Long Term Cryogenic Propulsion for Interplanetary Missions

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Interplanetary missions impose high velocity increment requirements and complex trajectory strategies; the use of chemical storable propellants implies limited specific impulse and consequent high propellant mass.

The state of the art of cryogenic storage technologies is now mature to consider such systems as possible alternatives to storable propellants for such applications.

The combined usage of advanced power sources such as fuel cells and very high specific impulse systems such as electric propulsion could provide a further advantage.

A fast track technology feasibility study was performed at ESA/Estec Concurrent Design Facility (CDF) adopting a concurrent engineering approach, with the aim to study the potential of such technologies.

A selection of potentially attractive and representative missions was performed; soon the use of fuel cells as power source was found not applicable when considering the fuel cells as a primary power source or as a secondary power source during specific mission phases.

Electric propulsion concepts were as well excluded from the study due to the specific system requirements imposed by their high power demand.

Study baseline was full cryogenic propulsion concept adopting LH2/LOX as propellants.

Sample missions selection was performed bearing in mind general mission cost aspects: adoption of enabling technologies already under development in Europe to limit non recurring costs and use of small-medium launcher to challenging missions that would not meet the velocity increment requirements with traditional storable propellant systems. The system design cases studied revealed a good advantage versus storable propellants but still the most challenging missions were hardly reachable in terms of compliance with the reference target P/L masses due to the complexity of structures and thermal systems. With the same boundary conditions, a liquid oxygen/hydrocarbon propellant solution was preliminary analyzed; in spite of the reduced mass specific impulse, the increased volume specific impulse and reduced thermal control needs provided sensible advantages, provided attractive results versus those obtained with storable propellant solutions.

Nomenclature

<table>
<thead>
<tr>
<th>Acronym</th>
<th>Description</th>
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<tbody>
<tr>
<td>AOCs</td>
<td>Attitude Orbit Control System</td>
</tr>
<tr>
<td>AU</td>
<td>Astronomical Unit</td>
</tr>
<tr>
<td>CDF</td>
<td>Concurrent Design Facility</td>
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<tr>
<td>CFRP</td>
<td>Carbon Fiber Reinforced Polymer</td>
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<tr>
<td>CSG</td>
<td>Centre Spatial Guyanais</td>
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<tr>
<td>DSR</td>
<td>Deimos Sample Return</td>
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<tr>
<td>EOI</td>
<td>Europa Orbit Insertion</td>
</tr>
<tr>
<td>GTO</td>
<td>Geostationary Transfer Orbit</td>
</tr>
<tr>
<td>HGA</td>
<td>High Gain Antenna</td>
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<tr>
<td>Isp</td>
<td>specific impulse</td>
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I. Introduction

Cryogenic propulsion is adopted for space transportation purposes due to the well known advantages brought by the highest Isp reachable by chemical propulsion concepts and the consequent advantages in relation to the very high ΔV demands.

On the other hand, planetary transfer missions have so far seen the exclusive utilization of storable propellants (i.e. MMH/MON) due to their versatility, proven maturity status and acceptable performance.

This category of missions, of interest for exploration and science applications, is therefore limited in terms of maximum P/L and furthermore very often it has to account for the utilization of heavy launchers with proportionally heavy cost impacts on the overall mission cycle.

To assess the suitability of cryo propulsion concepts to such missions it is necessary to evaluate the key system aspects in conjunction with the relevant technological needs, in light of the current SoA and projected future developments. The fast track concurrent engineering study conducted at ESA/Estec Concurrent Design Facility had the aim to study both system concepts and technology challenges.

The purpose of this paper is to present the major findings of the investigations.

In a preliminary phase of the activities, attractiveness of combination of cryogenic technologies with fuel cell advanced power production was assessed with the main aim to integrate the fuel cells with solar panels during specific mission phases to provide power to the cryogenic system and/or to the spacecraft itself. This idea revealed to provide a level of complexity not compensated by the corresponding advantages; the main power source could not practically be replaced and solar panels have to be regarded as the baseline among the power source concepts available currently and for the next years to come.

The core phase of the study was dedicated to the system analysis of a number of potentially attractive missions, taken the source of the main system requirements, and the conceptual design of the relevant spacecraft.

II. Sample missions

Sample missions were selected among realistic study cases currently being investigated either for science or exploration applications. This approach enabled study start up with realistic spacecraft concepts and parameters, saving time and effort on spacecraft design and allowed to avoid idealized LTCP concepts that could disregard specific mission requirements and constraints. Moreover, a direct comparison with LTCP and storable propellants combination missions was possible since the latter constitutes the baseline propulsion concept.

