Elements for propellants requirements

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ABSTRACT

The objective of the study was to define propellant requirements. The propellant identified requirements will guide the development in the project of a suitable formulation with a high specific impulse.
Generally to compare new propellant the product ρ.Is is used, this way to proceed is not really relevant, if the effect the specific impulse is always of first importance, the effect of density and of burning rate on the lay-out is important moreover the result is very different if is considered a first stage or an upper stage. So, to be pertinent an advice have to be funded on an analysis taking into account as much as possible the real context

Three reference missions were selected:

- Upper stage of Vega launcher
- Apogee/deorbitation motor
- Mars ascent vehicle

Based on ADN and GAP and an energetic fuel (Aluminium or Alane), two “Ideal” propellants were defined. Their potential performance increases were quantified in respective mission by a comparison –using fist design and trajectory tools-with references cases and some of their required properties and a domain of formulation were identified.
The most important properties are:

- Burning rate
- Mechanical properties
- Specific impulse

The first mandatory point is to be able to obtain a propellant with a basic burning rate in the range of 7 to 15 mm/s at 7 MPa. This range could probably be extended for some applications when high acceleration can be accepted.
The second prerequisite is to obtain a propellant with good mechanical properties, at least of the level of classical HTPB propellants, to enable a case bonded grain. For space applications end burning grain will also be a common situation even if relatively low burning rate propellants are achievable, so an axis of work would be to look for propellant with mechanical properties much better than the current one so to be able to realize full bonded end burning grain

These requirements satisfied, the only important parameter is the level of practical specific impulse. With the formulations understudy the performance gain could be dramatic with a potential increase up to more than 30% of the payload of the Vega Launcher by replacing only the propellant of the third stage. For a Mars Ascent vehicle the saving on the lift-off mass could be also impressive but in such an application (launcher of small size) to obtain compatible burning rate level is of first importance and a more detailed system analysis has to be performed.

Acronyms

AP	Ammonium Perchlorate (NH_4ClO_4)
GAP	Glycidyl Azide polymer
AND	Ammonium Dinitramide
HTPB	Hydroxyl-terminated polybutadiene
Alane	Aluminium Hydride AlH3
I_{SP}	Specific Impulse, s

1. Introduction

The objective of the study was to define propellant requirements. The propellant identified requirements will guide the development in the project of a suitable formulation with a high specific impulse.
Generally to compare new propellant the product ρ.Is is used, this way to proceed is not really relevant, if the effect the specific impulse is always of first importance, the effect of density and of burning rate on the lay-out is important

moreover the result is very different if is considered a first stage or an upper stage. So, to be pertinent an advice have to be funded on an analysis taking into account as much as possible the real context by using a first design method.
Three reference missions were selected:

- Upper stage of Vega launcher
- Apogee/deorbitation motor
- Mars ascent vehicle

1. Performances competition

Based on ADN and GAP and an energetic fuel (Aluminium or Alane), new propellants have to be better than the current propellants for selected space applications.

Propellant		Mixture Ratio	Equivalent Density (kg/m^3)	Is th (Pc 7MPa, Σ=40)
Solid (AP/Al/HTPB)		68/18/14	1750	315
Hybrid (NTO HTPB)		80/20	1280	329
Hybrid (H2O2 HTPB)		82/18	1320	328
Liquid Bi Prop	NTO/MMH	2.37	1200	341
	H2O2/RP1	7.0	1320	314
Monoprop : N2H4			1020	230

Table 1: Main storable competitors including a reference solid propellant.

In term of intrinsic performance a potential formulation without ejection of particles of alumina may exist around 17% of GAP, Versus a non aluminised HTPB/AP , a GAP/ADN is always better (see figure 2) .Versus a reference Aluminised HTPB/AP propellant, its theoretical specific impulse is quite equivalent (313 vs 315) but with a lower combustion temperature (3130 vs 3400 K) leading to lower thermal loads. The density is only slightly lower. If the two phase flow losses are taken into account, such propellant will become better than the reference HTPB propellant; these two phase flows are scale sensitive and to be estimated need to have a method that take into account the alumina particles sizes and to know these one.
For small size of throat, the 2 phase flow may reach a tenth of seconds, so if it is needed to improve the mechanical properties of the propellant, the amount of GAP may be probably increased in the range 17-25% (to be verified by tests/computations), this propellant without aluminium could remain equal or be better than the reference solid propellant Nevertheless, a formulation GAP/ADN will be competitive with a HTPB propellant without Aluminium of about 25s, taking into account a minimum percentage of 14% of HTPB

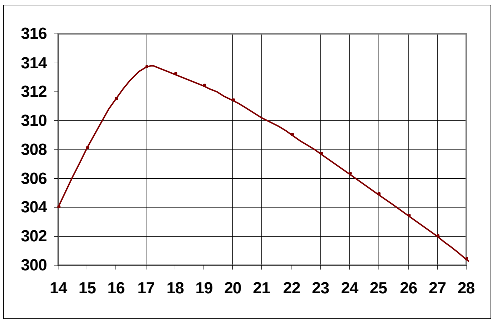


Figure 1. Specific Impulse (s) as function of the percentage of GAP (Area ratio 40, Pc 7 MPa).

