A trendsetting Micro-Launcher for Europe

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ABSTRACT

In this paper we analyse a potential, future European launcher for micro payloads (200kg in LEO). This study comprises ascent trajectory calculation, aerodynamics and mass estimation. The most remarkable feature is the reusable first stage (RFS). Therefore, beside the ascent analyses, reentry calculations have also been performed. To bring the first stage back to an airbase we suggest in-air capturing. In addition to the technical performance analyses we discuss the potential this kind of launcher has for European space transportation.

Subscripts, Abbreviations

LFBB	Liquid Fly Back Booster
SSO	Solar Synchronous Orbit
SRM	Solid Rocket Motor
RLV	Reusable Launch Vehicle
STO	Sun-Synchronous Transfer Orbit
RFS	Reusable First Stage
CoG	Centre of Gravity

1 INTRODUCTION

Market analyses [1] for satellites show that there is potential for a growing number of compact satellites in the range of 100 - 250 kg. So far only few launcher Systems are suited for carrying that class of payload into orbit, and there are no European ones amongst them. To correct this deficiency CNES started the ALDEBARAN study [2] in which several European agencies and numerous industrials have been invited to present their idea for a new European micro launcher.

Subject of this paper is a launcher configuration which resulted from SART's contribution to ALDEBARAN. It has to be noted that the whole concept is driven by two distinguished premises dictated by ALDEBARAN; the concept shall be technologically innovative and result in low operational costs. The strongest emphasis is on the low cost requirement.

The launcher system presented has a reusable first stage and a solid rocket motor as second stage. The launcher takes off vertically and lands horizontally. Given the above mentioned requirements in ALDEBARAN we restricted ourselves to existing engines. The motivation of choosing such architecture is based on two arguments:

a) The first stage is based on the thoroughly studied boosters of LFBB [3]. Hence, we can profit from this former study (e.g. wind-channel measurements, CFD-Calculation and sound mass analyses).

b) We are convinced that it is reasonable to enter the sophisticated field of RLVs with a small launcher.

This paper presents a comprehensive system analysis of an innovative micro launcher for a STO reference Mission from Kourou. Based on the results we will evaluate the concept in the context of the economical aspect and technological benefits. The recovery of the first stage is also part of the study.

2 REFERNCE MISSION

The reference mission was predetermined in ALDEBARAN [2]: A micro satellite shall be launched in into a SSO-Transfer Orbit. We aimed for a payload of at least 200 kg. The exact conditions can be found in Table 1.

Place	Kourou
Azimuth	100°
Transfer-Orbit	50 km x 800 km
Final-Orbit	800 km x 800 km
Inclination	98°

Table 1: Reference Mission

To reach the final orbit a Δv of at least 210m/s is required. The circularization of the payload can be performed with a suitable apogee motor which will not be discussed in detail in this paper. However, a special kick stage with a ceramic combustion chamber for RFS was subject of a separate investigation. More on the ceramic combustion chamber (cryogenic kick stage) can be found in [5].

3 LAUNCHER ARCHITECTURE

As stated earlier during the launcher design process we restricted ourselves to existing European engines. The advantage of this approach is the potential reduction of development cost and independency on foreign suppliers. Figure 1 shows a sketch of the launcher. It consists of two stages. The first one is winged and reusable. It is propelled by two modified Vinci engines. The second stage is made of a solid rocket motor. Even two Vinci aggregates are too weak to deliver the necessary lift-off thrust, hence additional strap-on booster are inevitable. The next table gives an overview of the launcher architecture.

	Engine type	burn time [s]	Mstruc [t]	Mt [t]	Mstage [t]	Struc. Index [-]	Length [m]	Diameter [m]
1. Stage	2 mod. Vinci	190	4.6	15.6	20.8	0.28	19.8	2
2. Stage	solid	190	0.46	2.3	2.8	0.2	1.5	2
Booster (each)	solid	125	0.64	4.7	5.3	0.136	7.8	0.55

