Promising Space Transportation Applications Using Hybrid Propulsion

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Abstract

ORPHEE (Original Research Project on Hybrid Engine in Europe) is a co-funded project between the EU, industries and universities; one of its purposes is to identify potential space applications for hybrid propulsion. Engines based on this innovative propulsion concept offer promising advantages like thrust performance, thrust modulation, re-ignition, versatility, simplicity and safety.
In such context, ASTRIUM SAS aims to define the most interesting applications and missions making use of the favourable capabilities and performances of this future propulsion system.
Mission analyses made by ASTRIUM SAS, with objective assumptions for hybrid engines characteristics that will be consolidated subsequently within the ORPHEE project, allow to select four platforms for which the hybrid technology could favourably replace the existing propulsive systems. These are:

- an hybrid upper stage on top of a small launcher,
- an hybrid 1st stage of a small low-cost launcher,
- a Lunar lander,
- a Martian lander.

Preliminary designs of the hybrid propulsive systems and related stages are also included in this paper.

Acronyms

HRB : *Hybrid Rocket Booster*
HTPB : *Hydoxyl Terminated PolyButadiene*
HUS : *Hybrid Upper Stage*
Isp(v) : *Specific Impulse (in vacuum)*
LOx : *Liquid Oxygen*
LRB : *Liquid Rocket Booster*
Mp : *Propellants mass*
O/F : *Mixture ratio*
Pdyn : *Dynamic pressure*
Qtot : *Total mass flow rate*
RP1 : *RP1 grade kerosene*
SSO : *Sun Synchronous Orbit*
Tb : *Burning time*
TVC : *Thrust Vector Control (actuators, power device…)*

1. Introduction

Hybrid propulsion is based on the injection of an oxidizer (liquid or gaseous) into a combustion chamber where an exothermal reaction is realized with the solid fuel, the emitted hot gases are then exhausted through a nozzle providing thrust.
Theoretically, hybrid propulsion offers several advantages compared to classical solid or liquid propulsion like a better Isp (with regard to common solid propellants and close to LOx/RP1 Isp), simplicity (only one liquid to manage), safer (no hazardous component), flexibility with the thrust modulation, extinction/reignition.
Nevertheless, hybrid propulsion presents drawbacks, which could explain its lack of use in space applications. Among them, the low burning rate of the fuel (HTPB for instance) implies the manufacturing of complex grain shapes (typically the wagon wheel), an experimental specific impulse lower than expected, the combustion instabilities that could have significant consequences at the System level.
Considering these latter points, a consortium formed by European industries and universities and co-funded by the European Community within the seventh framework programme, under the space theme, is working to find new fuels (named advanced fuels) allowing to remove, or at least to reduce the drawbacks inherent of hybrid propulsion with high performances, high regression rates, high throttle capabilities while considering the environmental constraints. It is the **ORPHEE** (***O****perational* ***R****esearch* ***P****roject on* ***H****ybrid* ***E****ngine in* ***E****urope*) Project.

2. ORPHEE project

The main objectives of **ORPHEE** are to increase versatility of space propulsion systems, to ensure a significant increase of hybrid engine performance, to improve the solid fuel technological maturity from TRL 1 to 3, to gather European skills on hybrid propulsion and to economize on the European access to space.

In near future, the availability of new hybrid engines will allow the access to new space transportation missions. By consolidating the knowledge on this innovative technology and by implementing solutions in upcoming space agencies roadmaps, the European space propulsion community will strengthen its global position.
The project is split in five main work packages:
1. WP 100 - Project Coordination & Management, dissemination and exploitation plan,
2. WP 200 - System level definition including missions analysis and requirements,
3. WP 300 - Trade-off and solid fuel optimization: make a significant progress in the fuel composition,
4. WP 400 - Hybrid engine modelling: provide numerical models to simulate the operating of a hybrid engine,
5. WP 500 - Roadmap and hybrid engine demonstrator designs including the definition of demonstrators requirements.

The appendix 1 presents the participants involved in the consortium.

3. Promising applications

Brief presentation of the hybrid propulsion
As described in past publications [1, 2, 3, 4] (and many others), hybrid propulsion offers a lot of advantages, at least at the theory level. Among them, it can be noticed:

- simplicity of use: one liquid to manage inducing the capability to stop and restart the combustion by cutting off the oxidizer feeding,
- performance: Theoretically, the specific impulse of a hybrid propellants couple using liquid oxygen and HTPB is greater than a classical solid propellant in the same nozzle expansion ratio, but widely lower than pure cryogenic couple (LOx/LH2). With a gain of about 30 s on the Isp, the hybrid couple is as performant as current bi-liquid propellants (like LOx/RP-1). With expected advanced fuels (more energetic but to be determined), the specific impulse can rival with semi-cryogenic liquid propellants (LOx/CH4) as presented in the next figure (calculated with a chamber pressure equal to 7 MPa).

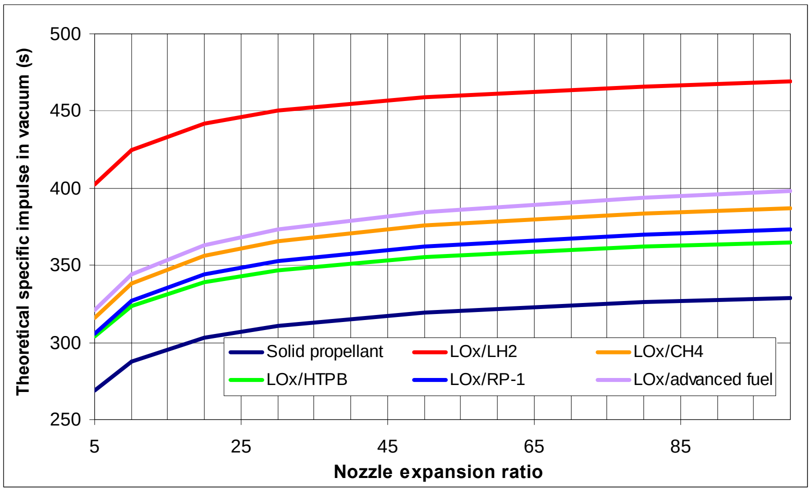

Figure 1

  Moreover, high thrusts can be obtained with advanced fuels (increased regression rates et better Isp) compared with standard solid propulsion, which could be a good alternative for future propulsion systems.
- flexibility: thanks to only one liquid, the thrust modulation is obtained modifying the oxidizer mass flowrate and/or the mixture ratio which alters the specific impulse. This thrust modulation allows to manage the dynamic pressure on launchers flights or to compensate the space platform weight for a soft landing
- safety: the oxidizer and the fuel are stored separately which reduces the self ignition of the engine. Moreover, in the case of fully inert material (it is one of the most important objectives in hybrid propulsion), i.e. without pyrotechnical additives, the engine manufacturing, transportation, assembly and the operations on launch pad are safer, the catastrophic failure is reduced compared to classical chemical propulsion. By its way of combustion, the limited cracks within the grain are without effect on the combustion, because the pressure is not the main combustion driver as in a solid motor, which gives to the hybrid engine a tolerant character during manufacturing and operating,
- reduced costs: the use of theoretical inert and safe materials (no specific human and/or ground installation protection is required), the simplicity of such a propulsion system, the development, and recurring costs and the operations on the launch pad should be minimized with regard to bi-liquid system for instance,

- environment impacts: during manufacturing (to be confirmed especially at the advanced fuel level), on launch pad and in flight, they should be very reduced. For instance, with the LOx/HTPB couple, no hydrochloric acid is exhausted, which is not the case on solid propulsion.

Among the hybrid propulsion drawbacks, the main one is the low regression rate of classical fuels like HTPB which imposes the manufacturing of complex grain shapes like wagon wheel grain to compensate that low burning rate. A second drawback is the combustion instabilities partially linked to the regression rate and the interaction between the injection and the grain combustion. These enhancements are the major objectives of **ORPHEE** project.

Platforms selection

The potential applications of hybrid propulsion depend on the size, the expected performances (high Isp), the requirements to be met (reignition, thrust modulation…)

The following diagram presents the improvement of maturity of hybrid propulsion with regard to several aimed targets and the date of their availability, as evaluated in the next diagram.

This gross roadmap shows that the very first operational application could be linked to kick stage or space tourism due to needs in term of performance and implied propellant masses that are not excessive, compared to mandatory safety and reliability.

