SOLID PROPELLANT HYPersonic X-vehicle Options AS Rlv
Technology Stepping Stones

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Vital new technologies for reusable launch vehicles for which a strong necessity of in-flight testing exists are identified. As a basic constraint of the study, powered stages are assessed, which allow to explore the hypersonic flight regime without reaching orbital conditions.

Of the two different propulsion options liquid and solid, the latter has been selected in this investigation due to the lower risk and cost expected. After screening suitable motors, two promising vehicle configurations are selected for a more detailed study. This includes a preliminary aerodynamic shape design and vehicle mass estimation based on available component data and empirically based calculations. After checking on trimmability and flight stability requirements, flight performance simulations are carried out.

The investigation of different launch modes (sub- and supersonic air-launch or self propelled vertical ground launch, straight forward flight or a closed return loop) are the addressed major issues. Options for operational test sites of the flight demonstration vehicle are examined. This includes safety concerns, flight range estimation, and the necessity of a suitable telemetry acquisition infrastructure.

Finally, a preliminary total cost estimation of a reference liquid powered demonstrator’s development and experimentation plan is performed. The assessed amount is benchmarked to the US X-15 program with respect to current economic conditions. The presented data shows that flight vehicles with solid propulsion offer significant cost advantages compared to the reference system.

### 1. Introduction

The introduction of a European reusable space transportation system, which is envisioned in the next decade, requires considerable technological progress. Up to now, no practical experience exists in the development and operation of such types of vehicles in Europe. Therefore, experimental flight demonstration of critical technologies is essential before starting to design this new kind of launcher.

Within the system studies of the German future launcher technology research program ASTRA, two reusable first stage designs are under investigation. The first one is a high separation Mach-number, horizontal take-off, reusable vehicle with expendable upper stage (‘Hopper’) [1]. The other one, dedicated to more near term application with the existing expendable Ariane 5 core stage, is called a winged fly-back booster [2], [3]. Both options for future reusable stages require in-flight demonstration vehicles flying at least partially similar missions.

### 2. Objectives of Hypersonic In-flight Experimentation

In preparation for the FLPP study ESA evaluated different experimental vehicles [4]. SOCRATES is the project name for an intended powered hypersonic demonstrator of future reusable first stages. The objectives mentioned are, to gain the first practical experience with respect to re-utilization of a rocket propelled vehicle and to assess RLV technologies in a realistic operational environment. Operational and technological drivers for the automatic flight of such a vehicle with respect to recurrent cost and turn-around time should be further explored [4].
Considerable progress has been achieved in computational fluid dynamics (CFD) of hypersonic bodies in the past, and large data-bases of wind tunnel tests exist. However this background is by far not sufficient to permit the design of an operational launch system. Both exploration methods assume ideal flow conditions which will never exist in reality. High altitude atmospheric conditions (e.g. density shears) have significant impact on the flight control system as well as on vehicle heating. As observed by the Space Shuttle, the large density fluctuations can cause situations when the angle-of-attack corridor limits are reached. [5]

None of the upper atmosphere and hypersonic regime conditions have ever been explored by Europe with a winged vehicle. To avoid substantial risks in developing a reusable first stage, it seems highly advisable to gain practical knowledge in designing such vehicles.

The main requirements for a powered, non-orbital, hypersonic experimental vehicle are:

- a maximum speed between Mach 4 and Mach 8,
- a maximum altitude between 40 km and 90 km,
- a mass of about 12 tons and an empty mass of about 4 tons,
- a horizontal landing in order to use the automated landing techniques to be demonstrated by Phénix I.

Concepts for such a future vehicle are investigated in this paper. Two launch options are considered: Air-Launch from a carrier aircraft and vertical launch from ground.

3. PRELIMINARY DESIGN OF FLIGHT DEMONSTRATION VEHICLE

3.1. Selection of Propulsion System

In general, two propulsion options exist: Liquid rocket engines and solid rocket motors. Some liquid propelled demonstration vehicles have recently been proposed [4]. Since no suitable European engine is available, Russian LOX-RP motors (e.g. RD-0124) are drawn into consideration.