Table 2: Launcher architecture

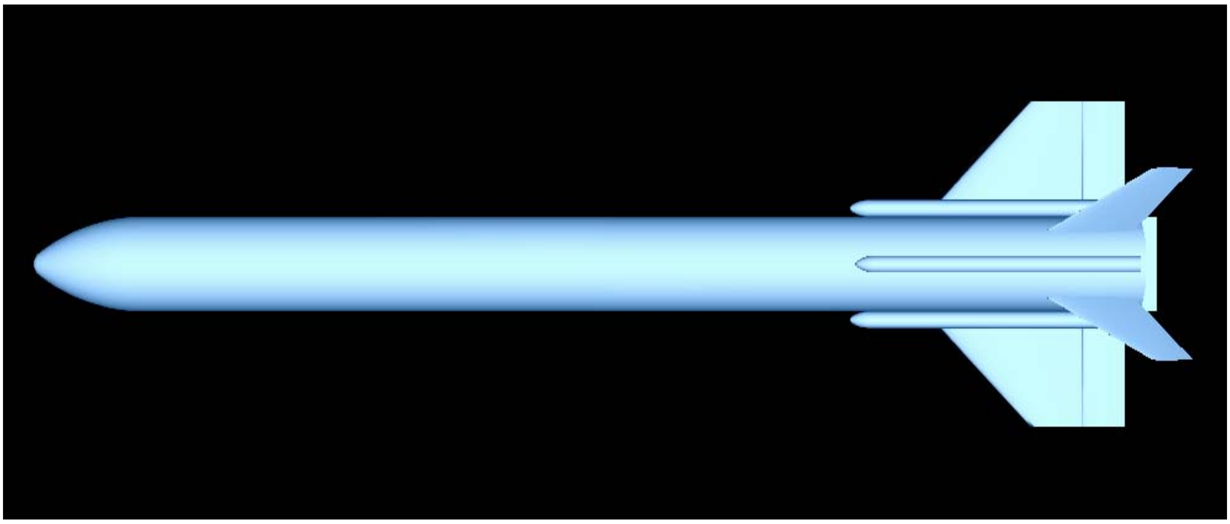


Figure 1: Sketch of RFS

The remarkably low structural index of the reusable first stage may be explained by the omission of air breathing engines for the return. In-air capturing requires only little more mass in comparison. The gross lift off mass of the launcher is **40 t**, the total length is 25m and the wing span 6.95m. A more detailed mass break down of the stages and booster together with the geometry can be found in Appendix A.

3.1 Engines

Because the Vinci engine is not designed to operate on ground-level conditions the nozzle has been shortened. A performance comparison of the original and modified version is shown below

	Exp.Ratio [-]	Ispvac [sec]	Thrust sea [kN]	Thrust vac [kN]
Original	240	465	-	180
Mod.	30	431	126	173

Table 3: Specification of modified Vinci Engine

The latest specifications of the Vinci engine can be found in [6]. For the three strap-on Booster we assumed a generic thrust-profile and a specific sea level Impulse of 255 sec.

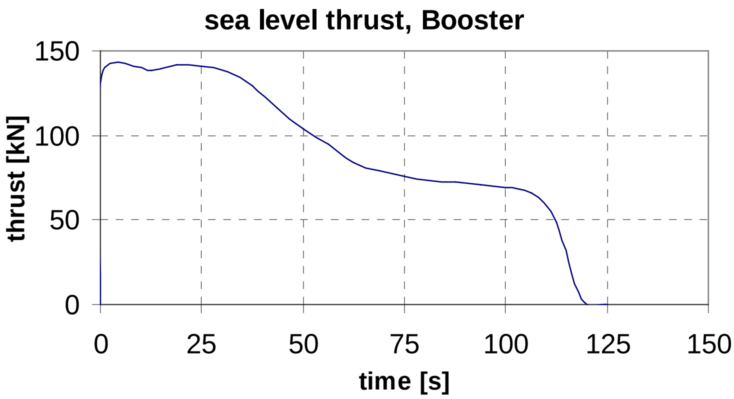


Figure 2: Thrust Profile of the booster

The take-off thrust is about 135kN for each booster. The long booster burn time is another consequence of the low thrust of the main engine which needs to be supported till a significant amount of propellant is consumed. The second stage consists of a solid rocket motor. Though such a SRM does not yet exist in Europe it is cheaper to develop than a liquid upper stage. However, note that it holds the disadvantage of having a comparably low specific Impulse of 295 s. The assumed thrust profile is shown below.

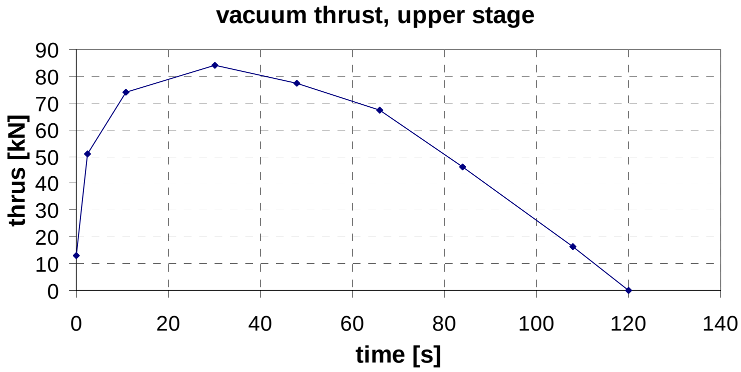


Figure 3: Thrust Profile of the solid upper Stage

3.2 Trajectory and Vehicle Performance

Figure 4 shows the trajectory of the launcher optimised for maximum payload. It is noticeable that the burn-out of the reusable first stage occurs at a high altitude of 101 km and the separation velocity is almost 4 km/s. This is mainly due to the inefficient solid rocket motor of the upper stage which is not capable of delivering a sufficiently high momentum without increasing heavily in mass and thus affecting strongly the first stage. This is the result of a stage optimisation that has been performed. On the other hand, trade offs showed that an appropriate cryogenic upper stage (70kN, like HM7B), would be able to deliver a higher total momentum. This in turn leads to lower separation velocity and altitude of the first stage.

Single events during ascent are listed in Table 4. The presented concept is able to deliver **250kg** into a sun-synchronous transfer orbit, as specified in Table 1. This corresponds to a payload fraction of 0.6%. This is a remarkably low value that has different reasons, which will be discussed in chapter 5.