Care was taken in the selection of mission profiles that included various thermal conditions, both in terms of general transfer strategy and spacecraft orientation with respect to heat exchange sources. Consequently, thermal aspects of trajectory phases towards solar system inner and outer planets were analyzed. Planet swing-by phases were studied taking into account the heat entries on critical spacecraft elements elaborated through optimization of the spacecraft orientation along the trajectory, correlations with AOCS needs and constraints were made that allowed preliminary definition of control system requirements.

All the missions were assumed to be launched from CSG Kourou, the European launch base, using a medium size launcher (Soyuz) able to perform orbit insertion of a suitable spacecraft either in LEO or GTO.
With the described criteria, the following potential sample missions were assessed:

- **Far planet mission: Neptune orbiter**
  This deep space mission requires very long trip time to target and imposes challenging thermal control needs, this study case is not feasible with SoA power sources (i.e. solar arrays and batteries) and fuel cell hydrogen demand would be unrealistic, it is considered not of interest for LTCP applications.

- **Fast space telescope delivery: L2 point**
  A faster transfer and increased payload to target could make this mission attractive also with a limitation in the instruments Helium boil off, resulting in increased science operations time. In practice, cruise time of this category of missions is already short (few weeks), the potential gain of LTCP adoption is then limited.

- **Jupiter moon (Europa) explorer (JME)**
  The Jovian minisat explorer mission is currently under investigation; the system configuration assumes storable propellants propulsion. The LTCP transfer/manoeuvre module was studied keeping all the defined spacecraft characteristics, including the final payload target mass.

  The challenging points of this mission are the power availability for LTCP, the long transfer time that challenges the thermal control design and the need of Venus swing-by which as well imposes careful definition of thermal control provisions and general spacecraft architecture.

- **Mars moon (Deimos) sample return (DSR)**
  The Deimos sample return mission is as well under investigation with storable propellants. As per the Jovian case the general spacecraft definition was kept. Compared to the Jovian case, it appears less constraining for LTCP since no swing-by and no power availability limits are present. However its interest is related to commonalities to more general Mars exploration missions.

For the purpose of the study, JME\(^1\) and DSR\(^2\) were the selected sample missions, in the following respectively marked as Design Case 1 and 2.

In Figure 1 and Figure 2, the general layout of the reference JME and DSR spacecrafts are reported, they constituted the starting point for the LTCP investigations.

The reference mission scenario for Design case 1: JME foresees two spacecraft launch options: LEO and GTO, launch mass about 8000 and 3000 kg, LTCP performs all manoeuvres up to and including Jupiter insertion and Europa insertion. Gravity assist strategy is Venus-Earth-Earth, transfer time to Jupiter is 6 years.

The reference mission scenario for Design case 2: DSR foresees two spacecraft launch options: LEO and GTO, launch mass about 8000 and 3000 kg, LTCP performs all manoeuvres up to and including Deimos arrival, Earth return mission is performed by storable propulsion system. Gravity assist is not necessary for this scenario, distance to Sun is \(>1\) AU, transfer time to Deimos is 1 year plus up to 1 year of stay time in the Mars-Deimos system.
III. System trades and assumptions

In this chapter the main trades performed to identify system characteristics of the selected missions will be presented.

Before analyzing the specific results obtained it is important to highlight that the sample design cases are intended to be means for the investigation of LTCP technologies, therefore the presented results should not be understood as valid to assess feasibility of any missions nor as an exhaustive means to finally evaluate general feasibility of LTCP technologies.

The following system trade offs and assumptions were performed.

The LTCP propellant masses for both design cases were found very similar and equivalent to about 6000 kg, this fact allowed high level of commonality between the two cases. The LTCP was then conceived as a common module, separated from the main spacecraft in order to approach the conceptual design of a common planetary transfer platform, in principle applicable to several mission cases.

To minimize gravity losses, the LTCP module is bound to perform high thrust burns for the Earth escape and first kick to the target planet; towards the final mission phases, the reduced overall spacecraft mass due to the majority of the propellant consumed suggests a much reduced thrust in order to keep up with the maximum allowed thrust-to-weight ratio. This parameter is constrained by solar panels structural static and dynamic performances. Structural analyses and numerical simulations on advanced solar array drive mechanisms provide boundary conditions on thrust-to-weight ratio time history during the mission, leading to definition of maximum allowable thrust levels. As a conclusion of the thrust level trades it was found the maximum total thrust allowed in the worst case (mission final phases) is about 400 N while during Earth escape manoeuvres 1200 N total thrust provides a reasonable balance between gravity losses and spacecraft thrust-to-weight ratio.