H2	0.6222
H2O	1.2645
N2	1.5522
CO	0.1762
CO2	0.6300

Table 2: Exhaust products: 25%GAP 75%ADN (moles/100g)

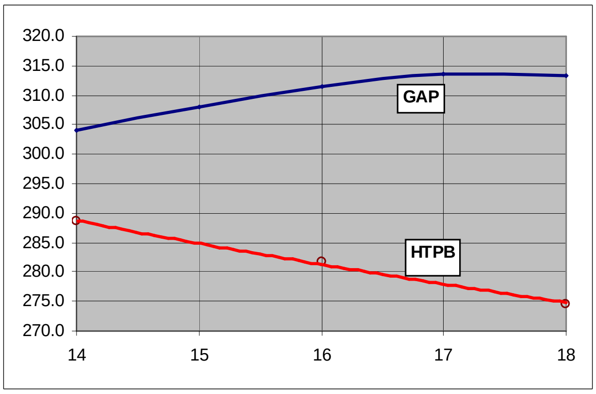


Figure 2. Specific Impulse (s) as function of the percentage of GAP or HTPB (Area ratio 40, Pc 7 MPa) for a GAP/ADN and a HTPB/AP propellant.

Figure 3 shows the theoretical Specific Impulse (equilibrium conditions), for an area ratio of the nozzle of 40, of propellant with aluminium an intrinsic optimum exist in a range of 16-20% of aluminium and 20-24% of GAP. For small throat space motor, the optimum will be found for lower amount of aluminium (optimum taking into account the two phase flow)
With Aluminium the important percentage of GAP may probably lead to propellant with good mechanical properties

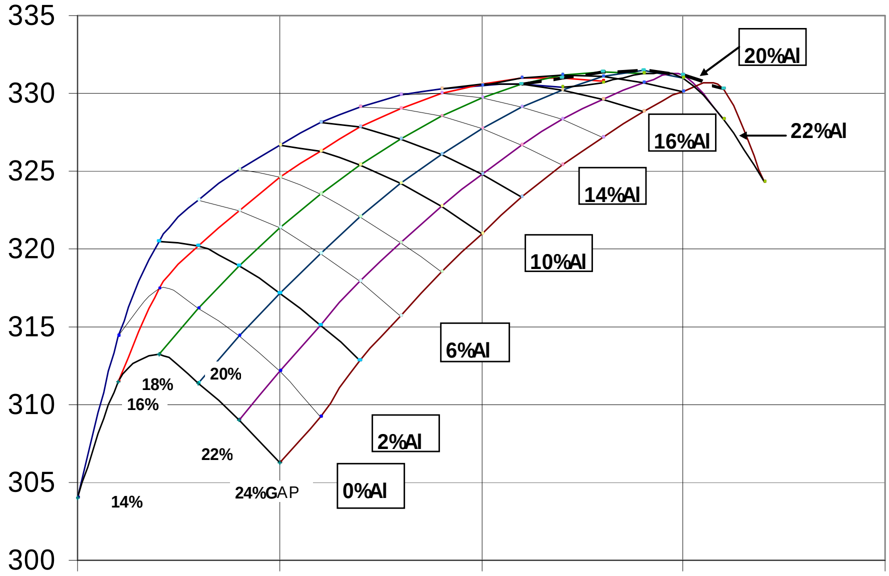


Figure 3. ADN/GAP/Aluminium: Is theoretical (Σ=40, Pc 7MPa) function of percentage of GAP and Al.

With Alane , if an intrinsic optimum exists it will be for a low percentage of GAP and a high percentage of Alane ; at this stage for a priori feasibility reasons a minimum percentage of GAP(13%) was retained

So, for the applications studies two references propellants were selected

Table 3: ADN/GAP/Fuel: “ideal propellant” theoretical performances.