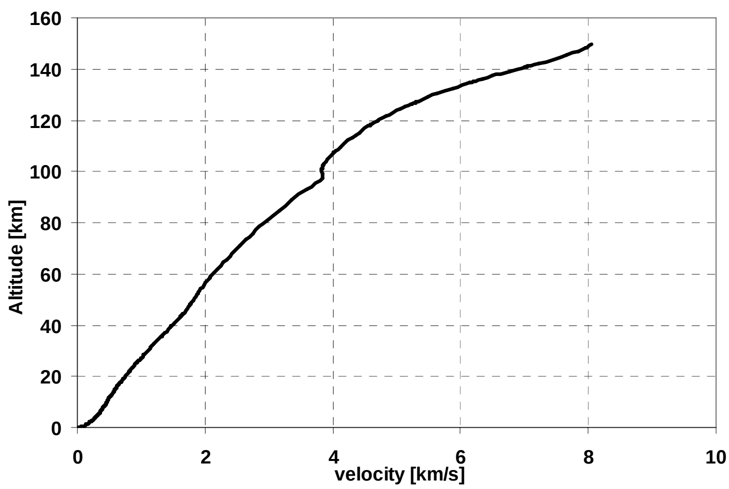


Figure 4: Ascent Trajectory of RFS

It shall be noted that the acceleration of the second stage reaches 6g at burn out. This is considered acceptable for compact satellites, however it shows a principle problem concerning upper stages with small payload; after separation from the first stage, the launcher has a velocity of 4 km/s and a flight path angle of 21°. Thus the upper stage still needs to deliver 4km/s.

	Altitude [km]	Velocity [km/s]	Mach [-]
Booster	51	1.8	5.7
1. Stage	101	3.8	14
2. Stage	150	8	-

Table 4: Separation Conditions

Hence, to reach transfer orbit thrust of the second stage must be sufficiently high. On the other hand the whole stage mass is low (small payload). This leads eventually to such high acceleration values that need special consideration to be kept in tolerable limits. More trajectory parameters are shown in Appendix B.

3.3 Fly-Back and Controllability

There are two modes of how to bring the first stage safely back to ground or launch area. The first one uses in-air capturing. In this case the vehicle continues its ballistic flight after burn-out and performs a moderate re-entry (Ma = 14). When the Mach number drops below one, the first stage will be captured up by an airplane and dragged back. The same technique was also discussed as a measure to bring back the liquid fly back booster (LFBB). A detailed discussion on this technique can be found in [4].

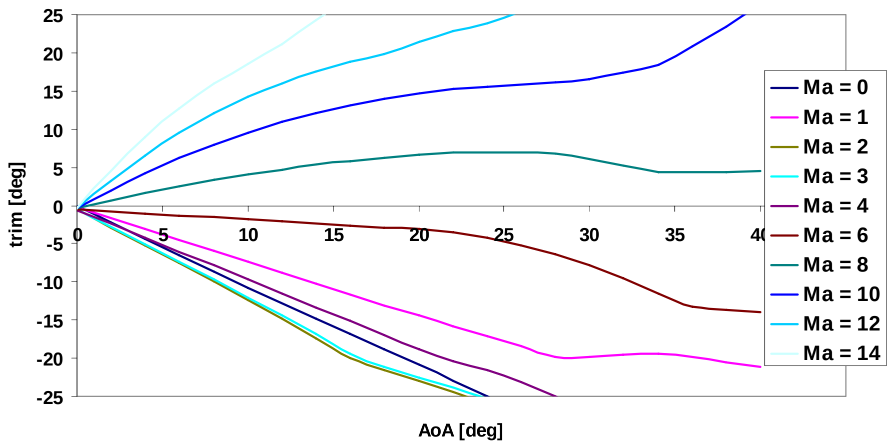


Figure 5: Flap deflection to trim RFS for different Mach numbers

To successfully accomplish this manoeuvre it must be possible to trim the first stage. This is indeed possible for the Mach number domain of interest as can be seen in Figure 5. The vehicle is stabilized by aileron deflection. For hypersonic flight a positive deflection of about 15° is sufficient to guarantee the desired angle of attack, whereas for low Mach numbers the necessary deflection is of the same absolute value but negative. This means that the centre of pressure is behind the centre of gravity (i.e. to the rear) for hypersonic flight, and in front of the CoG. (i.e. to the nose) for Mach numbers below 6.