Concerning comparison between the two design cases, JME is certain the worst case due to the larger solar panels surface required, whereas DSR imposes much lower constraints in this respect, however taking into account other possible robotic missions to Mars have demand standards very similar to JME, the latter was used as reference for both design cases.

The following table summarizes the thrust level cases during the mission:

<table>
<thead>
<tr>
<th>Thrust [N]</th>
<th>Impulsive DeltaV [m/s]</th>
<th>Comments</th>
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<tbody>
<tr>
<td>400</td>
<td>606</td>
<td></td>
</tr>
<tr>
<td>600</td>
<td></td>
<td></td>
</tr>
<tr>
<td>1200</td>
<td></td>
<td></td>
</tr>
<tr>
<td>1500</td>
<td></td>
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</table>

The worst case due to the larger solar panels surface required, whereas DSR imposes much lower constraints in this respect, however taking into account other possible robotic missions to Mars have demand standards very similar to JME, the latter was used as reference for both design cases.
The second system trade was relevant to the propulsion concept and architecture.

Taking as a starting point the relatively low thrust level requirement, the adopted approach is to minimize the chamber pressure in order to keep low engine design and operational complexity, the nominal required chamber pressure is 6.7 bar.

Pressure fed and electric pump fed propellant pressurization approach were traded. Taking into account the propellant volume requirement and the projected tanks MEOP, the resulting tank structural masses make the pressure fed solution unfeasible.

From a functional point of view the electric pump solution was considered feasible since the needed pump head corresponding to nominal engine inlet pressure implies affordable power needs, details in the propulsion chapter.

The Zero Boil Off concept was analysed and traded versus a boil off approach during the whole spacecraft mission.

Since in general the overall heat entries to the spacecraft critical areas, typically the hydrogen storage, highly depend on the shielding design adopted, it was found that the spacecraft architectural choices and the spacecraft orientation strategy during the mission play a very important role.

Consequently, a high level architectural constraint was adopted on the spacecraft design to limit the complexity of the thermal control system and the masses of the relevant provisions; the hydrogen tank is shielded by the oxygen vessel and an additional panel shielding with 3.7 m diameter is included between the LTCP module and the payload; the warm spacecraft areas are structurally and thermally linked to the oxygen tank in order to create a constant decreasing thermal gradient towards the hydrogen tank.

With a suitable orientation strategy, the maximum heat entries on hydrogen tank were very limited, excluding the Venus swing-by phase where hydrogen and oxygen are let boil off for the short duration (4 days) of the heat-critical flight in Venus vicinity.

It is worth to note that with the designed spacecraft architecture, assuming full boil off during the mission, the worst case calculated hydrogen boiled off would be about 140 Kg, corresponding to 2 m³ tank volume. Purely in terms of hardware mass, the thermal control provisions and related hardware needed to compensate the calculated heat entries for ZBO is estimated about 70 Kg.

The net mass saving with ZBO has to be assessed including the mass impact of the increased tank volume and the overall trade analysis has to take into account the added complexity and technology needs related to ZBO implementation.

The spacecraft assembly configuration studies had to take into account all the above mentioned aspects and were directed to allow a positive assessment of the assembly structural masses; several options were investigated, in Figure 5 and Figure 6 report the two main configurations analyzed.

Due to its relatively high specific impulse, initial focus was on the combination of liquid oxygen and liquid hydrogen, LOX/LH₂, as propellant. Later in the study the use of liquid methane, LCH₄, was studied as well, to check whether its higher density (smaller fuel tank) and higher storage temperature (less thermal constraints) would outweigh the disadvantage of a lower performance versus the use of liquid hydrogen (lower Isp of LOX/LCH₄) and results in advantages over the use of LOX/LH₂ LTCP. In Figure 7 the resulting configuration is shown, to be noted in this case a common bulkhead tank is affordable due to similar propellant boiling temperatures.

Filling and cooling of the propellant tanks on the launch pad would be required, which means that disconnectable propellant loading and cooling pipes would have to be fitted through the launcher fairing. It was assumed the launch pad and fairing would be adapted to fit LTCP loading equipment for the sample mission cases studied.

During launch no power for cooling of the LTCP propellant, and specifically LH₂, can be provided, while the spacecraft would experience relatively high aero thermodynamic loads. It was deemed possible to load the LH₂ tank with a mix of liquid and ice (slush), whereby the melting ice would provide cooling for the duration of the launch and until the solar arrays are opened.
The attitude of a spacecraft with LTCP will be constrained by both the need to minimise the thermal loads on the cryogenic propellant and the need to be able to point the solar arrays to receive sufficient levels of sunlight. It is therefore very likely that LTCP missions with High Gain Antennas (HGA) will require antenna pointing mechanisms, they were therefore included as part of the design.

