Space S.L.U. as prime contractor and responsible for the mission design, Lockheed Martin UK Ampthill for the system design, mass budget and risk analysis and MDA Corporation for the payload mechanisms. During the course of the study a significant effort has been dedicated to the explorations of a very high number of configuration architectures to consider the combination of different propulsion systems for the different mission phases and to take into account several possibilities for the staging sequence. As a result several hundreds of options have been analysed with a high level of detail both at mission analysis and mass budget level, to properly identify robust and feasible solutions.

The three most promising options make all use of both staging and aero-braking to reduce the spacecraft wet mass, and rely either on chemical propulsion or on a mixed system combining a chemical engine with 2 very powerful electrical thrusters (ARM engines). This result is the consequence of the tight timelines imposed on the mission which do not leave any room to less powerful and more efficient electrical propulsion systems. To accommodate such demanding plan, the mission design is forced to assume certain risk like the ones associated with staging and aero-braking techniques.

The outcome of this study shall be considered as a preliminary architecture assessment to be further refined through the currently ongoing detailed phase-A study for MSR-ERO mission.

7. Acknowledgment

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