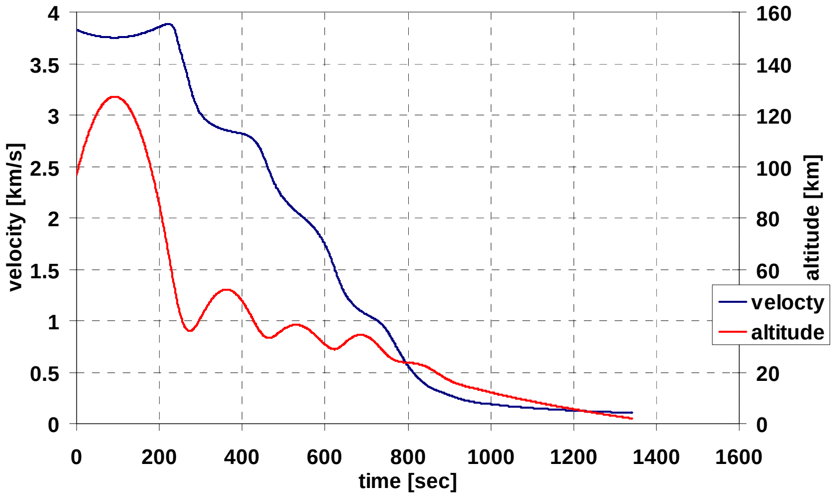


Figure 6: Re-entry of RFS

Figure 6 shows the re-entry of the first stage. It can be seen that after reaching an apogee of 127 km, the first stage performs a suborbital re-entry. Three moderate skipping flight states occur. No measures to avoid the skipping have been taking into account (no banking). The angle of attack history together with the normalised load in z direction is shown below.

The guidance is quite simple and ensures that the normal g-load factor stays below 3.5 and that the skipping does not become violent. The maximum dynamical Pressure does not exceed 40 kPa. This value can be further reduced by starting the re-entry with a greater angle of attack, for instance 45° instead of 35°. Appendix C contains more data of the re-entry.

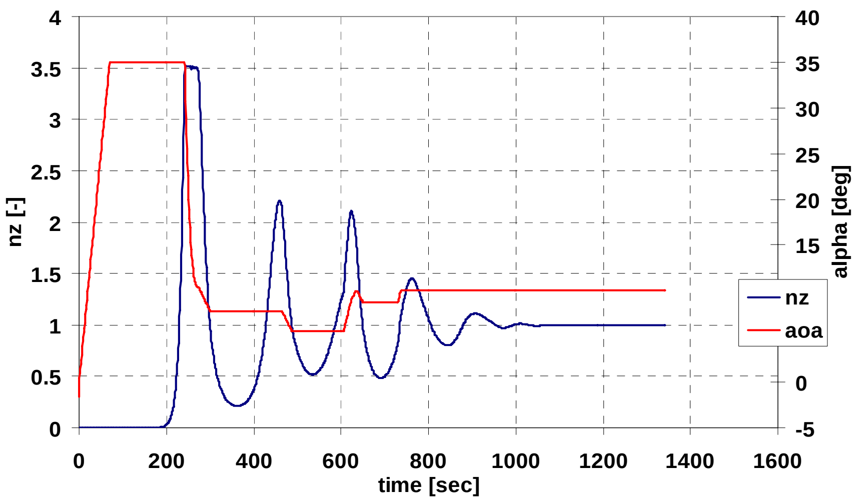


Figure 7: AoA of RFS and loads on RFS

The second possibility to fly back requires integrated air breathing engines. This variant has also been studied for LFBB [3]. Self-propelled flight is a tempting variant, since it promises more flexibility and less mission-complexity (in-air capturing requires the coordination of two airplanes). However it holds several problems: First, the separation velocity in this particular case is very high, resulting in a far away descending point (2130 km from launch pad). To fly the way back requires a significant amount of propellant. Second, the mass of the engine and its propellant add to the structural mass of the first stage, thus increasing the structural index considerably and lowering the performance of the whole launcher. It is therefore believed, that a self-propelled fly back is an option for a more massive first stage but not for a small one, like the one considered here. It should be noted that this argument is rather a qualitative one, a trade off was not performed. The next two figures show the descent of the first stage and the re-entry of the second stage, respectively.

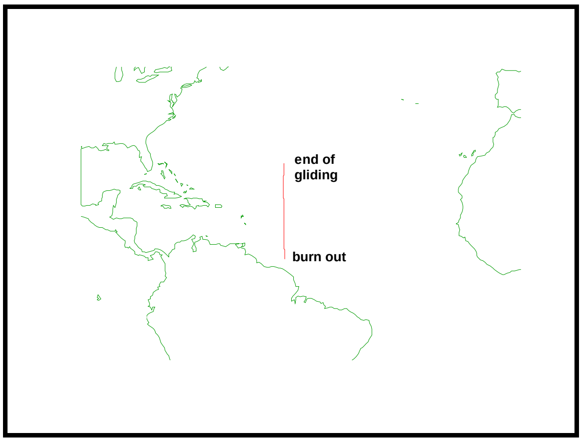


Figure 8: Descent of RFS

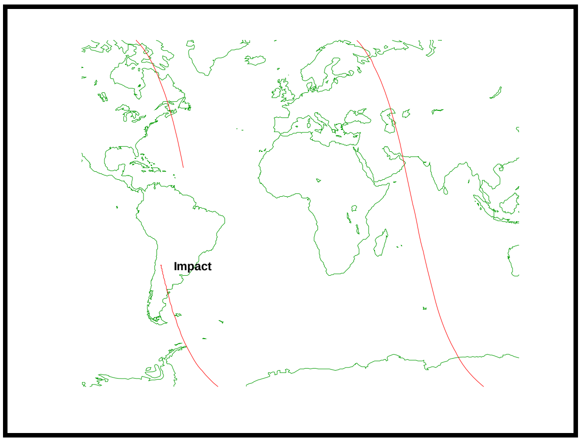


Figure 9: Re-entry and impact of the second stage

The point of impact of the second stage has not been optimized so far. For this is an inherently subtle issue in any SSO missions. An active de-orbiting of the second stage might be inevitable if changes of the ascent vector are not desired.