Later, it can be envisaged more complex systems for exploration that require, in addition of safety and reliability (in particular for manned space missions) higher propellant masses, more performant propellants and the ability of multi ignitions, throttlability, which demand a longer development schedule.

Around 10 years after the beginning (if the funding follows), with very high regression rate and specific impulse, low cost first stage or heavy boosters with high thrust and performant upper stages can be proposed to answer either the satellite market where satellites are bigger and bigger or low cost launch services.

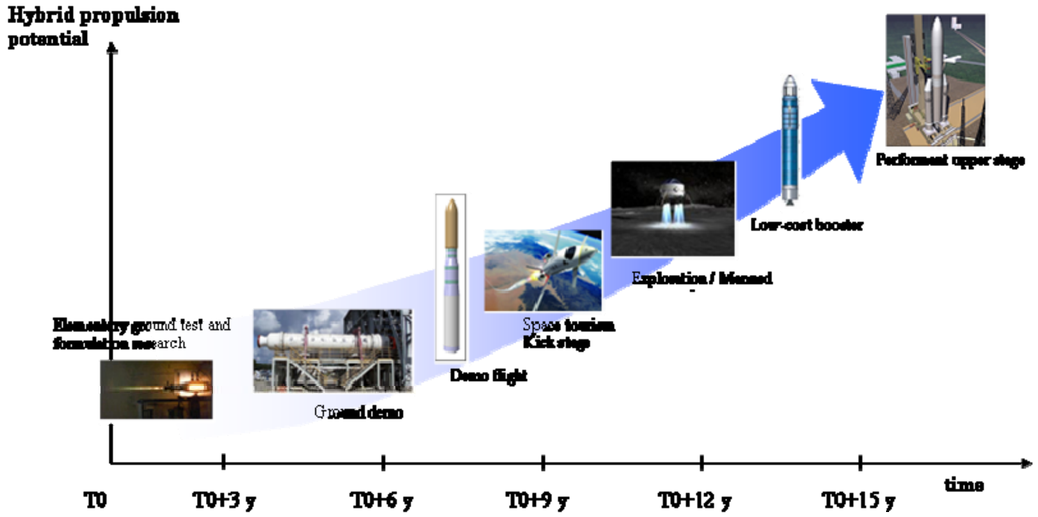

Figure 2

Thanks to a common method used within ASTRIUM SAS (ADO "Aide à la Décision Opérationnelle" - operational decision helper method), the most promising platforms for hybrid propulsion will be selected.

After this work, a quotation of each application was performed in order to rank them. It is shown on the next table.

Table I

Space application	Rate (%)	Rank
manned spacecraft	73,8	1
exploration spacecraft	70,1	2
performant upper stage	66,3	3
low cost engine	66,2	4
space tourism	63,5	5
maneuvering and transfer vehicle	63,1	6
semi-reusable first stage	63,1	6
airborne launcher	63	8
heavy strap-on booster	55,4	9
satellite kick stage (if only distancing is required)	53,8	10
launcher kick stage (if only distancing is required)	53,8	10

According to ASTRIUM SAS analysis, the surrounded applications represent the most promising applications for hybrid propulsion even if the other applications are also open to this kind of propulsion.

- the exploration (automatic or manned spacecraft) thanks to the safety inherent of hybrid propulsion, the throttlability, its reignition capacity and its compactness,
- the upper stages for its potential high thrust and its ability to be extinguished and reignited,
- the low cost launchers for mainly for the cost objective,

In comparison with the space market described in the previous paragraph, it can be noticed that there is a quite good consistency between the future needs (space exploration, low cost launchers to reach low orbits) and the capacities of hybrid propulsion in term of performance, robustness, safety and costs.

For the mission analysis, these selected platforms are presented hereafter (the manned spacecraft is not considered in this study because it does not seem to be an application promoted by ESA in the next years).

4. Mission analysis of the selected platforms

4.1 Hybrid upper stage on top of a small launcher

To perform our analysis to demonstrate the gain reachable with hybrid propulsion, it has been to select a VEGA type launcher (SSO mission) and to replace the upper stages (Z9 and AVUM) by a hybrid equivalent supposing the same mechanical interfaces with regard to the payload adaptator/fairing and the second stage (same stage diameter)
For information, in order to have a reference point, the VEGA launcher performance has been calculated from data coming from opened literature [5, 6, 7] (**ORPHEE** project constraints).
Thus, the VEGA launcher mission characteristics are summarized in the following table:

Table II

Launch pad	KOUROU (French Guyana)
Mission	SSO: 800 km - Inclination: 98.6 °
Launch azimuth	0 °
1st and 2nd flight	flights performed at null aerodynamic incidence (except during initial pitch manoeuvre)
Constraints	Flux at fairing jettison < 1135 W/m² Intermediary orbit periapsis > 160 km
Payload mass	1300 kg
Injection apoapsis	800 km
Injection periapsis	800 km
Inclination	98,6°
Max. Pdyn	59 kPa

On the basis of these data, the work was to evaluate the hybrid upper stage with objective performances in order to define preliminary requirements for the fuel formulation and the engine performances.

With SAFRAN-SME support, different abaci were calculated in order to determine the optimum configuration. Of course, the structural index and the specific impulse are the main contributors on the launcher performance.
According to these results:

- the burning time Tb does not have a significant impact. However, the Tb of 180 s results in slightly greater performances and allows reducing the acceleration during the upper stage flight.
- the optimal configuration among the computed cases is :
  - Mp = 11 tons
  - Tb = 180 s
  - Isv = 360 s
  - Structural index = 0.15
  - Payload mass = 2.6 tons

A preliminary hybrid stage design has been done from the mission analysis requirements to consolidate the performance evaluation.
The chosen oxidizer is the liquid oxygen. This choice is justified by the following facts:

- the mission of such a launcher is classical and well spread,
- the LOx is a commonly used in space applications
- the implementation of LOx is well known within the space industries.

For the solid fuel, it is more difficult to fix any characteristic because the advanced fuels do not exist for the time being. Therefore, it is defined arbitrarily some elements with regard to some theoretical results with past studies like paraffin or HTPB including metallic hydrides or other chemical components improving the global solid fuel performances (mechanical strength, improved regression rate, etc.).
For this study, no constraints are considered concerning the fuel formulation, its implementation (in particular it is supposed out of pyrotechnical division, e.g. absolutely inert) and its difficulty to obtain the desired grain (the industrial tools are supposed available and qualified). Only its intrinsic characteristics with respect to the targeted performance are considered and will have to be confirmed within the **ORPHEE** program. An update of this sizing will be performed at the end of **ORPHEE** project with regard to the results progress (formulation, lab scale tests, numerical models…).
The preliminary design of the HUS is synthesized in the *Table III* and *Figure 3,* the trajectory results being illustrated by the *Figure 4.*

Table III

Component	Mass (kg)
Combustion chamber	6176
Solid grain	5250
Inert (insulated case, nozzle, igniter)	917
Injection system	9
Pressurisation sub-system	362
Turbo-pump	72
Gas generator	290
Oxidizer	5250
Tank	198
Helium pressurisation system	12
TVC	45
Miscellaneous	45
Total	12088
Propellants	10500
Inert	1588
Mass index (%)	15,1
Geometry	**mm**
Overall diameter	1905
Combustion chamber length	3966
Liquid tank length	2020
Overall length	6400
Average throat diameter	133
Nozzle areas ratio (average)	80
Performance	
Mean fuel regression rate (mm/s)	4,6
Pmean (MPa)	7
O/F mean	1
Isv mean (s)	350
Qtot mean (kg/s)	58.3
Payload mass updating	2100 kg
Injection apoapsis	800 km
Injection periapsis	800 km
Inclination	98,6°
Max. dynamic pressure	57 kPa
Lift-off mass	138 T
Vacuum thrust	200.2 kN
Overall combustion time	180 s
Flow-rate (average)	58.3 kg/s
1st Boost propellant mass	10294.9 kg
1st Boost duration	176.5 s
Ballistic phase duration	2177 s
2nd Boost propellant mass	205.1 kg
2nd Boost duration	3.5 s