GAP/ADN/Fuel	Fuel	d (kg/m^3)	Is 40 (s)	Cd (s/m)
18/65.5/16.5	Al	1772	330.0	0.000606
13/60.5/26.5	Alane	1631	346.8	0.000574

H	0.0963
H2	0.8126
H2O	0.7188
N2	1.3029
CO	0.4899
CO2	0.0906
Al2O3	0.3058

Table 4: Exhaust products: 18%GAP 65.5%ADN 16.5% Al (moles/100g)

2. Evaluation of Performances on Reference Missions

2.0 Methodology

The reference missions will be studied using design tools and trajectory codes (earth to orbit or trajectory from Mars to a Rendez-vous orbit) or ΔV analysis for Apogee motors. This implies not only propulsion system design but the complete spacecraft must be taken into consideration. The computational design tools to be used in the work include an in-house model, for solid rocket motor design (SOME), an in-house model for liquid rocket engines (PLISE), and a computer code PERFOL for trajectory calculations developed by the company SISOP for The Inner Arch [1].
Three reference missions were identified where the use of high performance solid propellants will have large impact on mission benefits in comparison with existing solutions, such as conventional solid propellant (AP/HTPB/Al) and liquid bipropellants (MON/MMH). The three reference cases selected were:

- ·Upper stage of Vega launcher (Reference Case Z9A and Z9B)
- ·Apogee/deorbitation motor
- ·Mars ascent vehicle

The first stage is to calibrate the codes on the reference case using the literature data
The two ideal propellants selected in Table 3 were used in respective mission and the performance was evaluated. If otherwise not stated, it was assumed at this step of the study, that these propellants hade the same burning rate and the same pressure exponent as classical HTPB propellant. It is also assumed identical motors from the others characteristics point of view (grain shape). With these assumptions, a maximum gain of performances was estimated.

2.1 Vega Third Stage

Reference Case Z9A and Z9B

In a SRM the combustion time is strictly dependant of the thickness of propellant to burn (“web”). Nominal Z9A is a finocyl grain configuration with a radial combustion. Taking into account this configuration to respect a specification of an acceleration of 5g maximum is leading to the choice of a low burning rate propellant (about 7mm/s).
The new propellants understudy may have a higher range of burning rate, so the first step for this application is to determine if a configuration of grain may allow using high burning rate propellants and in which range. So, a solution with an end burning grain with the greater possible web was envisaged (intermediate solutions are however also possible).
Z9A is the nominal motor of the third stage of the Vega launcher.
Z9B is a head-end grain solution allowing using a propellant with a high burning rate (18 mm/s instead of 7 mm/s) and using a much less erosive propellant. The design is made to obtain the same maximum acceleration during the Z9 flight. This solution imply a rear igniter (like on the ATK Star motors) and a special design of internal insulation.

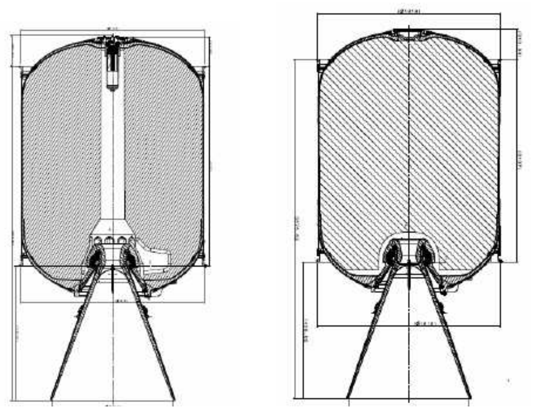


Figure 4. Current Z9A (left) and alternative with an end burning grain Z9B (right).

Table 5. Functional Characteristics.

	Z9A	Z9B	
Action Time	117.1	121.1	s
Total propellant mass	10568	10596	kg
MEOP	8.59	8.59	MPa
Exit Dia.	1273	1273	mm
r (7 MPa)	7	15.2	mm/s
Average Area ratio Σ	52.2	56.5	
Is on Action Time	295.2	296.2	s

Performance of Vega launcher with the two reference motors

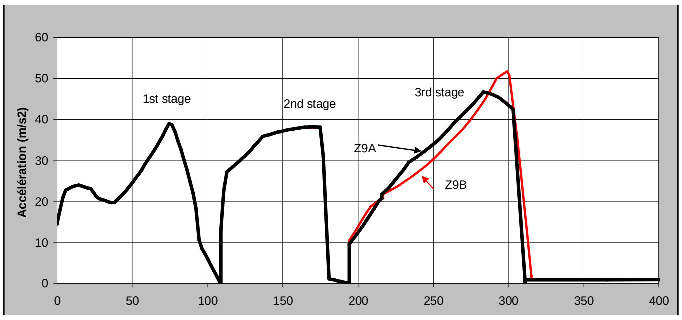


Figure 5. Z9 acceleration vs time (in red Z9B).