4 DEMONSTRATOR

The presented concept holds at least three new technological features:

a) Re-entry of the first stage, though suborbital
b) This stage is winged.
c) In-air capturing.
d) Reusability

To demonstrate these new technologies we suggest launching only the first stage without the strap-on booster. The thrust of the two Vinci engines is sufficient to accelerate a fully tanked first stage. A demonstration mission can also be flown with less fuel to reduce the fly back range. Figure 10 shows the ascent trajectory of RFS till burn out. The angle of attack is held constant and the pitching rate is 0,08 deg/s for 8 seconds, after 8 seconds of vertical ascent.
The burn-out velocity is slightly above 4km/s and hence similar to the separation condition of the first stage from the upper stage. This is because of the strap-on booster that compensate for the gravity losses induced by the upper stage.

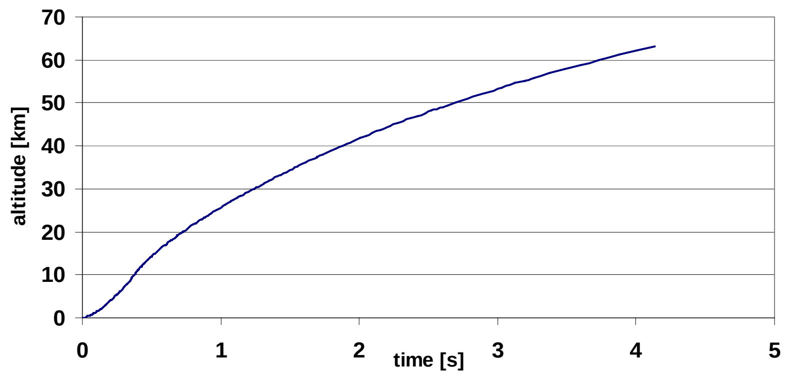


Figure 10: Ascent of RFS without booster and second stage

Although no special means (AoA) where taken into account to keep the loads in acceptable limits, they are not critical as can be seen in Figure 11.

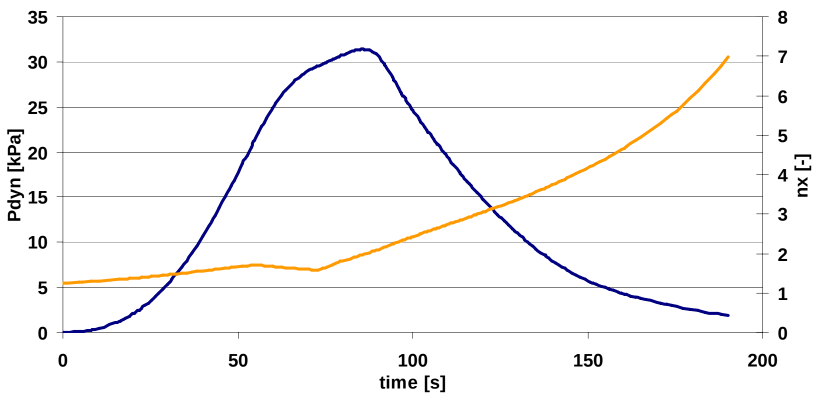


Figure 11: Loads on RFS during ascent

5 SUMMARY AND DISCUSSION

5.1 Summary

We presented a micro-launcher capable of delivering 250 kg payload in a sun-synchronous transfer orbit (50km x 800km). The launcher has a reusable (winged) first stage, a solid upper stage and three strap-on booster. The analysis encompasses the trajectory, the aerodynamics and a mass estimation based on empirical functions. Since the first stage is reusable, we also performed a re-entry calculation and a static stability analysis of the lateral movement. To bring the reusable first stage (RFS) back to the launch pad (or any other basis) we suggest to use in-air capturing.

5.2 Discussion

The whole study is part of ALDEBARAN which was initiated by CNES. Hence, the study guideline was dictated by the same project: The launcher shall be innovative, incorporate as far as possible European technology and be ambitiously cheap in operation.
The innovative technology here is the reusability of the first stage and the in-air capturing. As much as we believe that ALDEBARAN goes into the right direction, we suppose that the

requirements inhibit a sustainable launcher development in Europe. For Europe's space transportation is somehow in an impasse and lacking visions. An ordinary expandable launcher will not lead out of this crisis.

We believe that a micro launcher, if trendsetting, is a way out of this crisis. The launcher (RFS) presented in this paper exhibits key-technologies that are path-breaking for Europe's space transportation future.

For instance the reusability of the first stage: The reusability of a whole launcher or parts of it has been in discussion for a long time in Europe. To gain experience and make secure steps also in respect of the development costs we suggest entering this sophisticated field – winged re-entry – with a small launcher.