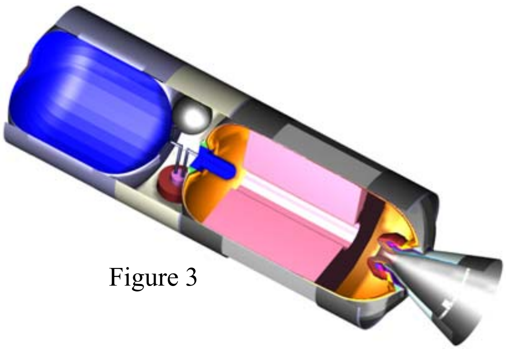

Figure 3

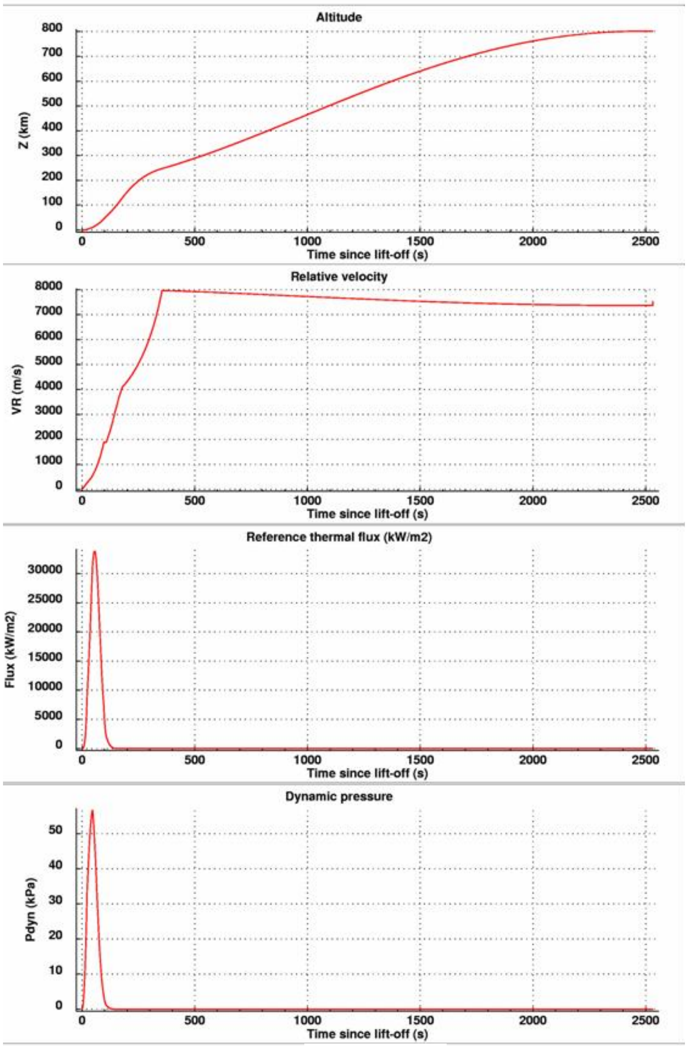

Figure 4

A very preliminary design of a hybrid upper stage dedicated to replace a classical stage (solid or bi-liquid propulsion) for a VEGA type launcher in order to improve the launcher performance has been done in the present document taking into account mission analysis data and assumptions for the pre-design.
This preliminary study shows that such a HUS could be envisaged to improve the performance of a VEGA-type launcher. The assessed payload gain is +800kg (≈ +60%). Nevertheless, some parts of the design have to be improved (dimensioning and ballistics model) and consolidated (mechanical dimensioning, technologies choice…) in a further phase as to precise the real impact on the payload mass.

4.2 Hybrid propulsion module for a moon lander

The aim of this section is to show the interest of hybrid propulsion for soft-landing on the Moon surface, with an emphasis on the need in thrust modulation.
The selected descent scenario to be performed by the lunar lander is taken from an IAC paper [8] and presented in *Figure 5*, where is depicted a lunar robotic mission sent with Ariane 5 ME launcher. This paper refers at a landed mass of 2 tons (featuring the inerts and the payload). The lander mission starts from a circular and polar LLO (Low Lunar Orbit) at an altitude of 100km.

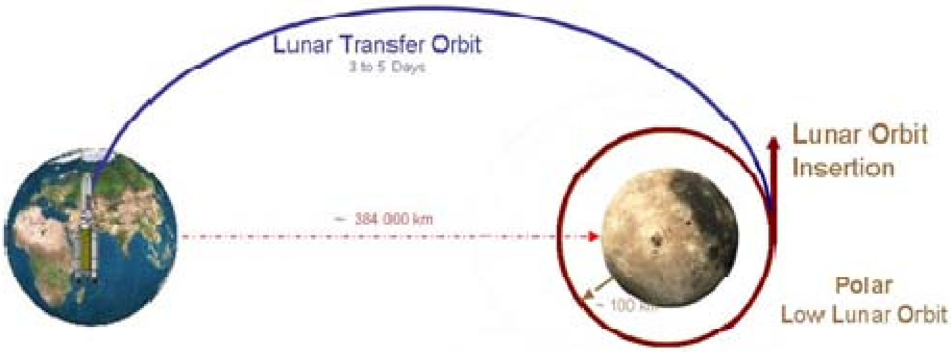

Figure 5

The descent and the softlanding flights sequences are shown on the next figure.

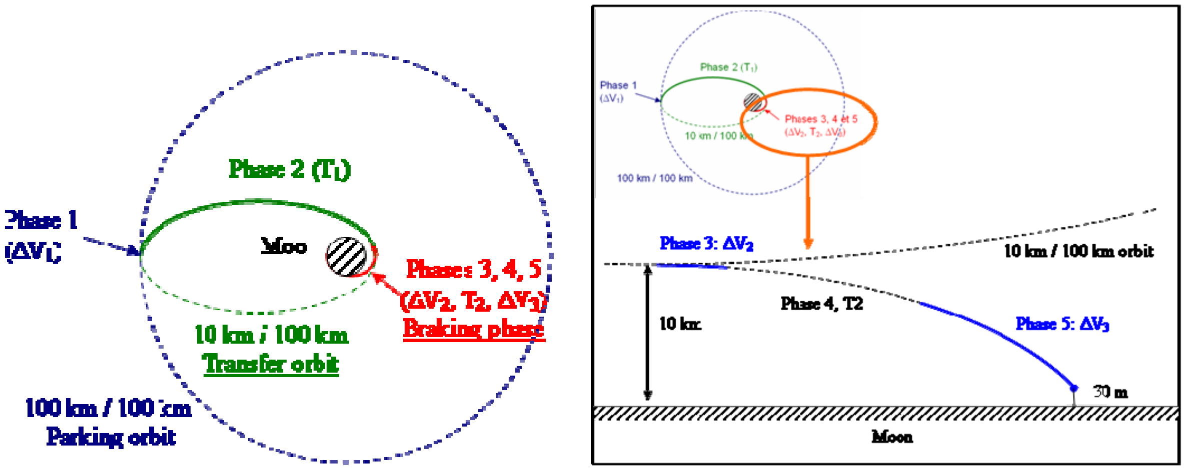

Figure 6

The mission first manoeuvre is the vehicle de-orbitation from its parking orbit (Phase 1). This manoeuvre sends the vehicle on an elliptic transfer orbit, which periselene altitude is 10 km (periselene usual minimum altitude with respect to navigation constraints, Apollo missions are closed to 15km).
A coast phase (Phase 2) follows the parking orbit de-orbit manoeuvre in order to get nearer to the transfer orbit periselene (the location of the braking phases' start is optimised, as it depends on the available maximal thrust level).
The braking and landing are then made in three phases:

- elliptic orbit de-orbit maneuver (Phase 3), at maximum constant thrust
- ballistic phase (Phase 4)
- braking phase (Phase 5) at maximum constant thrust

The braking phases are performed at maximal thrust in order to limit the gravity losses. The braking phase ends at 30 meters from lunar surface, where the final vertical descent begins (Apollo scenario [9]).
In order to perform soft-landing, the module must touch the lunar surface as vertical as possible, with low velocity modulus and nearly cancelled horizontal velocity.
The chosen conditions at the beginning of the vertical descent are:

- An altitude of 30 meters [9]
- A vertical relative velocity of 1 m/s (typical value for soft-landing final velocity in open sources)
- Null horizontal relative velocity

The vertical descent is performed at constant relative vertical velocity until touchdown (1 m/s). Therefore, the thrust must counter lunar attraction. The lander attitude is vertical as the thrust is along its longitudinal axis (see *Figure 7*). As the landed mass is 2 tons, the related minimal thrust to be considered is the one that counters the weight: $Thrust_{minimal} \approx 3.2$ kN (as $g_{Moon} = 1.622$ m/s²). This minimal thrust level is only applied during the vertical descent, which lasts for 30 seconds.
During phase 3, the thrust direction considered is the opposite of relative velocity direction (180° angle of attack). This is called the gravity turn command law, and aims at reducing the relative velocity.
Phase 5 is shared into two phases, each one having a different command law:

- the first part of phase 5 (phase 5.1) is performed with the gravity turn law (see *Figure 7*)
- the second part of phase 5 (phase 5.2) is performed using a linear attitude command law. This allows differentiating the thrust direction from the relative velocity direction and thus, can decrease the relative velocity slope (down to -90°). This is mandatory to be compliant with the constraints at the start of the final vertical descent. The command law is optimized through the initial value of the local pitch angle θ and its constant derivative.