Although delivering a good performance, the associated technical risk of liquid propulsion should not be underestimated. In any case a dedicated propellant feed and tank pressurization system has to be developed and qualified. This is neither a simple adaptation of an existing stage nor will it be a prototype of the future RLV-stage, due to considerably different size and operation conditions. Furthermore, the flight envelope of a reusable, winged stage is notably different to that of a vertically lifted conventional stage. This is due to the fact that it will operate for a large portion of its mission at low flight path angles. Hence, severe impacts on the design of the tanks and the pressurization system are implicated. It is worth to note the development challenges the X-34 propellant feed system had to face. The LOX and RP tanks had to be subdivided into several compartments (at least three for each tank) to avoid propellant sloshing and hence uncontrollable flight states by center of gravity movement in case of premature engine cut-off [6]. These compartments had to be interconnected by sophisticated check valves. In addition, a unique tank pressurization system was required to take into account the more or less horizontal surface level of the liquids. Such an arrangement is not necessary for almost all proposed RLV-applications with vertical launch mode. In those cases a classical layout with a pressurization-gas diffuser at the top of the front tank dome can be implemented. In contrast, in quasi-horizontal launch mode the center of the front dome is submerged at engine ignition.

Although a liquid propelled hypersonic flight demonstrator should not be completely ruled out for subsequent steps, a considerably higher development cost is to be expected compared to solid motors (see chapter 5). Considering today’s tight budget constraints, DLR SART started therefore investigations to identify alternative, more cost efficient solutions using solid propulsion [7].

Many solid motors are in principal suitable for a hypersonic flight demonstrator. First basic trajectories simulations have been performed in an initial evaluation step. It was made use of a generic demonstrator vehicle, with an identical aerodynamic data set used for all simulations. The respective motor inert and propellant masses, an adaptation of the body dimensions to the motor size as well as the maximal sustainable demonstrator loads (normal ($n_z$) and axial ($n_x$) acceleration, dynamic pressure) were used to adapt the mass model and to obtain the GLOW. An eastbound launch from Kourou was assumed as initial conditions for all launches in this preliminary evaluation.

Six motors have been investigated in the initial phase of the study: P7, S43, Z9, S40-TM, MIHT-3 and Castor 1. Not all chosen motors are suitable for use in the lower atmosphere. This is the case for example for the P7 and Z9 motors, which are conceived for third stage propulsion, i.e. with a long nozzle. The over expansion of the exhaust flow prohibits the use of these motors with an unmodified nozzle at low altitudes. Therefore, it had been necessary to cut the nozzle of P7 and Z9, hence reducing expansion ratio and degrading vacuum Isp.

All listed motors were assessed for all launch scenarios.

3.1.1. Air-launch

The Z-9 motor is oversized in view of the target performance even for the subsonic scenario. The P-7 motor is also exceeding the upper performance limits, when the supersonic scenario is taken into account. Both motors are therefore ruled out as being suitable propulsion motors for an air-launched demonstrator of this class. The Castor 1 motor is the smallest motor evaluated. Its exclusion criterion is however the unacceptable high dynamic pressure [7].

The S-40 motor is eventually chosen against the S43 for the following two reasons:

- Smaller and lighter vehicle
- Lower maximal Mach number, keeping technical challenges and therefore risk at a lower level

3.1.2. Vertical launch

The selection of the motor for the detailed study of the vertical launch configuration is fairly straight forward. The use of the S-43, S-40 and MIHT-3 motors lead to an
excessive maximal dynamic pressure and have to be therefore excluded. Both, the P-7 and the Z-9 motors are in contrast promising candidates. As the P-7 is a project which has evolved in the mean time into the Z-9, it was obvious to choose this latter one for the in-depth study [7].

Technical data of the two selected solid motors for the preliminary flight demonstrator design are listed in Table 1.