Note that usually propulsion system and vehicle design are strongly coupled in the whole launcher design process. This was only partly realised in the current study –new solid upper stage– because of the ALDEBARAN requirements. A three stage launcher with an appropriate liquid first stage engine, **500kN** class, and two solid upper stages can increase performance and flexibility of the launcher. This also has the advantage of decreasing the separation velocity (down to 3km/s) which in turn leads to a lower separation range and thus fly back range. The booster is then not essential to fulfil the reference mission, but an optional addition to increase payload or mission flexibility.

In that sense we plead to give up the highly ambitious low start and development costs of a new micro launcher. The launcher presented in this paper has the potential of becoming a sound starting point for further launcher development with similar technology, i.e. partly reusable. This is an essential issue for European space transportation, especially since Europe is lacking in visions for space transportation.

6 REFERENCES

1. Euroconsult for CNES/DLA – ***Prospective study of world market for small LEO satellites***.

2. ALDEBARAN: Gotor M. et al.: ***A "System" Demonstrator Project for new Generations of Space Transportation***. IAC-08-D2.6.5

3. Sippel, M.; Armin Herbertz: ***Propulsions Systems Definition for a Liquid Fly-back Booster***, 2nd EUCASS Brussels (No. 2007-153)

4. Sippel, M., Klevanski, J.: ***Progresses in Simulating the Advanced In-Air-Capturing Method***, 5th International Conference on Launcher Technology, Missions, Control and Avionics, S15.2, Madrid, November 2003

5. Dietlein, I et al.: ***Kick stage for Aldebaran,*** Design Workshop Study SART TN-015/2008

6. James, P et al.: ***TECHNOLOGICAL READINESS OF THE VINCI EXPANDER ENGINE,*** IAC-08-C4.1.9

7 APPENDIX A (MASS BREAK DOWN AND GEOMETRY)

Mass break down of the first stage.

1st Stage		Mass [kg]	Dry Mass Ratio [%]
Structure group:			
Mass Structure group:	w/o margins	2245.3	
Mass Structure group:	including 12.0 % margins	2514.8	54.2
Subsystem group:			
Mass Subsystem group:	w/o margins	492.1	
Mass Subsystem group:	including 12.0 % margins	551.2	11.8
Propulsion group:			
Mass Propulsion group:	w/o margins	913.5	
Mass Propulsion group:	including 12.0 % margins	1023.1	22.1
Thermal protection group:			
Mass Thermal protection group:	w/o margins	488.6	
Mass Thermal protection group:	including 12.0 % margins	547.3	11.8
Stage Mass empty:		**4139.7**	
Stage Mass empty incl.marg.:		**4636.5**	100
Stage Structural Index:		**0.28**	
Orbit/De-orbit propellant:		20	
Residual propellant:		386.1	
Reserve propellant:		198.9	
Stage Mass @ burn out:		**5241.5**	
RCS propell. /inert flow mass:		0	
Ascent propellant:		15600	
GLOW Stage Mass:		**20841.5**	

Mass break down of the second stage.

2nd Stage		Mass [kg]	Dry Mass Ratio [%]
Structure group:			
Mass Structure group:	w/o margins	422.9	
Mass Structure group:	including 10.0 % margins	465.19	
Stage Mass empty:		**422.9**	
Stage Mass empty incl. Margin :		**465.2**	100
Stage Structure Index:		**0.2**	
Stage Mass @ burn out (fairing separated):		**415.19**	
Payload Mass:		**250.1**	
Ascent propellant:		2330.58	
GLOW Stage Mass (w/o payload):		**2795.77**	

Mass break down of the booster.

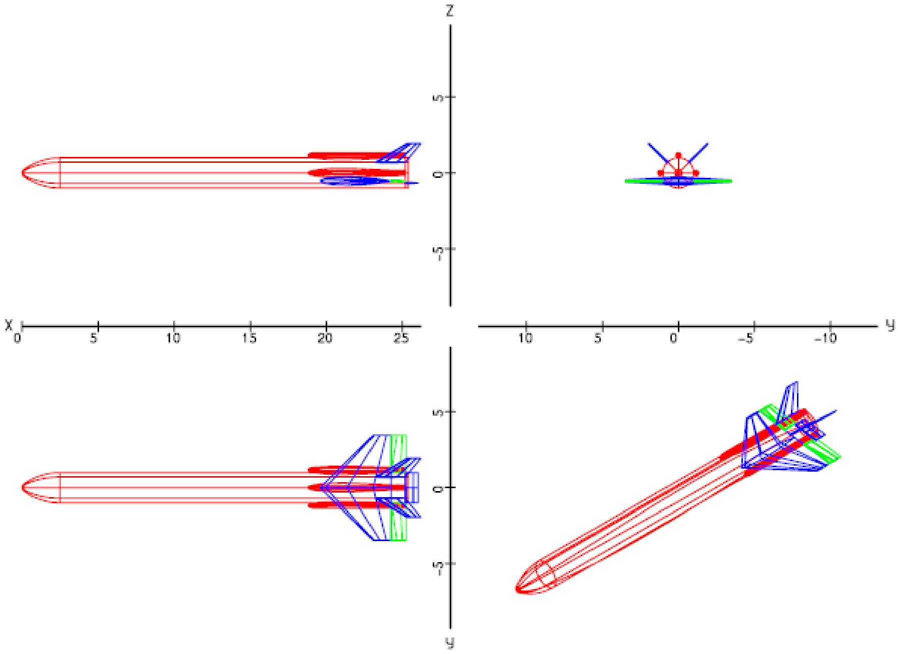
P2 Booster	Mass [kg]
Stage Mass empty: (stage coordinates)	642.1
Stage Mass empty incl.marg.: (global coordinates)	642.1
Stage Structural Index:	0.136
Stage Mass @ burn out:	688.8
Ascent propellant:	4674.758
GLOW Stage Mass:	5363.558

Mass break down of the whole launcher (overview).