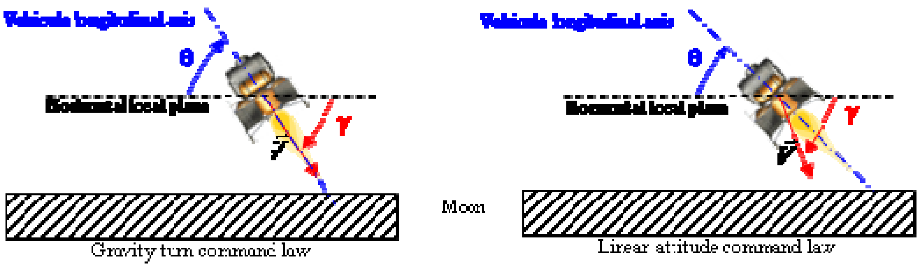

Figure 7

The following trajectory parameters are optimized, in order to minimise the required propellant mass (which is similar than minimizing the ΔV):

- the braking phases' start (phase 2 duration: T1)
- phase 3 duration
- ballistic phase 4 duration T2 (if existing)
- phase 5.1 duration (if not merge with phase 3)
- phase 5.2 duration
- θi and $\dot{\theta}$ during phase 5.2

The descent and landing trajectories and performances are computed with the same in-house flight-proven software as in section 5.1, for several preliminary propulsive characteristics:

- Isv of 350s, 325s and 300s at maximal thrust level, as thrust modulation impacts the Isv
- Modulation ratio between 2.5 and 15

The results are presented hereafter.

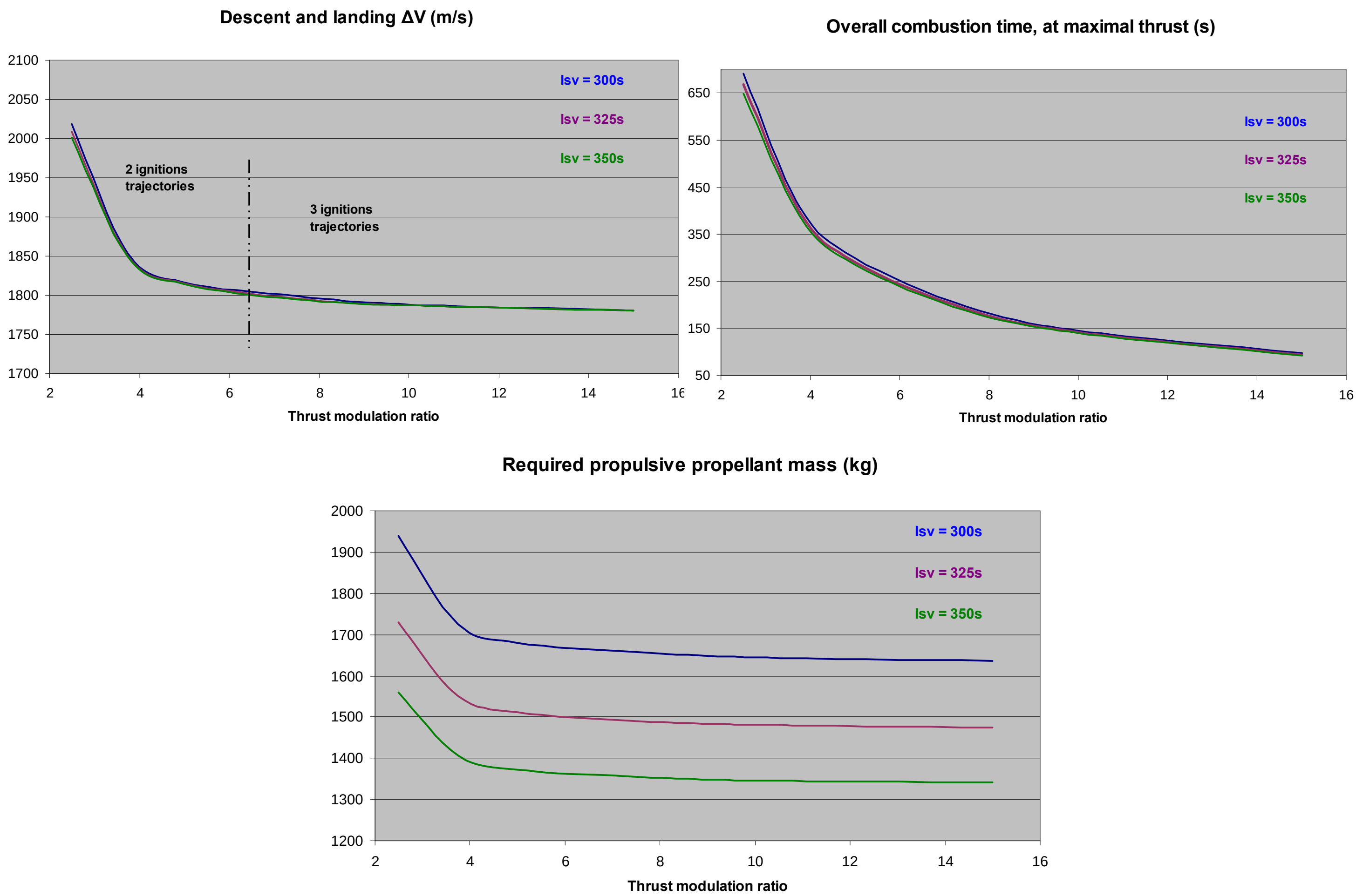

Figure 8

On the basis of the mission requirements, a very preliminary design of the hybrid propulsion system was performed taking into account the following assumptions:

- oxidizer: LOx (the mission duration is supposed compatible with the hold in temperature of the oxidizer, aspect to be analysed in a more detailed study),
- pressure fed system
- the centre motor gives the two first boosts (ΔV1 and ΔV2) for the successive de-orbitings,
- the four engines are operating during the maximum deceleration (ΔV3),
- the centre motor is cut off, and only the three outside engines are operating until the soft landing (ΔV4).

The three outside motors are designed to produce each one between 1/3 of minimal thrust (lander minimum thrust ≈ 3.2 kN) and 2,5 times the minimum thrust ($Thrust_{maximal}/Thrust_{minimal}$ =7.5). The last phase before landing the three outside motors are operating together and produce the minimal thrust in such a way that the thrust/weight ratio remains equal to 1. The central engine is defined to give the missing thrust to obtain a maximal thrust of fifteen times 3.2 kN.

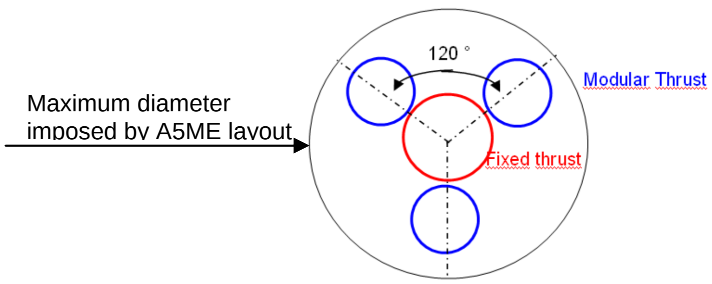

Figure 9

In these conditions, the mass budget and the performance obtained are the following:

Table IV

Component	Mass (kg)
Outside Combustion chamber (unit)	153
Solid grain	117
Inert (insulated case, nozzle, igniter)	34
Injection system	2
Central Combustion chamber	408
Solid grain	338
Inert (insulated case, nozzle, igniter)	68
Injection system	2
Oxidizer	689
Tank	46
Helium pressurisation system	42
Miscellaneous	13
Total	1657
Propellants	1378
Inert	279
Mass index (%)	20.25

Combustion chamber – outer engine	
Overall diameter (mm)	542
Solid length (with nozzle) (mm)	926.5
Average throat diameter (mm)	48
Reg. rate in max thrust (mm/s)	2
Reg. rate in min thrust (mm/s)	0.3
Pmean (MPa)	2

Combustion chamber – central engine	
Overall diameter (mm)	673
Solid length (with nozzle) (mm)	2074.8
Average throat diameter (mm)	72.6
Regression rate (mm/s)	3
Pmean (MPa)	3

Oxidizer tank (common tank)	
Overall diameter (mm)	900
length (mm)	1203.5

The next picture gives a global layout of the propulsion system. The *Tables VI and VII* give the ballistic characteristics and the consistency of the required ΔV.