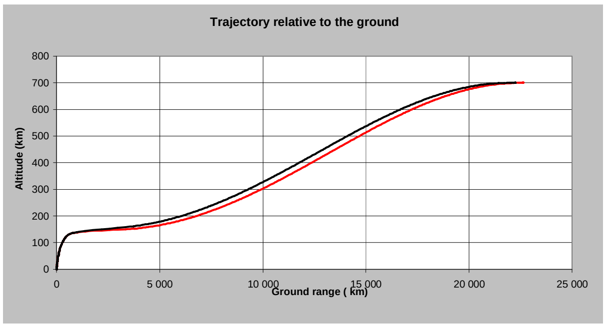


Figure 6. Altitude vs ground range (in red Z9B).

The performances of the two versions are equivalent

As input of SOME model of Z9A the propellant is replaced by an “ideal one” and then the performance gain is evaluated with PERFOL (Trajectory code).

Table 6. VEGA potential global performance increase with a new propellant.

			Nominal Is losses			No two phase flow losses		
Case	ΔMi	ΔL	ΔIs	Δpayload		ΔIs	Δpayload	
	kg	mm	s	kg	%	s	kg	%
VEGA Z9A	0	0	0	0	0	15.7	159	11
"Ideal Alu"	10	35	13.8	193	13	29.4	430	29
"Ideal Alane"	81	264	22.8	249	17	45.4	610	41

All the motors have the same average area ratio and burning time, Pmax , average area ratio and diameter. The propellant is replaced by the ideal one, with the same amount of propellant mass, so the length of the motor is increasing and so is its dead mass.
The first column of ΔIs is estimated with two phase flow semi empirical model of [18].In the second one these losses are not taken into account assuming a perfect combustion producing very small alumina particles.

Conclusion on Vega

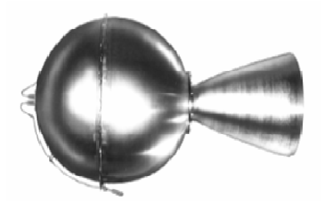
The two configurations are giving roughly the same level of performance in terms of payload into orbit. With an End Burning grain the **useful burning rate could be extended from 7 to 15 mm/s** with the same case (L/D of the grain). Such a configuration, usual on the Star motor from ATK, would however need special developments (Internal thermal insulation, igniter, etc). The interest to lower the burning rate such is to remain in a range of L/D allowing a radial burning grain configuration.
In the case of Vega 3rd stage the effect of the Specific impulse could be dramatic with a potential increase up to more than 30% of the payload and so to lower the cost of the kg into orbit.

2.3 Deorbitation motor

Typical debris to deorbit is weighting around 1800-2000 kg with an apogee around 800 km (Ariane stages, SPOT, Russian stages…). The ΔV needed for a direct deorbitation is of 260-280 m/s [2]. Among many potential solutions, a kit plug on this debris may include a solid rocket motor, as for instance the ATK Star motor, an attitude control system and a connection system. With the assumption of a total mass of the deorbitation kit of 350 kg, including a Star 24 motor, it will be able to deliver the required ΔV. The STAR 24 [3] rocket motor was qualified in 1973

Propellant			Nominal Is losses		No 2Φ flow	
	ΔMi (kg)	(mm)	ΔIs (s)	ΔV (%)	ΔIs (s)	ΔV (%)
TPH 3062[a]	0	0	0	**0**	13.5	**11**
Ideal Alu	0.3	16	12.9	**13**	26.6	**29**
Ideal Alane	1.5	64	22.5	**17**	42.5	**41**

Table 7. Deorbitation Kit. Potential global performance increase with new propellants.



2.4 Mars Ascent Vehicle

The Mars Ascent Vehicle, MAV, has to return on a 500 km circular orbit at an inclination angle in a range of 45°-0.2° at a latitude of 30° North, with a payload of 36 kg, including comprised of a 5 kg orbiting sample (OS) [4], plus 31 kg which includes the OS interface and separation mechanisms, avionics (Attitude Control System (ACS), Command & Data Handling (C&DH), power), telecomm, cabling, thermal control, structure, a reaction control system, and a 3 kg contingency. This payload is transferred to the vehicle returning to the Earth ‘ejection in a basket. The MAV propulsion system must survive a long period in the Mars environment with minimal thermal conditioning, potentially up to one Martian year [5]. Both solid and liquid propellant version of the MAV has been considered, shown in Figure 7 .

Figure 7. MAV solid (left) and liquid (right) versions. Image credit: NASA/JPL and ESA respectively.

Baseline storable liquid

This system uses the conventional storable propellant combination of Nitrogen Tetroxide (NTO) and Monomethyl hydrazine (MMH).

Our model of design of liquid stage was calibrated using a design described in [6] where a relatively detailed mass breakdown is given, the design used component masses based on existing off-the shelf hardware designs and used main engine mass and performance data based on slight modifications of the existing Kaiser Marquardt R-40B rocket engine. For a useful propellant mass of 153 kg, the dead mass is 43 kg, Figure .