Most of the LTCP propellant is used for Earth escape (between 60 and 80% depending on the design case), which means during most of the mission the LTCP tanks are more than half empty. Especially for LH2 the empty volume in the tank will be large, which will make it more complicated to ensure a proper propellant flow when the thrusters need to restart in microgravity. Propellant management technology for LTCP tanks will be a critical development required.

IV. System results

A. General
The two main cases are JME and DSR mission. Both cases are split into a Soyuz launch into LEO without the Fregat upper stage (option 1) and an insertion into GTO utilising Fregat (option 2). Both options are then split again into two alternative tank designs. Alternative “a” considers a bulkhead configuration with in fact the two tanks bulkheads are separated by a vacuumed volume, whereas alternative “b” represents fully separated tanks, subsystem details are reported in structures section.
In option 1c the results of the investigated Methane case (LCH4) replacing LH2 is shown. A common bulkhead tank configuration is the baseline in that case.
In Appendix the studied cases and options are summarized for reference.

B. Study case1: JME
The JME mission concept is based on storable propellants, and has the objective to be launched by Soyuz from the European Space Port in Kourou (French Guyana). A recent review of the mission definition pointed out that in its current conception JME is unfeasible, as its mass is exceeding by 780 kg the launcher P/L capability.
If LTCP could be used to enable the launch of JME with a Soyuz, this would clearly show a major benefit of the use of LTCP for Jupiter and more in general for interplanetary science missions.

The overall spacecraft mass budget is reported in in Figure 8 where elements 1 and 2 stand for the payload masses while element 3 reports LTCP module total mass. The LTCP module includes all deltas with respect to the baseline.

For all the following mass budget summaries, the launch mass target considers the adapter (specific to each mission) to remain on the launcher, which means, the actual target launch mass is i.e., 8400 kg plus adapter mass.

The launch target excess is minimal for option 1-c being 173.79 kg and thus 77% below the baseline design. The LH2 options show all a significant lower reduction of the mass excess. Generally the option b results in an overall better improvement against the baseline. All results are superior with respect to the baseline.

Reviewing the mass breakdown of the three elements identifies the structures as the main

mass contributor. The second and third contributors are the thermal and power domains including all relevant components, i.e. structural heat shield, coolers etc.
Since the structures have been clearly identified as the main mass contributors, increasing relatively in the same order as the mass benefit with respect to the original JME baseline, it is the domain where effort in reducing mass will bring the highest benefit.

To reduce the mass, a CFRP design of the tanks has been considered resulting in significantly lower structural mass figures. Figure 9 gives the resulting total mass budget summary of the entire stack.

Although the structural mass of the design shows very high values it has been shown that the actual mass distribution of the subsystems is comparable to that of the baseline mission.

The launch mass target enabling a Soyuz launch can not be met; however, it is possible to reduce the excess significantly by 30-50% using conventional tank design and LOX/LH2 as propellant.

Involving advanced materials in the tank design (CFRP) results in a total reduction of the excess by up to 90% with respect to the baseline.

Using the carbohydrate Methane (CH4) as propellant instead of LH2, results in major relaxations of the structural and thermal design constraints (lower volume, storage, etc.) which is reflected in the significant lower total mass of the LTCP element.

Using CFRP tanks and Methane, the launch mass excess can be reduced by up to 90% as well.

The remaining total mass excess in this case, of about 80 kg, has to be seen in respect to the fact that the elements 1 and 2 have been scaled only on systems level! A detailed design of this specific mission could likely be feasible.

The study flow has in general shown that the apparently attractive propellant combination LH2/LOX is inferior to CH4/LOX.

A design comprising cryogenic propulsion with CH4/LOX and advanced CFRP tanks shows with respect to LH2/LOX:

- Relaxed constraints on structures, configurations, thermal and power systems
- The same mass reduction potential at same performance (same launch target excess for this specific mission)

C. Study case2: DSR

The options studied for this case are all carried out for a LH2/LOX design. However, the results obtained on JME for LCH4 are applicable to this mission as well. Main data such as the structural mass have been derived from the study case 1.

Figure 10 shows the mass budget of the stack.

Overall stack mass budget for CFRP tanks is reported at Figure 10.

Comparing the two study cases, it was found there is no difference from the thermal point of view in flying to Mars or to Jupiter if boil-off is assumed during the Venus fly-by.

This has two main reasons:

- the configuration of tanks and heat shield is the same due to the constraints of having the LH2 tank facing deep space (in this specific missions studied)
the equipment is due to that reason the same since the number and characteristics of coolers are the same (boil-off is considered during Venus flyby in the Design Case 1).

This two reasons lead to a different power demand of the Thermal domain with the same mass since the equipment remains the same.