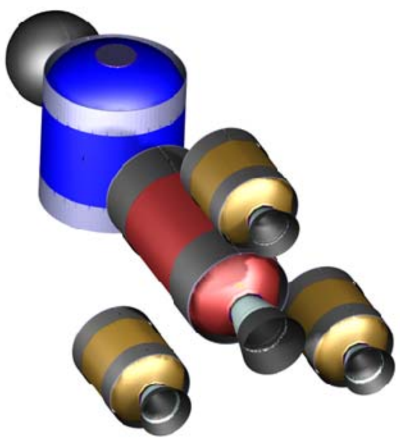

Figure 10

Table V

Pmean at max. thrust	3 MPa
O/F mean at max. thrust	1
Qtot mean during phase at max. thrust (all motors fired up)	14 kg/s
Qtot mean during phase at min. thrust (only the 3 outer motors are operating)	1.1 kg/s
Isv mean during phase at max. thrust	350 s
Isv mean during phase at min. thrust	300 s

Table VI

	Isv mean (s)	time (s)	Final mass (kg)	ΔV (m/s)
Boost 1	350	3.4	3357.6	20.6
1st ballistic phase		3400		
Boost 2	350	3.8	3334.8	23.3
2nd ballistic phase		570		
Boost 3	350	92.5	2039.8	1688.2
Boost 4	300	30.3	2006.5	49

This study shows that hybrid propulsion can be envisaged for a Moon lander mission. Hybrid engines have the advantage to be able to perform soft-landing at low thrust levels, as well as the descent, using their high thrust levels, subject to an experimental verification of the computed ballistic parameters
With the present assumptions, the total mass of the lander at the beginning of the mission in LLO is 3378 kg including 1378 kg of propellants. So on the moon surface, the dry mass is equal to 2 tons with about 280 kg of engines inert. In these conditions, the payload total mass (including the different cases, plateaus, equipment bay, electrical devices, etc, and the real payload) is equal to 1720 kg. A preliminary study of the global lander architecture will allow to define the effective payload mass.
Nevertheless, some assumptions and results of the design have to be improved (dimensioning and ballistics model) and consolidated (mechanical dimensioning, technologies choice…) in further **ORPHEE** steps; as well as their impact on the payload mass.

4.3 Hybrid propulsion module for a Mars lander

The aim of this section is to show the interest of hybrid propulsion for Mars pinpoint soft-landing, with an emphasis on the need in thrust modulation.
The selected descent scenario to be performed by the Martian lander is taken from an AIAA paper [10], where is performed a preliminary reconstruction of Mars Phoenix lander descent scenario, which successfully landed on the northern arctic plains of Mars. The landing occurred 21 km further downrange than the predicted landing location.
The next generation Mars mission will further increase landing accuracy resulting in a delivery of the lander to within 10 km. However, the need to perform pinpoint landing (PPL) will be required. PPL is defined as guiding a lander spacecraft to a given target on the surface with a good accuracy, typically of one hundred meters. The ability to perform pinpoint landing is gaining importance due to renewed interest in both manned and robotic exploration of Mars. For example, it may be required to land next to scientifically interesting targets located within hazardous terrain, or to land in the vicinity of other pre-positioned surface assets such as a rover from a previous mission.

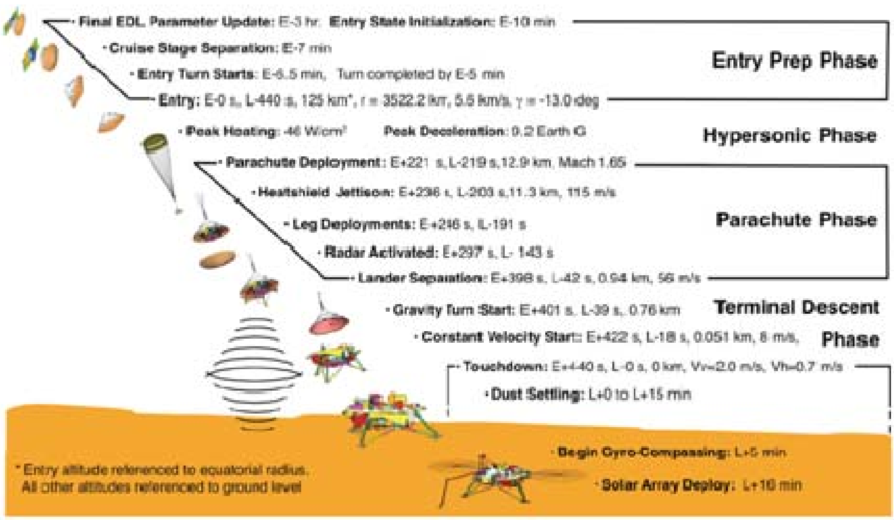

Figure 11

This section focuses on the powered terminal descent phase of the Phoenix scenario as shown hereafter, which is replaced by a pinpoint landing scenario. The final specifications for the hybrid propulsion module are designed to make up for an inaccuracy of 10 km after the parachute descent phase.
The conditions at landing are compliant with a soft-landing:

- Height = 0 m (Mars surface)

- Horizontal relative velocity = 0 m/s
- Vertical relative velocity = 1 m/s
- Relative velocity slope = -90°
- Null horizontal acceleration
- Null vertical acceleration, which means that the thrust counters Mars gravity

The locations of the landing site (expressed in down and cross ranges from terminal descent start) are parameters of this study.
The selected final landed mass mf is close to phoenix lander mass and is 350 kg.

The basic PPL guidance problem aims at finding a trajectory that transfers the spacecraft from any given state at engine ignition to a desired terminal state without violating state constraints (e.g., reaching the target without flying subsurface), in a constant gravitational field and without considering aerodynamics effects.
These hypotheses are justifiable as the powered terminal phase starts at low altitude (for the constant gravitational field) and at low relative velocity in Mars rarefied atmosphere (to neglect aerodynamic forces).
The spacecraft may travel several kilometers from the point of engine ignition to reach the landing site (due to the trajectory dispersions during the parachute phase). The guidance problem is formulated as a minimum required fuel/energy optimization problem in order to limit the consumption.
Several solutions to this guidance problem can be envisaged and are compared in document [11]:

- minimum-fuel optimal control, using convex programming
- Apollo-like quadratic guidance laws
- higher-order polynomial guidance laws, minimizing the required energy (a^2, with a the applied acceleration)

The minimum-fuel optimal control has the best performances. However the implementation of this kind of solution is not ensured in a real-time scenario as it requires complex programming, according to [11].
In this study, the selected guidance program for the pinpoint soft-landing problem is a high-order polynomial solution. It is computationally simple and it is a feasible solution, as it is based on flight-proven Apollo polynomial quadratic guidance. To use higher polynomial order improves computed trajectories in terms of fuel consumption/energy. It is stated that a polynomial order of 3 is sufficient for Mars pinpoint problem [15].