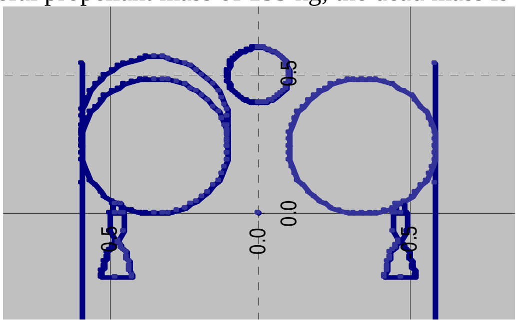


Figure 8. Scheme of a 153 kg of propellant O/F 2.1.

After this calibration phase, coupling the results with a trajectory code, a possible configuration answering to the requirements was defined. It is a two stage configuration.

For both stages, engines are starting at full thrust and ended in a blow-down mode to half thrust.

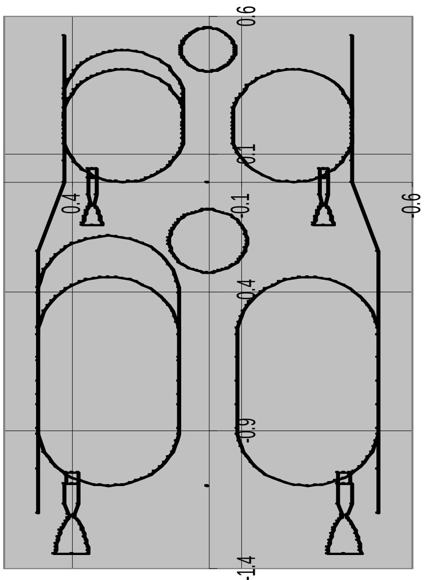


Figure 9. MAV configuration.

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Characteristics of the second stage

Useful Propellant: 69 kg
Diameter: 0.85 m
Length: 0.65 m
4 motors 200 N ATV or equivalent (extended nozzle)
4 Titanium Tanks
Pressurised with Helium- Blow down mode
Total Inert mass: 30 kg
Is = 312 s
On-off control

Characteristics of the first stage

Useful Propellant: 208 kg
Diameter : 1.0 m
Length : 1.15 m
4 motors RD4 ATV or equivalent (extended nozzle)
4 Titanium Tanks
Pressurised with Helium- Blow down mode
Total Inert mass: 52 kg
Is = 308 s
On-Off control

Performance analysis

The data used for the trajectory code are issued of [9].To go into a 500 km circular orbit a □V of 4200 m/s is needed. The flight sequence include a coast phase during the flight of the second stage of 3303 s. The results are presented in Table . The payload found is **43 kg**; the margin with regards to the specification is 7 kg.

Table 8. Synthesis of Liquid MAV performances.

Initial mass (ton)	Payload(kg)	PdynSep 1-2 (kPa)	Flux max (kw/m2)	Accel max (m/s^2)	Pdyn max (kPa)	Z apogee (km)	Z perigee (km)	Inclination (°)
0.40	43.4	0.3	7.7	13.0	0.7	499.5	498.5	30.0

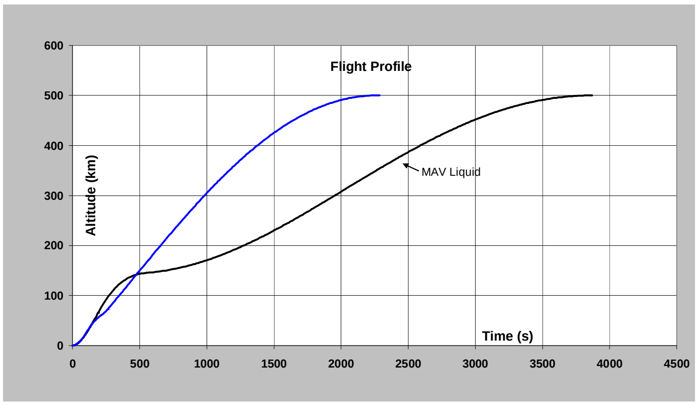

Figure 10. Comparison of the trajectories: MAV Liquid (black) and Solid (blue)..