The Power contribution is significantly lower due to the closer distance to the Sun (1-1.5 AU mean) with respect to the Design Case 1.

The Propulsion domain shows also the same mass contribution as in Design Case 1 since the equipment (i.e. the thrusters) is the same assuming same thrust levels.

In general it can be said that the results and conclusions are similar to those drawn for Design Case 1.

V. Mission analysis

A. Trades

Mission Analysis determined that missions to Mars and Jupiter with faster transfers than allowed with storable propellants might be feasible, but only at unacceptable cost in terms of payload capability. The loss of payload capability in comparison to the shortened transfer time was deemed too high to make such fast transfers an interesting option for further study.

The potential use of LTCP to enable faster transfers for space telescopes to orbit locations beyond Earth orbit (such as Langrange points) did not turn out to be an interesting application. Even though potentially faster transfer would allow longer science operation times for space telescopes using liquid helium coolant, the average duration of such phase is too short to justify the usage of LTCP.

B. Missions outline

The following information is relevant to the Design Case 1: JME starting from LEO since it is the most critical from mission analysis point of view. Differences between LEO and GTO options are only in the escape sequence, the rest of the mission is identical.

The JME mission outline is shortly reported as follows:

- LEO Launch with Soyuz (no Fregat) from Kourou:
  - initial mass 8400 kg for inclinations < 29 deg.
- Transfer from Earth to Jupiter:
  - VEEGA Strategy
  - Launch in December 2016
  - Arrival in August 2022
- Earth escape using multi-burn strategy

Figure 12 Venus fly-by altitude

<table>
<thead>
<tr>
<th>Element</th>
<th>Option 1 (LEO)</th>
<th>Option 2 (GTO)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Launch mass (includiing adapter)</td>
<td>734.92</td>
<td>734.92</td>
</tr>
<tr>
<td>Element 2</td>
<td>256.74</td>
<td>256.74</td>
</tr>
<tr>
<td>Launch mass (includiing adapter)</td>
<td>255.49</td>
<td>255.49</td>
</tr>
<tr>
<td>Element 3</td>
<td>LTCP (specific)</td>
<td>LTCP (specific)</td>
</tr>
<tr>
<td>Launch mass (includiing adapter)</td>
<td>7458.62</td>
<td>7458.62</td>
</tr>
</tbody>
</table>

Figure 11 Summary mass budget LTCP DSR with CFRP tanks (launch mass of element 3 includes adapter remaining on launcher)
A. Trades
The LTCP modules structural design was driven by the results of the trade-offs performed that were mainly
driven by thermal configuration and heat flux optimization needs.
Pump fed versus pressure fed propulsion trades were soon resolved by an analysis of the tanks overall design
parameters; the 12 bar requirement for the pressure fed system was too stringent and lead to significantly higher tank
masses than the 3 bar requirement for the pump fed system.
The tanks configuration trades were driven by the conflicting thermal requirements to place the LOX tank
between the LH2 tank and the P/L and the structural requirement to achieve the required natural frequency. It was
decided to place the LOX tank in the middle for thermal reasons. The structural mass and natural frequencies
became show stoppers for the LOX-LH2 configurations though. Also, two different tank concepts have been
analysed; reverse bulkhead and separate tanks.

B. Structural analysis
The propellant for the FEM was modeled as rigid body elements with
lumped mass and a best estimate of the inertias. I/F rings were not modeled.
All secondary masses were modeled as non-structural mass and the P/L was
modeled by an effective mass model with appropriate properties.
In particular the Quasi-Static Loads, natural frequencies and internal tank
pressures were taken into account for structural sizing.
It was demonstrated that the natural frequencies were the main driver for
the structures due to the large volumes and mass of the propellant and due to
the configuration. Placing the heavy LOX propellant between the LH2 and
the PL makes the stack very flexible as well as leading to an increased CG
height.

C. Venus fly-by
The Venus fly-by analysis was the defining case for thermal cooling system, in Figure 12and Figure 13 the
flight geometry is plotted for 4 days around Venus swing-by with a Sun distance of 108 million km. The minimum
Venus altitude calculated is 10,000 km. In the figures, distance and Sun-spacecraft-Venus-angle are shown. Based
on this input data, the thermal calculations for this mission phase were carried out, results and details are reported in
the Thermal section.

D. Structures

Figure 13 Venus fly-by angles

Figure 14 bulkhead tank first
lateral mode at 11.2 Hz
The FEM established for the JME and DSR LEO cases, allowed the structural analysis of the tank assembly and a preliminary insight of the characteristics of the tank configurations being traded; the separate tanks solution initially revealed more efficient from mass point of view since the bulkhead configuration was penalised by the special inter-tank structure, main requirement on this part was to provide vacuum separation between the LOX and LH2 tank, keeping a given minimum distance between the two tank walls that made it very heavy.