Thanks to these assumptions and considering storable propellants (long duration mission) with a specific impulse of 320 s, the mission analysis results are:

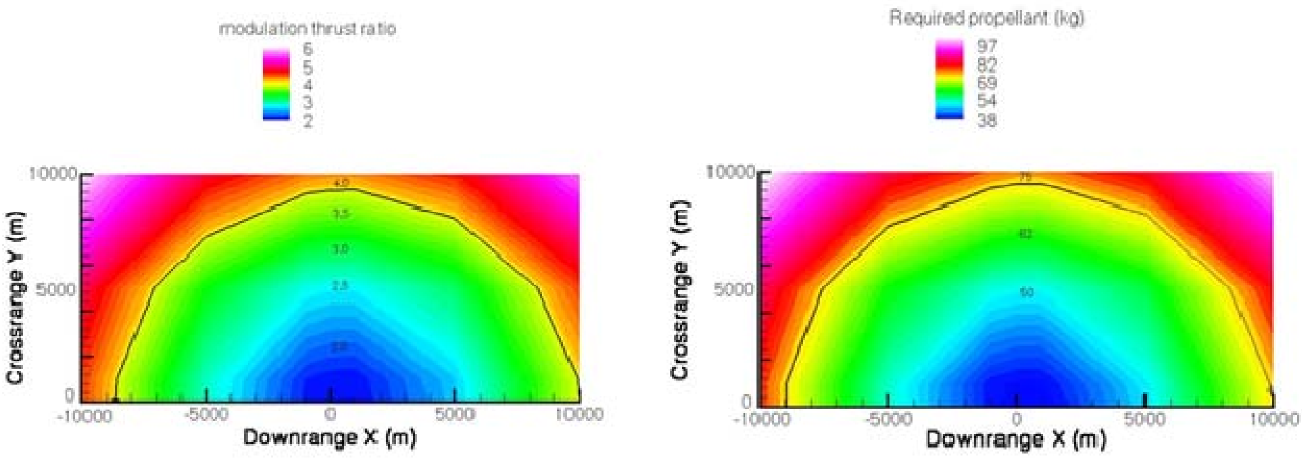

Figure 12

The required thrust modulation ratio increases with the ground surface the spacecraft must cover, e.g. the area over which this kind of guidance finds an eligible trajectory.
The positive downrange requires a little less thrust (and fuel) as the initial velocity direction tends the trajectory towards X >0.
The next generation Mars mission will further increase landing accuracy resulting in a delivery of the lander to within 10 km, after the parachute phase.
In this particular case of Phoenix descent scenario terminated by a pinpoint soft-landing performed using a polynomial guidance law, a thrust modulation ratio around 4 can cope with 10 km dispersions.
The fuel consumption increases with the distance to the initial position.
To travel a distance of 10 km during Phoenix terminal descent phase is feasible with 75 kg of propellant, with the selected guidance algorithm.

Note: the durations of all these computed trajectories that minimise the thrust modulation ratio go up to 70 seconds. This value is retained as the maximal required burning time.

The following selected specifications for the Mars lander propulsion system are related to a covered landing area with a 10 km radius.

Table VII

Minimal thrust (N)	1300
Maximal thrust (N)	5200
Thrust modulation ratio	4
Burning time (s)	70
Isv (s)	320
Propellant mass (kg)	75

With these requirements and considering H_2O_2 (90%) as oxidizer (pressure fed cycle), the preliminary sizing of the engine is given by the next table.

Table VIII

Component	Mass (kg)
Combustion chamber	60,7
Solid grain	41.6
Inert (insulated case, nozzle, igniter, injector)	20.1
Oxidizer	33.4
Tank	8.5
Helium pressurisation system	2.6
Miscellaneous	10
Total	113.6
Propellants	75
Inert	39.6
Mass index (%)	52.8
Geometry	**mm**
Combustion chamber	
Overall diameter	389
Solid length (with nozzle)	800
Nozzle areas ratio (average)	80
Oxidizer tank (cylindro-torus)	
Overall diameter	568
length	300
Helium sphere diameter	325
Performance	
Pmean (MPa)	2.7
Pmax (MPa)	3.8
O/F mean	0.87
Isv mean (s)	320
Maximum Regression Rate (mm/s)	3.85

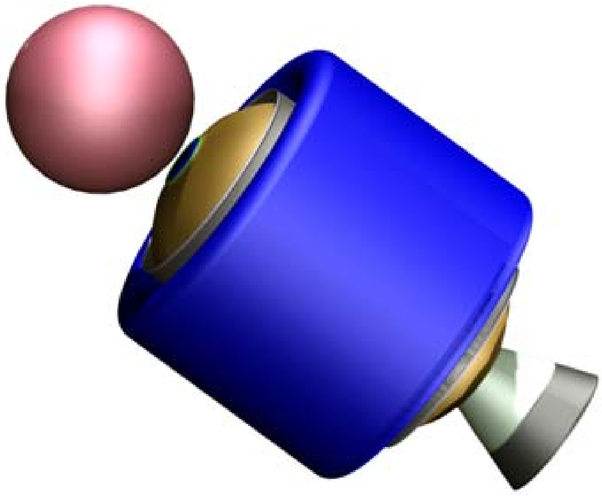

Figure 13

Further exploration missions on Mars surface require an accurate landing. A hybrid engine, through its thrust modularity, can be used to perform pinpoint landing. Flight-proven guidance algorithms, such as polynomial laws used for Apollo missions, result in modulated but continuous acceleration profile. This requirement fits hybrid propulsion.

With the present assumptions (guidance, fuels…) and the targeted coverage of a 10km radius landing area, to land a dry mass of 350kg on Mars surface results in an engine inert mass of 40kg. The required (and embedded) propellant mass goes up to 75kg and the required thrust modulation ratio is 4.

In these conditions, the remaining landing mass (including the different cases, ACS, equipment bay, electrical devices… and the effective payload) is equal to 312 kg. A preliminary study of the global lander architecture would allow to define the effective payload mass.

4.4 Hybrid booster on a low-cost launcher

The last platform to be studied is the use of hybrid propulsion for a first stage (or HRB). A FALCON1-type launcher is chosen as reference because of its cheapness and simple architecture.

The aim of this section is to give a verdict on the interest and potential payload gain due to the replacement of FALCON1-type 1st stage by a hybrid stage, while respecting reference dynamic pressure and acceleration constraints using thrust modulation.
The very first computation was to determine the reference mission of the FALCON1 launcher from Kourou in order to compare the performances with the hybrid booster.
The data relative to FALCON1 launcher come from the user manual [12].
The vacuum specific impulse has been fixed at 310 s for the hybrid engine, and the optimal loading found after convergence of this analytical process is 21653 kg of propellant with its related preliminary structural index of 11.7%.
Considering the optimal loaded mass, several trajectories are computed to determine the best mass flow rate while respecting the Falcon1-type dynamic pressure and acceleration. The trajectories are computed for:

- several flow-rates : 130 kg/s, 155 kg/s, 180kg/s and 205kg/s,
- with and without thrust modulation.

The following diagrams compare the different results obtained by ASTRIUM SAS:

Without thrust modulation | ***With thrust modulation***

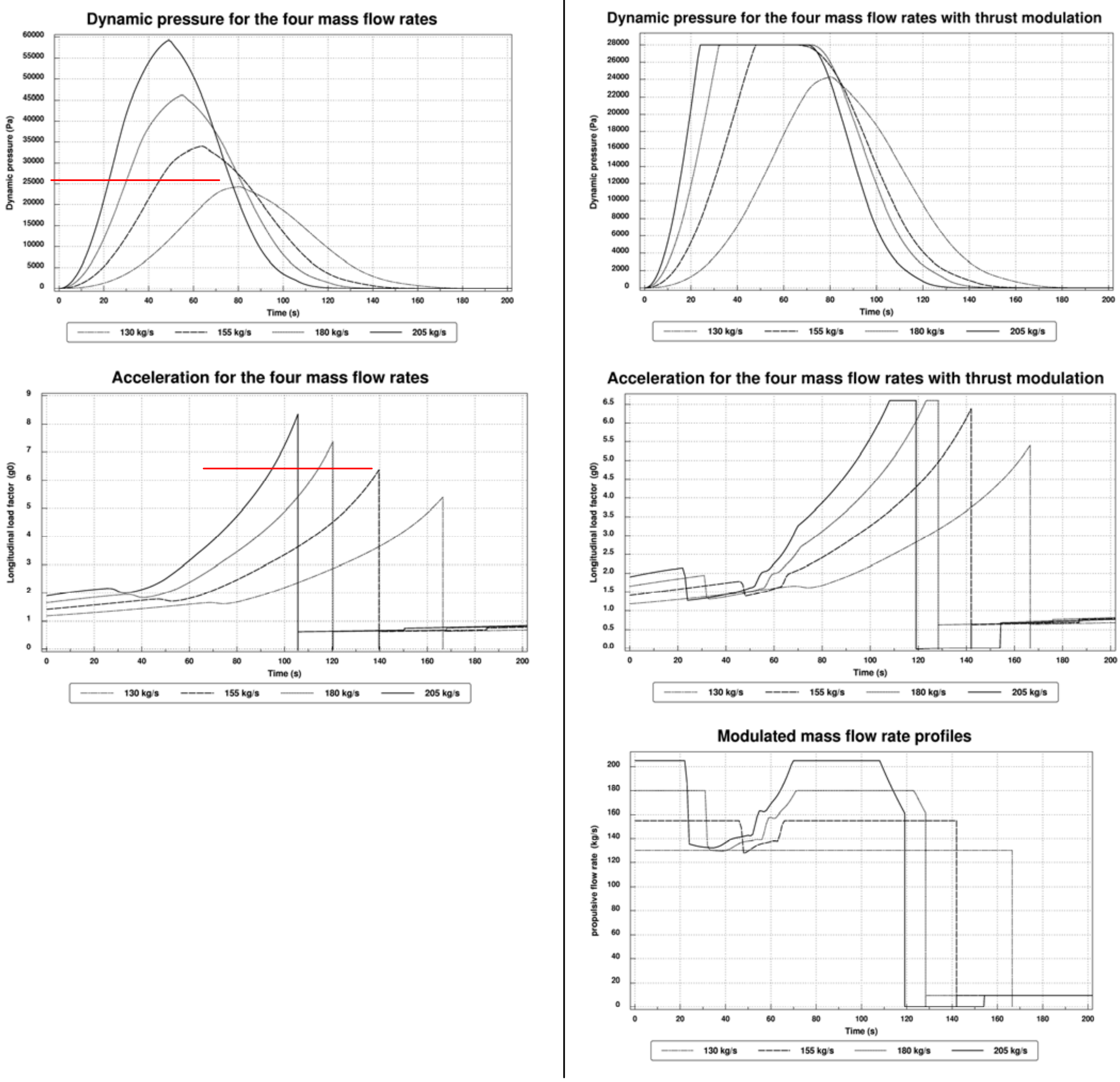

Figure 14

In order to match the FALCON1-type launcher's limits, the engine must decrease its mass flow rate from 100% to 65%, depending on the maximal available flow-rate.
According to the results of the computations, the best and selected configuration of the hybrid 1st stage for a FALCON1-type launcher is a 21653 kg of propellant with 11.72 % of structural index stage. The maximum mass flow rate is 180kg/s and must be reduced to 130kg/s (72%). With this stage, the payload mass is 390kg for a 800 x 160 km orbit, and the FALCON1-type launcher's flight limits are respected.
Here are summarized the required preliminary specifications of the hybrid engine:

Table IX

Parameter	Value
Propulsive propellant mass	21650 kg
Vacuum specific impulse	310s
Max. vacuum thrust	547 kN
Min. vacuum thrust	395 kN
Overall combustion time	128 s
Max. flow-rate	180 kg/s
Min. flow-rate	130 kg/s
Thrust modulation ratio	1.4

The operating point defined here-before by the mission analysis for the hybrid motor is: burn time = 128 s, Isv = 310 s, couple: LOx/advanced solid fuel, the pressurization being ensured by a turbo-pump.
The preliminary sizing taking into account advanced fuel characteristics coming from SAFRAN-SME is given in the next table.

Table X

Component	Mass (kg)
Combustion chamber	12064,3
Solid grain	10800
Inert (insulated case, nozzle, injection)	1264.3
Pressurisation sub-system	775.7
Oxidizer	10800
Tank	324
Helium pressurisation system	34.7
TVC	80
Miscellaneous	130
Total	24208.7
Propellants	21600
Inert	2608.7
Mass index (%)	12.1
Geometry	
Overall diameter (mm)	1700
Solid length (mm)	5515
Liquid length (mm)	4874
Overall length (mm)	11188
Average throat diameter (mm)	226
Nozzle areas ratio (average)	18
Performance	
Pmean (MPa)	6.7
Pmax (MPa)	12
O/F mean	0.97
Isv mean (s)	310
Qtot mean (kg/s)	166.15
Mean regression rate (mm/s) (*)	5
Tb (s)	128

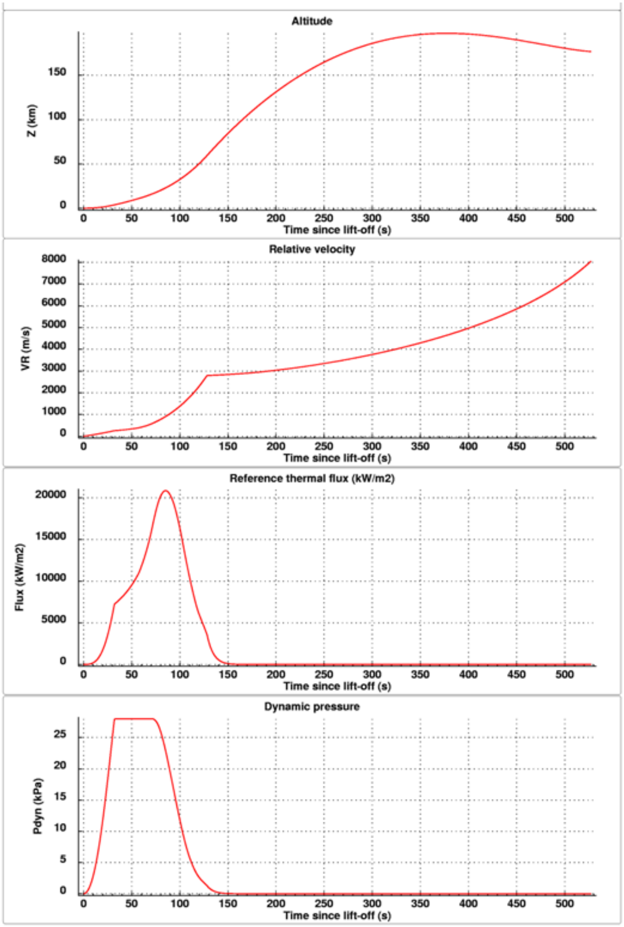

Figure 15

With these propulsion data, a mission analysis was performed in order to evaluate the global performance of the launcher featuring a hybrid first stage.

Table XI

Payload mass	390 kg
Injection apoapsis	800 km
Injection periapsis	160 km
Inclination	98.6°
Max. dynamic pressure	28 kPa
Lift-off mass	29.2 tons

It can be noticed that the launcher performance is increased significantly with this kind of 1st stage. Nevertheless, the different assumptions taken into account have to be consolidated and a global system analysis of the new launcher has to be done in order to have a more accurate performance gain.

The second objective of this activity relative to the first stage was to determine the payload domain keeping the same stage but only changing its operating point in the framework of the reduction of costs.
The following thrust profile leads to a performance of 210 kg in the 160*800km (98.6°) orbit:

- 130 kg/s during 70s
- 90 kg/s during 140s
- Overall combustion time : 210s

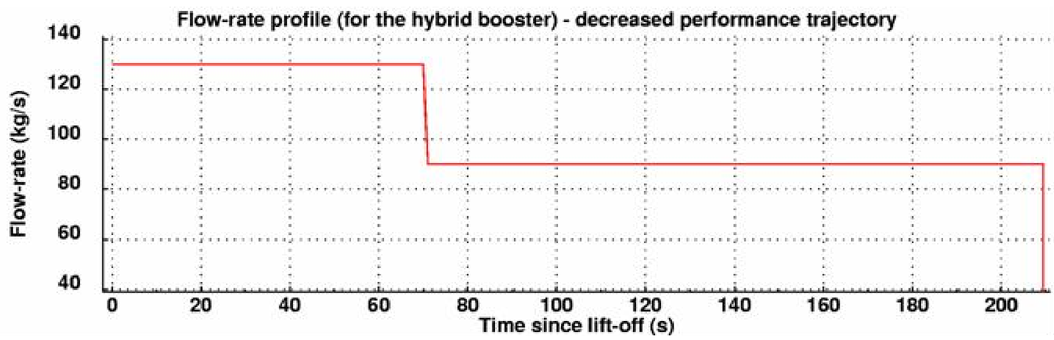

Figure 16

This performance of 210 kg is not the absolute inferior payload mass. However, it is difficult to get satisfactory trajectories for lower flow-rates.
Therefore, it can be concluded that a domain from about 200 to 400kg in SSO transfer orbit can be covered if the following thrust modulation specifications are conceivable:

- applied maximal flow-rate: 130kg/s
- theoretical maximal flow-rate: 180 kg/s
- applied minimal flow-rate: 90 kg/s
- required modulation ratio: 2

A hybrid booster is interesting as regards to structural constraints; in particular dynamic pressure and longitudinal load factor. It is possible to limit these constraints at lower levels, controlling the engine flow-rate, while not decreasing significantly the performance. Indeed, the payload increases of almost 200 kg (+100%) for a Falcon1-type launcher, replacing the reference first stage (LOx/Kerosene) by a hybrid booster, while still matching the same reference structural constraints. This still has to be consolidated as the consequences of the payload increase on the reference upper stage (fairing mass, structural index…) have not been studied yet.
Also, hybrid technology allows versatility as the covered domain (in terms of injected payload mass) in SSO goes, at least, from 200kg to almost 400kg, just modulating the hybrid booster flow-rate.
This design of hybrid propulsion system offers a motor which meets high level requirements defined by mission analysis with simplicity and efficiency. A second loop will be done at the end of **ORPHEE** project with respect to the results of the studies on advanced fuel.