Micro Launcher	Mass [kg]
Total Vehicle Mass empty:	6488.9
Vehicle Mass empty incl. margins:	7028.0
Total Lift-off Mass:	39727.9
Payload Mass	250.1
Gross Lift-Off Mass:	39978.1

If a detailed mass break-down is required please contact the author.

Sketch of Geometry



8 APPENDIX B

Ascent:

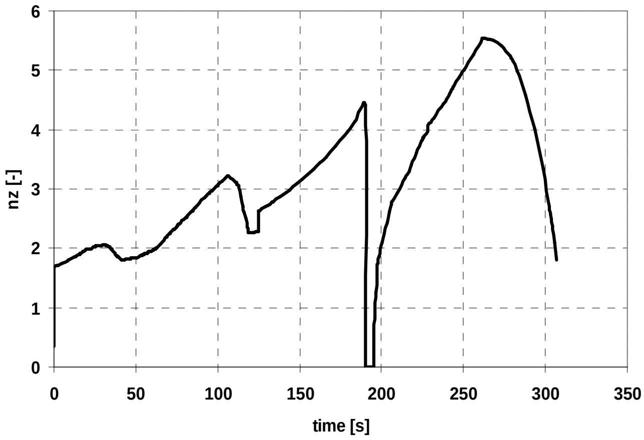


Figure 12: G load factor in flight direction

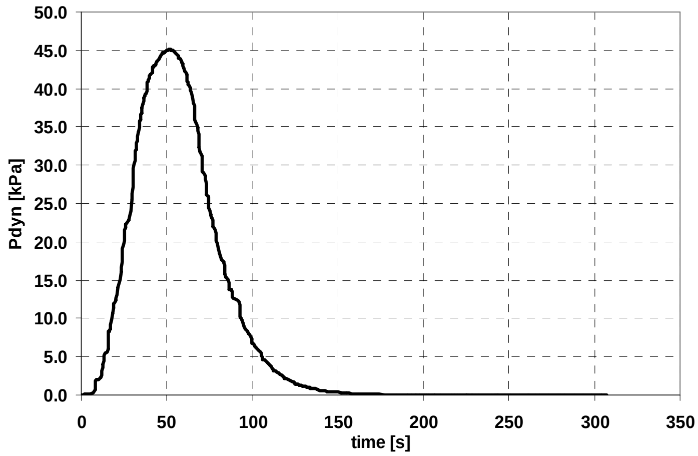


Figure 13: Dynamic Pressure over time during ascent

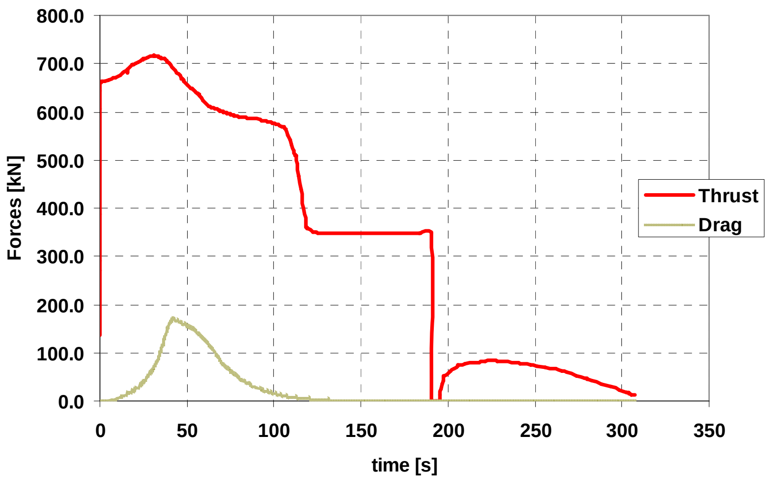