![Table 1: Data of the selected solid rocket motors](attachment:table1)

<table>
<thead>
<tr>
<th>Z9</th>
<th>S40-TM(*)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Overall Length, m</td>
<td>3.600</td>
</tr>
<tr>
<td>Outer Diameter, m</td>
<td>1.904</td>
</tr>
<tr>
<td>Propellants Mass, kg</td>
<td>8 996.00</td>
</tr>
<tr>
<td>Inert Mass, kg</td>
<td>690.64</td>
</tr>
<tr>
<td>Gross Mass, kg</td>
<td>9 727.00</td>
</tr>
<tr>
<td>Structural Index (%)</td>
<td>7.52</td>
</tr>
<tr>
<td>(Empty mass / Gross mass)</td>
<td></td>
</tr>
<tr>
<td>Propellant Type</td>
<td>HTPB</td>
</tr>
<tr>
<td>Average Mass Flow, kg/s</td>
<td>76.90</td>
</tr>
<tr>
<td>Max. Thrust, kN</td>
<td>275.00</td>
</tr>
<tr>
<td>Average Thrust, kN</td>
<td>221.20</td>
</tr>
<tr>
<td>Isp vacuum, s</td>
<td>277.97</td>
</tr>
<tr>
<td>Isp s/l, s</td>
<td>237.36</td>
</tr>
<tr>
<td>Nominal Burn Time, s</td>
<td>117.00</td>
</tr>
<tr>
<td>Estimated Total Impulse, kNs</td>
<td>25 878.0</td>
</tr>
<tr>
<td>(vac)</td>
<td>(vac)</td>
</tr>
<tr>
<td>Average Chamber Pressure, bar</td>
<td>49.00</td>
</tr>
<tr>
<td>Max. Chamber Pressure, bar</td>
<td>68.00</td>
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<tr>
<td>Exit Diameter, mm</td>
<td>692</td>
</tr>
<tr>
<td>Nozzle Expansion Ratio ε</td>
<td>17.8</td>
</tr>
<tr>
<td>Case Material</td>
<td>Graphite</td>
</tr>
</tbody>
</table>

(*) Stage 3 on VEGA, Italy. Adapted to ground launch
(β) Stage 2 on VLS, Brazil

3.2. Air-Launched Vehicle S-40TM

For the air-launched configuration, two representative initial conditions have been chosen, corresponding to a subsonic and a supersonic separation from the carrier aircraft (see chapter 4.1). The same vehicle has been taken for the two options, because preliminary results were satisfying for both. Thus, their mass and aerodynamic properties are identical.

An iterative design approach was necessary to obtain a design with satisfying mass distribution and an adapted aerodynamic layout, providing controllability throughout the required flight envelope. Only parameters for the final design are described within this paper.

The size of the demonstrator vehicle follows primarily the requirements of the solid motor. Mass distribution, aerodynamic layout and payload size (test and flight equipment) are complementary requirements for the design of the vehicle. The demonstrator consists of the following main elements:

- main body with a long nose,
- wings,
- vertical stabilizer,
- wing flaps in order to control the demonstrator,
- body flap in order to protect the motor during the re-entry and for additional flight control.

Moreover several subsystems are necessary for the flight of the demonstrator:

- solid motor (with adapted nozzle for the launch altitude (see paragraph 3.1),
- reaction control system (RCS) in order to control the attitude of the demonstrator when the aerodynamic control surfaces are ineffective,
- landing gear,
- brake parachute,
- avionics,
- electrics,
- hydraulics,
- thermal protection system (TPS), present in particular on the nose and on the wing leading edges,
- environmental control system (ECS),
- health monitoring system (HMS),
- air-launch interface,
- and flight test equipment.

The choice of the ogive nose over the conical nose has been made to obtain a smaller variation of the center of pressure location, which is favorable to provide stability over a larger Mach range. Moreover the wing position and shape (double-delta) are very important to obtain satisfactory stability and controllability. The elevons size is a non-negligible factor for the controllability of the demonstrator. Figure 1 shows that the solid motor is easily integrated within the fuselage.

The overall vehicle length is dictated by center of gravity position with respect to the aerodynamic center of pressure. To achieve a stable configuration for major parts of the flight envelope, cog has to be shifted forward by increasing nose length.