5. Conclusions

This analysis has preliminary designed the hybrid propulsive systems and related stages which could favorably replace the already existing propulsive systems. Here are summarized the different possibilities and performances reachable with the designed hybrid propulsion systems, knowing that the related propulsive specifications are detailed in the present document:

- A HUS (hybrid upper stage) is interesting on top of a small VEGA-type launcher. It allows a significant payload increase of more than 60% (+800kg) in SSO, compared to the performance reached with a combination of a solid and a storable upper stage (both replaced by the HUS).
- Hybrid propulsion fits the requirements of a propulsion system to be used to perform an efficient Moon descent and soft-landing scenario. Using its thrust modulation capability, a hybrid engine allows combining low thrust levels maneuvers for soft-landing and high thrust levels maneuvers for a low gravity-losses descent. With the present assumptions, to soft-land an inert mass of 2 tons on Moon surface from LLO required a hybrid propulsion system of 280kg (included in the landed mass) and a propellant mass of about 1400kg.
- An exploration mission on Mars surface requires an accurate landing. A hybrid engine, through its thrust modularity, can be used to perform pinpoint landing. Flight-proven guidance algorithm such as polynomial laws (used for Apollo missions) results in modulated but continuous acceleration profile. This fits hybrid propulsion. Considering a Phoenix-type Mars descent scenario and an inert landed mass of 350kg, to cover a 10km-radius landing zone required a hybrid propulsion system of 40kg and up to 75kg of propellant.
- A HRB is interesting as regards to structural constraints; in particular dynamic pressure and longitudinal load. It is possible to limit these constraints at lower levels, controlling the engine flow-rate, while not decreasing significantly the performance. Also, the payload increases of almost +200 kg (+100%) for a low-cost Falcon1-type launcher, replacing the semi-cryogenic first stage (Lox/Kerosene) by a hybrid booster, while still matching the same reference structural constraints. Also, Hybrid technology allows versatility as the covered domain (in terms of injected payload mass) in SSO goes, at least, from 200kg to almost 400kg, just modulating the hybrid booster flow-rate.

The table in appendix 2 synthesizes the obtained results in **ORPHEE** project. This table preliminary defines the high level requirements for the advanced hybrid propulsion system for the considered missions, in particular for the advanced fuels.
These results will have to be updated subsequently in **ORPHEE** project, with consolidated hybrid propulsion characteristics: advanced fuel characteristics, experimental specific impulse, quality of the injection... obtained at the end of the project.

Acknowledgement

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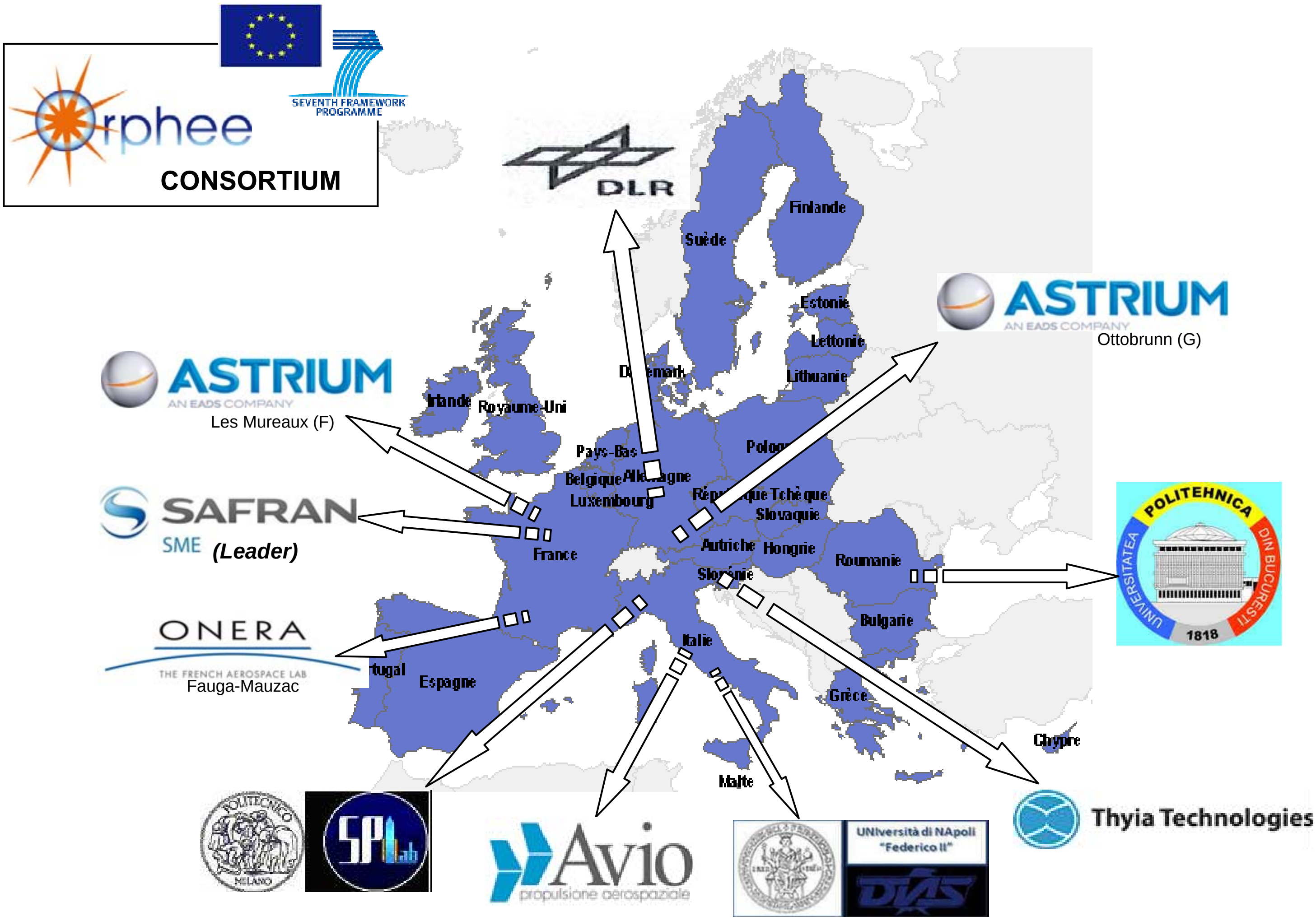
APPENDIX 1 –ORPHEE Consortium



Website: www.orphee-fp7-space.eu/

APPENDIX 2 – Synthesis

Platform / *Engine*	**Hybrid upper stage**	**Hybrid moonlander**		**Hybrid marslander**	**Hybrid booster**
		Central	Outer		
Propulsive characteristics					
Oxidizer	LOX	LOX	LOX	H_20_2 (storable)	LOX
Fuel	advanced solid fuel	advanced solid fuel	advanced solid fuel	advanced solid fuel	advanced solid fuel
Isv at max. thrust (s)	350	350	350	320	310
Maximal thrust (kN) - vacuum	200	24	8	5.2	547
Minimal thrust (kN) - vacuum	-	-	1.1	1.3	274
Regression rate domain (mm/s)	Up to 10	0.3 – 5		0.5 - 5	Up to 15
Modulation ratio	-	-	7,5	4	2
Burning time (s)	180	100	123	< 70	>128
Number of ignition(s)	2	3	1	1	1
Mass characteristics					
Propellant mass (kg)	10500	1378		75	21653
Inert mass (kg)	1588	279		40	2609
Related SI (%)	15.1	20.2		52.8	12.1