Following the initial sizing, the structure was modelled in order to determine its natural frequencies. The axial and lateral frequency requirements of the S/C and adapter are 27 Hz and 12 Hz respectively. While the bulkhead configuration could meet the longitudinal (25.7 Hz), the separated tanks solution showed a very low frequency (16.3 Hz). In Figure 14 and Figure 15 the lateral first frequencies of the two tank solutions for the LEO version are shown, none of them meets the requirement. The final structures mass budget, with increased thicknesses to meet frequency requirements, is reported in the systems section. Overall, the expected significant mass saving by removing the reversed dome was not compensating the increased thickness required due to the natural frequencies. Therefore the bulkhead tank solution was preferred due to a better thermal environment and the higher stack frequencies.

The mass was a critical issue at system level and it appeared that LOX-Methane might be favourable due to a better thermal environment and the reduced propellant volumes. The temperatures and pressures of the LOX and Methane tanks meant that a common dome could be applied, saving considerable mass of one dome and the ITS. The LOX tank could also be placed at the bottom of the stack, significantly improving the natural frequencies, as reported at Figure 16.

E. Thermal

A. General

Purpose of thermal studies was to evaluate the feasibility of the ZBO approach and provide preliminary sizing of the thermal control system.

Although both study cases were analysed in terms of cooling power requirements and thermal design, Study Case 1: JME was found the most critical since it envelops from thermal aspects the Study Case 2: DSR, therefore highlight on Study Case 1: JME will be reported in this section.

The mission was subdivided in the following phases to take into account the specific environmental conditions applicable to each of them:

- Launch preparation
- Earth-to-orbit
- Earth orbit
- Venus cruise
- Venus fly-by
- Jupiter

The established mission and system requirements allowed to consider for the majority of the environmental conditions, stack z-axis oriented towards the Sun, as shown in Figure 17.
B. Thermal trades

The two tank alternative configurations analyzed (see Figure 5 and Figure 6 for reference) gave different results in terms of heat load on the propellant tanks, making the bulkhead preferred to the separate tanks configuration. In the bulkhead configuration the LH2 tank shows a higher view factor with the LOx tank and a lower exposure to the warmer radiative heat source constituted by the inter-tank CFRP skirt, with a consequently lower heat exchange. Furthermore, the bulkhead configuration does not require the addition of a heat shield to avoid sun impingement on the inter-tank CFRP skirt, making the structural design lighter.

The separate tanks configuration was found critical in terms of power demand to allow ZBO during the Jupiter phase, the calculated power requirements forced to an increase of solar array area (and mass) with respect to the baseline case (see Figure 1 for reference), this enforced the preferred choice of the bulkhead tank, S/A and power calculations are summarized in Figure 18.

![Figure 17 stack orientation to the Sun](image)

<table>
<thead>
<tr>
<th>JME option 1</th>
<th>JME option 1</th>
</tr>
</thead>
<tbody>
<tr>
<td>Bulkhead tanks configuration</td>
<td>Separate tanks configuration</td>
</tr>
<tr>
<td>Power for ZBO at Jupiter</td>
<td>Power for ZBO at Jupiter</td>
</tr>
<tr>
<td>121 W</td>
<td>305 W</td>
</tr>
<tr>
<td>Increase of SA area / mass respect to reference study</td>
<td>Increase of SA area / mass respect to reference study</td>
</tr>
<tr>
<td>3.06m²</td>
<td>16.11 m²</td>
</tr>
<tr>
<td>18.4 kg</td>
<td>96.6 kg</td>
</tr>
</tbody>
</table>

![Figure 18 impact of separate tanks configuration on S/A](image)

The trade-off on full or partial ZBO during the mission was performed by modeling the two extreme cases for the above described mission phases: ZBO and full boil off, at the given environment and stack orientations. Calculations showed that only during Venus fly-by more than one cooler was needed to satisfy ZBO conditions. Since the extra propellant mass needed to compensate boil-off losses during Venus fly-by was determined lower than 20 Kg LOX and 5 Kg LH2, the selected solution was to allow boil off during Venus fly-by while during all the other phases ZBO was adopted. This avoided unjustified extra hardware mass and complexity to the thermal system. Figure 19 shows the calculated heat entries during Venus fly-by.

The Earth-to-orbit mission phase allowed boil off as well with a total propellant loss of about 15 kg. Details on design aspects of this approach were not studied.

The Methane option was found less stringent in terms of thermal requirement than the LH2 case.