Figure 14: Drag and thrust over time

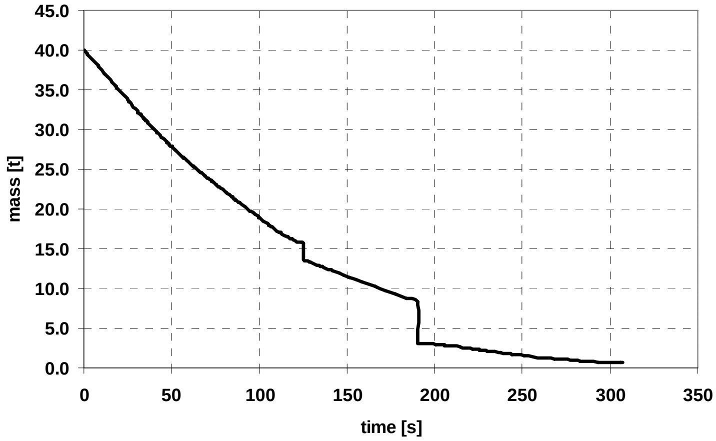


Figure 15: Mass variation of the Launcher over time

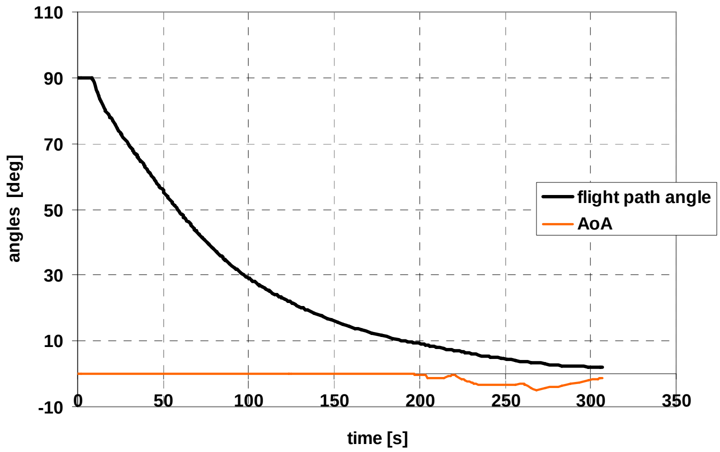


Figure 16: Guidance of the Launcher during ascent

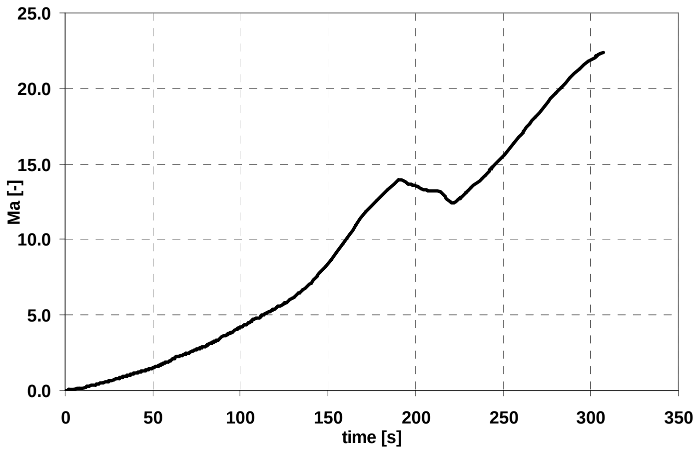


Figure 17: Mach number over time

9 APPENDIX C

Descent of the first stage:

Figure 18: Dynamic Pressure acting on RFS during re-entry

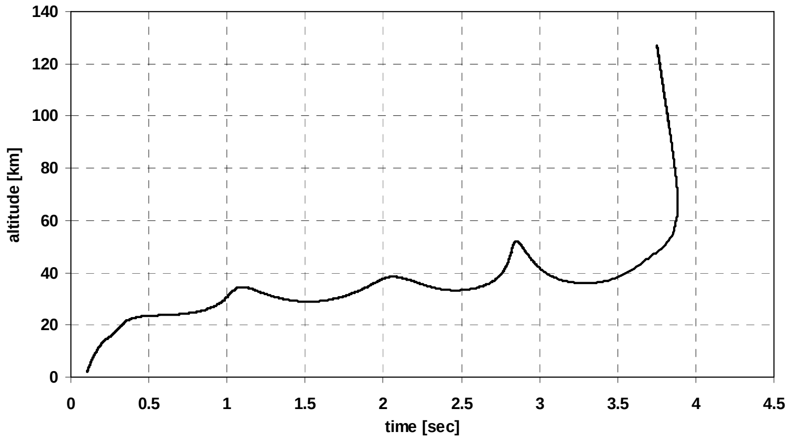


Figure 19: Descent trajectory of RFS

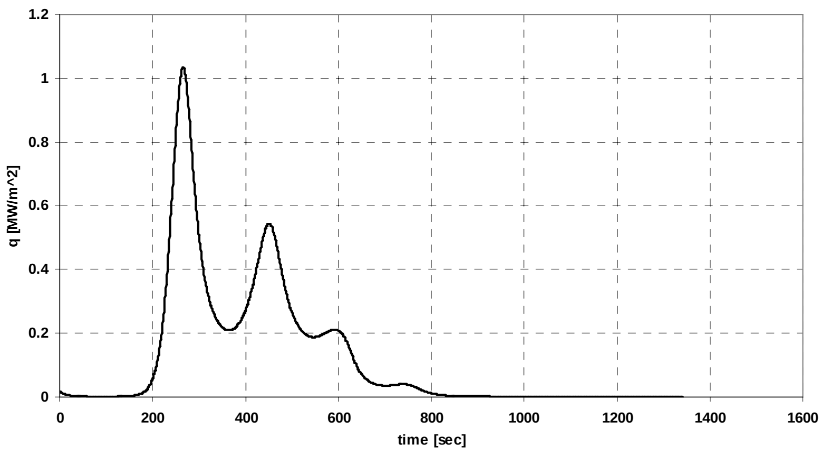


Figure 20: Heat load on RFS