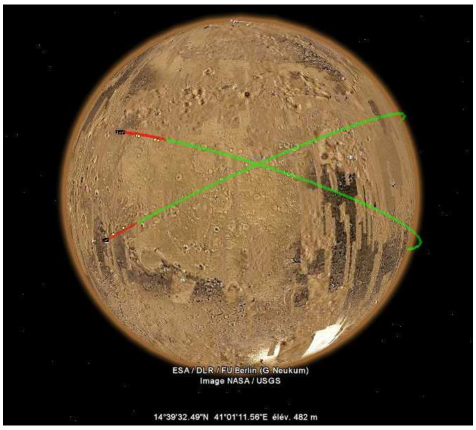

Figure 11. Google view of the MAV trajectory

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Baseline classical solid

The baseline solid MAV version studied by NASA [9] is shown in Figure .

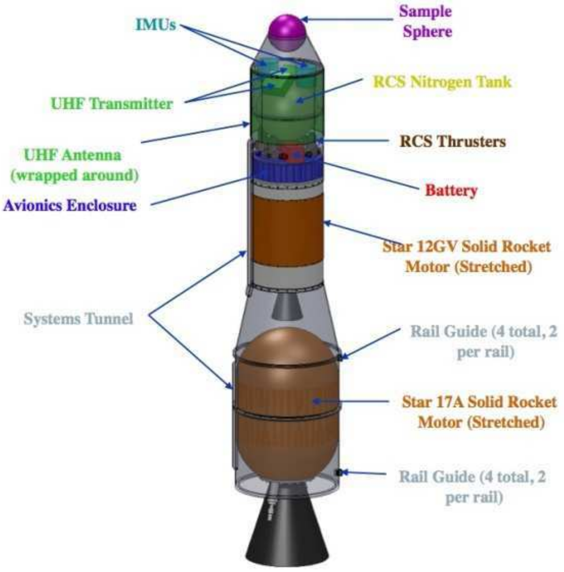

Figure 12. NASA baseline solid MAV [9]. Image NASA.

The nominal Star 12 GV is loaded with 32.9 kg propellant (Mi=9 kg, Is=284.7 s, Tc 13.9 s) but for this mission it has to be stretched. The nominal Star 17A is loaded with 112.3 kg of propellant, (Mi=20.8 kg, Is=286.2 s, Tc=18.6 s). Similarly as for the second stage it has to be stretched. The second stage is equipped with a flex seal so the stage can be controlled in pitch and yaw, the first stage has a fixed nozzle; so the compliance with the mission is a question mark.

Based on the NASA Configuration the first step is to design a configuration equivalent in performance to the liquid baseline with some differences:
The final orbitation is operated by a bi propellant ACS (NTO/MMH instead of nitrogen) using 3 kg of propellant for this phase, this ACS provides a 3 axis control; its dead mass is included in the payload mass.
The first and the second stages are equipped with a flexseal allowing to control the MAV in Pitch and Yaw (so the launch can be a vertical launch with a tilt-off manoeuvre); the flexseal configuration is an upstream centre of rotation.
The material of the case is a Titanium alloy and the propellant is assumed to be the TPH 3062 (see characteristics $2.3) burning at 7.38mm/s.
Such a configuration using short burning time solid rocket motors needs to introduce coast phases between the first and second stage burning phases (performance improvement around 6%), possible on Mars taking into account the local atmospheric conditions.

The model of SRM is calibrated on the Star technologies.

Data and layouts for the two stages are shown in Figure to Figure .

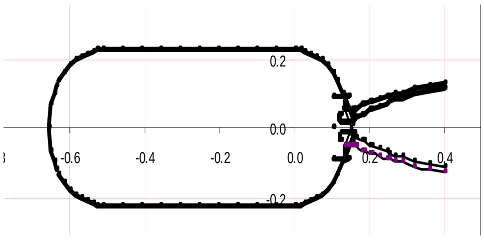

Figure 13. First stage 165 kg of propellant. Diameter 0.45 m, Length 1.060 m, Dead mass 27 kg, Ae/At=70 Ism 292.0 s, Tcu 100 s, Pc 6 MPa.

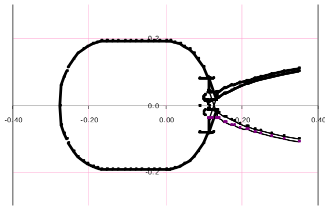

Figure 14. Second stage 50 kg of propellant. Diameter 0.38 m, Length 0.625 m, Dead mass 11 kg, Ae/Atm80, Ism 293.5, Tcu 50 s, Pc 5MPa, constant mass flow rate.

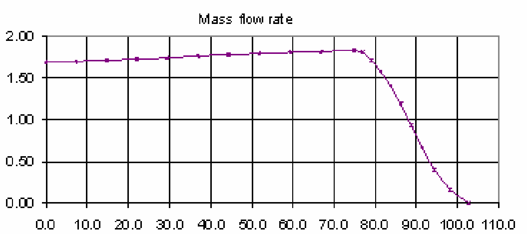

Figure 15. First stage 165 kg of propellant Mass flow rate law.