The thermal characteristics of LCH4 provided the possibility to design the propellant storage such that LCH4 and LOX are kept at the same temperature. The complex bulkhead configuration needed for the LH2 case was then deleted and a common LCH4/LOX tank bulkhead was designed, notably releasing structural complexity and mass. Moreover, having the two tanks a common structural part, the same storage temperature and heat loads less than 10 W implied the use of only one common cooler for the two propellants.
A full ZBO approach was then found possible since no critical heat entries were identified during the mission. For the same reason, no increase in solar array area need at Jupiter was necessary in comparison to the LH2 case.

As a final remark, it has to be mentioned the thermal modeling results in general revealed a very high sensitivity of the critical heat entries on the LH2 tank to the stack configuration and orientation during the mission. These two parameters have been taken as priority drivers for the system design, this resulted in a fairly simple heat management of the spacecraft but implied difficulties on other subsystems (i.e. structural masses). Deeper investigations would be advisable to increase the knowledge on the level of optimization obtainable through more refined thermal/structure/power compromise solutions.

For reference, the total calculated LH2 consumption (during cruise Earth-Jupiter, with Venus fly-by - 6 years) would be about 140 kg in the best thermal case (bulkhead tank configuration) and for the most demanding design case (JME).

C. Design

In this study the main requirement for the thermal subsystem was to avoid or limit the boil-off of the cryogenic propellant. To do this a combination of standard passive thermal control hardware (MLI, paints and foam) and cryo-coolers has been used. The firsts to limit the leaking heat going into the tanks, the seconds to extract this heat and avoid or limit the boil-off.

The critical system mass constraints were driver to choose a non redundant design of cryo-coolers and allow boil-off during certain phases of the mission, see Figure 20.

The tanks are in Aluminium with a thickness of 3 mm. The skirts / rings and Sun shields are in CFRP with a thickness of 1.3 mm according to structure subsystem design.

The tanks are covered with a 50 layers MLI except in the inner dome where the LOx tank is Aluminium finished and the LH2 tank covered with 60 mm thick foam.

In the bulkhead configurations all the skirts and rings are covered with MLI externally and silver finished internally while in the separate tank configuration the upper skirt on the external side (the one between S/Cs and LOx tank) is black painted as the Sun shield.

Two cryo-coolers have been used: one for LH2 ZBO that cools the tank till 20 K and one for LOx ZBO that cools the tank till 90 K, only one till 90K for the LCH4 option. From the performance curve in Figure 21 the following electrical powers have been deducted: 375 W are required to extract 1 W at 20 K stage and 150 W to extract 10 W at 90 K stage.

D. Cryo coolers technology

During this study European coolers in predevelopment phase were considered. Different technologies having similar performances are possible:

- Stirling coolers
- Pulse tube coolers
- Turbo Brayton coolers

All of them can be single or multi cooling stage. In some conditions, precocious intermediate stages by passive radiation to space improves their performances.
The last two have advantages with respect to the stirling coolers because they have no moving parts and as a consequence they are not subject to friction, wear and are vibration free.

E. Propulsion

A. Engine

The baseline engine is derived from a LOX/LH2 300N design developed for launcher ACS with the following features:

- Pressure fed (5 bar chamber pressure)
- Regeneratively cooled
- AR 57
- Isp 415 sec
- 4 re-ignitions
- Pre-cooling only for the feed lines
- Stainless steel construction
- Composite nozzle extension
- Development status - Flight prototype

In order to have a motor suitable for our application the following delta developments are required:

- Thrust increase 300 to 400N
- Increased area ratio and hence ISP
- Increased number of re-ignitions
- Switch to pump fed (delivery of the required inlet pressure)

Thrust increase is achieved by an increase in chamber pressure to 6.7 bar, the original 300N unit was flexible between:

- Thrust range: 240 - 480 N
- Chamber pressure: 4 - 8 bar
- Mixture ratio: 3.5 - 6.5
- Propellant mass flow rate: 0.06 - 0.12 kg/s

No problems are foreseen with this change as it lies within the original design parameters of the unit.

Area ratio increase can be achieved by implementing a new carbon composite nozzle extension, with a diameter increase from 165 mm to 370 mm the ISP is raised to 437 seconds, this figure has been adopted for the analyses carried out during the study.

The required number of re-ignitions is a key parameter for the LTCP mission, the majority of them will be needed during orbit raising and Earth escape for the LEO version of both study cases. In the reference engine a chemical system was capable of 4 re-ignitions are possible with 1.5 cc of Triethylaluminium. Suggestion is to increase to 7.5 cc to allow for planned ignition cases and degraded mission in the event of a unit failure, at the unit level this is a minor change.