Table 9. ΔV and losses (m/s).

n° Stage	Propulsion	Drag	Gravity	Thrust orient
1 (α=0)	540.3	-7.2	-24	-62
1	1547.2	0	-48.2	-10.1
2	1556.6	0	-77.5	-4.3
3	154.9	0	-404.9	-53.9
SUM	3799	-7.2	-554.6	-130.3

Performances Analysis

Table show the ΔV delivered by the stage and the OCS (stage 3) and the level of losses. The total ΔV needed is lower than for the liquid solution (3800 m/s instead of 4200m/s), which it is a consequence of the higher level of thrust.
Taking into account the low atmospheric pressure level on Mars, the trajectory begin by a short phase with a 0 angle of attack (Dynamic pressure and fluxes limitations). After the first stage burn-out, a first coast phase is introduced and a second one after the second stage burn-out, as shown in table 10.
The mission is achieved with a MAV weighting roughly 320 kg lighter than the liquid one. The maximum acceleration is lower than 5g so it could be envisaged to reduce the burning times using high burning rate propellants. The launcher itself, due to its small size, its electronics and the payload itself, may support acceleration greater than 10gs. The attitude of the launcher has to be controlled during the coast phases
The trajectories of the liquid and solid MAV are shown in **Erreur ! Source du renvoi introuvable.**0.The synthesis of the solid baseline MAV performances is shown in table 10**Erreur ! Source du renvoi introuvable.**
MAV being brought from the Earth, the driving parameter is the lift-off mass. Use of a solid may lead to an interesting weight saving versus a liquid solution even if, each solution have to be studied in detail and optimised to conclude definitively (320kg compared to 400kg see table 10).

Solution with "Ideal Propellants"

Data and the layouts for the two stages solid version using "ideal propellant 1" are shown in Figure and Figure , and the layouts for the two stages solid version using "ideal propellant 2" are shown in Figure and Figure . The comparison is made keeping the same diameter of the motors and the burning rate of the TPH 3062 propellant.

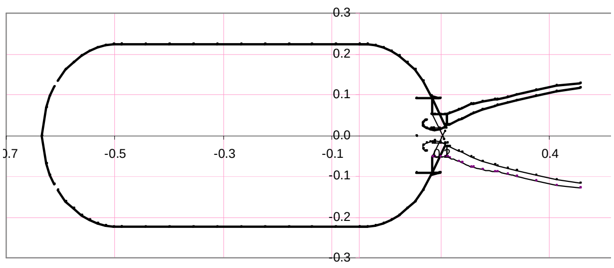

Figure 16. Ideal 1 First stage 144 kg of propellant, Diameter 0.45 m, Length 1..000 m, Dead mass 21.4 kg, Σm70 Ism 308.3 s Tcu 100s, Pc 5 MPa.

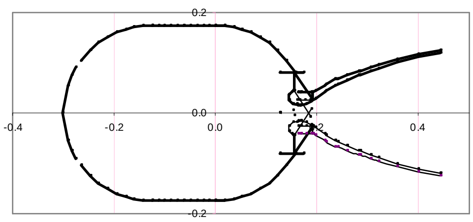

Figure 17. Ideal 1 Second stage 45 kg propellant, Diameter 0.35 m, Length 0.730 m, Dead mass 7 kg, Σm60, Ism 303.1 s, Tcu 21s, Pc 5MPa.

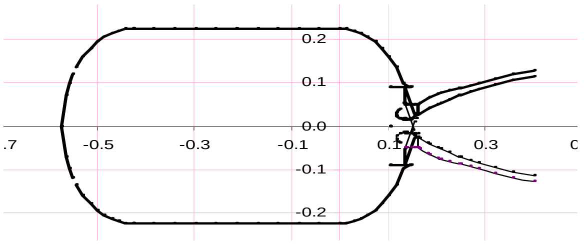

Figure 18. Ideal 2 First stage 138 kg of propellant Diameter 0.45m Length 1.010 m, Dead mass 23.5kg Σm70 Ism 314.2s Tcu 104s Pc 5MPa.

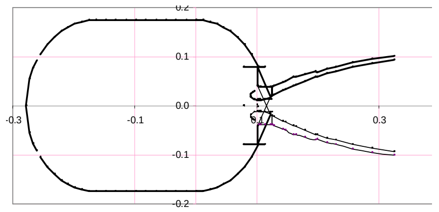

Figure 19. Ideal 2 Second t stage 40kg of propellant Diameter 0.35 m Length 0.600 m, Dead mass 7.66 kg Σm60 Ism 311.5 s, Tcu 19 s, Pc 5MPa.

The results in Table 10 show that with the "ideal propellant 1" a gain on the lift-off mass of 30 kg is found, with the "ideal propellant 2" a maximum additional potential gain of 15 kg is possible.
From the Specific impulse aspect, the model of losses is assuming an identical level of losses resulting of the two phase flow to a classical HTPB propellant. Additional gain is expected by using nano aluminium and Alane, since the 2 phase flow losses is expected to be much lower.
The maximum potential gain is obtained with the propellant "ideal 2 "assuming no 2 phase flow losses. The mass saving is then 60 kg on a reference mass of 320 kg.
During the HISP program, it could be useful to have a better estimate of the losses resulting of the two phase flow (it is unrealistic to have no two phase flow losses)

3. Conclusions

The first mandatory point is to be able to obtain propellant with a basic burning rate as low as possible at least in the range 7 to 15 mm/s (at 7 MPa) with a tuning capability.

Another prerequisite is to obtain a propellant with good mechanical properties, at least of the level of classical HTPB, propellant of class 1.3

For in-space applications end burning grain will be a common situation even if relatively low burning rate propellants are achievable. So an axis of work would be to look for propellant with mechanical properties much better than the current one to be able to realize simple full bonded end burning grain

These requirements satisfied, the only important parameter is the level of practical specific impulse. With the formulations understudy the performance gain could be dramatic with a potential increase of more than 20% of the payload of the Vega Launcher by replacing the propellant of the third stage. For a Mars Ascent vehicle the saving on the lift-off mass could be also impressive but in such an application (launcher of small size) to obtain compatible burning rate level is of first importance and a more detailed system analysis has to be performed.
To have a feasible propellant with a reasonable burning rate could decrease the expectation of gain (ballistic catalyst , mechanical additives, limits on the composition,…). The two phase flow losses will depends of the size of the produced alumina droplets of a given propellant; one aim of the programme will be to try to quantify the level of these losses.

Acknowledgement
The author likes to thank Gino Fundarò at AVIO, for providing data on the Vega Z9.

Table 10. Comparison between nominal and "ideal" propellants (Performances circ. orbit 500km).

				Reference	
	Mp (kg)	Mi (kg)	Tc (s)	Thrust	Ismean (s)
1st stage □=0	55	0	28.3	constant	290.7
1st stage	110	34	58.7	constant	292
2nd stage	50	11	42	constant	292.9
OCS	3	0	50	constant	300
	Initial Mass (ton)	Payload (kg)	Flux max (Kw/m2)	Accel max (m/s2)	Pdyn max (kPa)
	0.319	55.5	80.5	49.2	5.6
				Ideal 1	
	Mp (kg)	Mi (kg)	Tc (s)	Thrust	Ismean (s)
1st stage □=0	47	0	33	constant	306.4
1st stage	97	29.6	67	constant	308.3
2nd stage	45	10.1	22	constant	307.3
OCS	3	0	50	constant	300
	Initial Mass (ton)	Payload (kg)	Flux max (Kw/m2)	Accel max (m/s2)	Pdyn max (kPa)
	0.286	54.2	79	91.5	5.3
				Ideal 1 no 2Φ losses	
	Mp (kg)	Mi (kg)	Tc (s)	Type	Ismean (s)
1st stage □=0	43	0	33	constant	319.7
1st stage	89	27.5	67	constant	321.8
2nd stage	39	9.2	22	constant	323
OCS	3	0	50	constant	300
	Initial Mass (ton)	Payload (kg)	Flux max (Kw/m2)	Accel max (m/s2)	Pdyn max (kPa)
	0.266	55.6	83.1	82.8	5.3
				Ideal 2	
	Mp (kg)	Mi (kg)	Tc (s)	Type	Ismean (s)
	Mp	Mi	Tc	Type	Isp_mean
1st stage □=0	46	0	34	constant	311.8
1st stage	92	31.1	68	constant	314.2
2nd stage	40	9.6	50	constant	315.1
OCS	3	0	50	constant	300
	Initial Mass (ton)	Payload (kg)	Flux max (Kw/m2)	Accel max (m/s2)	Pdyn max (kPa)
	0.275	53.7	77.1	37.3	4.9
				Ideal 2 no 2Φ losses	
	Mp (kg)	Mi (kg)	Tc (s)	Type	Ismean (s)
	Mp	Mi	Tc	Type	Isp_mean
1st stage □=0	45	0	34	constant	334.5
1st stage	90	30.3	68	constant	336.9
2nd stage	28	8.6	50	constant	337.9
OCS	3	0	50	constant	300
	Initial Mass (ton)	Payload (kg)	Flux max (Kw/m2)	Accel max (m/s2)	Pdyn max (kPa)
	0.26	54.9	123.8	35	6.8

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