B. Pump

Assuming inlet pressure at double the chamber conditions, delta difference between tank supply pressure and inlet gives required head rise for pump. Based on the nominal 6.7 bar chamber pressure the required pump powers are: LOX - 95 W, LH2 - 255 W.

On this unit, development focus will centre on including the pump in unit with the tank to comply with pre-cooling requirements, this aspect will require a feasibility study and has un-quantified thermal implications.

Use of pumps represents an operational change to the engine and the start transient will be an additional key issue.
F. Conclusion

The results of this fast track system study performed at ESA investigated the feasibility of the general LTCP concept for the considered interplanetary missions. The in-depth analyses performed on key system areas revealed critical technical aspects that limit the advantages potentially brought by some of the studied technologies.

In particular, thermal and structural constraints connected to propellant high volumes confirmed the LOX/LH2 solution not yet mature and suitable to further development especially in the structural area.

Based on the hydrogen cases results, the hydrocarbon propellant option analysed revealed a very attractive overall performance that made nearly feasible the Jupiter Minisat Explorer mission when launched with Soyuz, this result, although positive, has to be seen still pessimistic; a more dedicated payload design, better adapted to the LTCP concept could reasonably bring further improved outcome.

Due to the demonstrated overall mission and system requirements commonalities, also the Deimos Sample Return mission study took advantage of the hydrocarbon solution, making it an attractive case study.

The required technologies, their current status and future developments were preliminary assessed during the activities even though details on these aspects have not been reported in this paper.

It is worth noting that the majority of the enabling technologies identified for structural, thermal control and propulsion systems are already under development in Europe in the frame of various space programs and technology development initiative, either under the leadership of the Agency (ESA) or with National Agencies follow up.

Appendix

Summary of study case options

<table>
<thead>
<tr>
<th>Sample Mission Case</th>
<th>Description</th>
<th>Option</th>
<th>Description</th>
<th>Version</th>
<th>Description</th>
<th>Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>Case 1: JME LTCP</td>
<td>Jovian Minisat Explorer with LTCP module and Soyuz launch (&quot;Jupiter mission&quot;)</td>
<td>1</td>
<td>Launch with Soyuz to LEO without Fregat, 3x400 N Earth escape, 400 N for all other manoeuvres</td>
<td>a</td>
<td>&quot;Inversed bulkhead tanks&quot; configuration</td>
<td>Eigenfrequency problem and high Structure mass.</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td>b</td>
<td>&quot;Separate tanks&quot; configuration</td>
<td>Increased Eigenfrequency problem w.r.t. Version a, maybe somewhat lower Stucture mass, very high Thermal and Power mass, unfeasibly large solar arrays (also limiting thrust to too low levels).</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td>c</td>
<td>LOX/Methane combination instead of LOX/LH2</td>
<td>Less configuration constraints, much smaller fuel tank, less eigenfrequency problems, etc.</td>
</tr>
<tr>
<td></td>
<td></td>
<td>2</td>
<td>Launch with Soyuz to GTO with Fregat, 400 N</td>
<td>a</td>
<td>&quot;Inversed bulkhead tanks&quot; configuration</td>
<td>High Structure mass.</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td>b</td>
<td>&quot;Separate tanks&quot; configuration</td>
<td>Lower Structure mass than for Version a, high Thermal and Power mass.</td>
</tr>
<tr>
<td>Case 2: DSR LTCP</td>
<td>Deimos Sample Return with LTCP module and Soyuz launch (&quot;Mars mission&quot;)</td>
<td>1</td>
<td>Launch with Soyuz to LEO without Fregat, 3x400 N for all manoeuvres (except final rendez-vous)</td>
<td>a</td>
<td>&quot;Inversed bulkhead tanks&quot; configuration</td>
<td>Eigenfrequency problem and high Structure mass, similar to JME LTCP Option 1a.</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td>b</td>
<td>&quot;Separate tanks&quot; configuration</td>
<td>Increased Eigenfrequency problem w.r.t. Version a, maybe somewhat lower Stucture mass, very high Thermal and Power mass (but lower than on JME LTCP Option 1b).</td>
</tr>
<tr>
<td></td>
<td></td>
<td>2</td>
<td>Launch with Soyuz to GTO with Fregat, 400 N</td>
<td>a</td>
<td>&quot;Inversed bulkhead tanks&quot; configuration</td>
<td>High Structure mass, similar to JME LTCP Option 2a.</td>
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<tr>
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<td></td>
<td></td>
<td></td>
<td>b</td>
<td>&quot;Separate tanks&quot; configuration</td>
<td>Lower Structure mass than for Version a, high Thermal and Power mass (but lower than on JME LTCP Option 2b).</td>
</tr>
</tbody>
</table>
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References