The overall dimensions of the demonstrator with the S40-TM motor are:

- Total length: 9.80 m
- Wingspan: 6.50 m
- Fuselage diameter: 1.60 m

Preliminary vehicle mass is assessed using fast empirically based relationships or known component mass data. Structural and equipment mass without the solid rocket motor is slightly above 2.5 Mg (Table 2).

| Mass Structure: | 1026 kg |
| Mass Subsystem: | 1145 kg |
| Mass Propulsion (including motor casing): | 1080 kg |
| Mass Thermal Protection: | 228 kg |
| Stage Mass empty without motor casing | 2549 kg |
| Stage Mass inert | 3478 kg |
| Propellant loading | 4432 kg |
| GLOW Stage Mass: | 7930 kg |

Table 2: Mass data of the hypersonic flight demonstrator with S-40 TM solid rocket motor
Figure 1 Different CAD views of the air-launched demonstrator with S40-TM motor

Figure 2: Different CAD views of the ground-launched demonstrator with Z9 motor
Aerodynamic data sets are calculated by the DLR preliminary-aerodynamic-design software CAC2. The suitability of this tool for hypersonic vehicles has been demonstrated by comparison with wind tunnel and numerical data of a large selection of similar configurations [8].

The maximum L/D reaches more than 6 in subsonic flight and slightly above 2 in the hypersonic regime. (see Figure 3)

\[
\begin{array}{c|c|c|c|c|c}
M & L/D & 5 & 10 & 15 & 20 \\
0 & 2 & 2 & 2 & 2 & 2 \\
1 & 2 & 2 & 2 & 2 & 2 \\
2 & 2 & 2 & 2 & 2 & 2 \\
3 & 2 & 2 & 2 & 2 & 2 \\
4 & 2 & 2 & 2 & 2 & 2 \\
5 & 2 & 2 & 2 & 2 & 2 \\
6 & 2 & 2 & 2 & 2 & 2 \\
7 & 2 & 2 & 2 & 2 & 2 \\
\end{array}
\]

\[\text{Angle of attack (degree)}\]

Figure 3: Calculated L/D ratio as function of the Mach number and the angle of attack for the demonstrator with the S40-TM motor

The comparison of calculated center of pressure and center of gravity location shows that the vehicle is stable until around Mach 6. This is sufficient for both studied options (subsonic and supersonic launch). The flight demonstrator is fully controllable within the whole flight envelope. The angle of attack limit for trimmed flight is about 10 degrees for supersonic speeds (with margins) and quasi unconstrained in hypersonic flight assuming a maximal flap deflection of -20 degrees [7].

3.3. Vertical-Launched Vehicle Z9

For the vertical launch and horizontal landing configuration, two different flight scenarios have been assessed: a straight forward flight assumed to be performed from Woomera and the return flight assumed to be launched from Kourou. The same design has been taken for the two options because the preliminary results were satisfying for both. Thus their mass and aerodynamic properties are identical.

The requirement for the Z9 demonstrator to move the center of gravity position forward is even more demanding than for the S-40. An iterative design loop had to be executed to obtain a design with satisfying mass distribution and an aerodynamic layout, providing controllability throughout the required flight envelope. Only parameters for the final design are described within this paper.

The size of the demonstrator vehicle follows primarily the requirements of the solid motor. This demonstrator vehicle is therefore relatively short and thick due to the considerable diameter of the Z9 motor. Note the small size of the shortened nozzle and the large available volume in front of the motor casing (Figure 2).

Mass distribution, aerodynamic layout and payload size (test and flight equipment) are complementary requirements for the design of the vehicle. The main elements are similar to the air-launched configuration. The overall dimensions of the demonstrator with the Z9 motor are:

- Total length: 8.13 m
- Wingspan: 5.40 m
- Fuselage diameter: 2.32 m

Structural and equipment mass of the Z9 vehicle without the solid rocket motor reaches 2.35 Mg and is about 150 kg below that of the air-launched configurations, while GLOW is considerably above (Table 3).

| Mass Structure: | 926 kg |
| Mass Subsystem: | 1075 kg |
| Mass Propulsion (including motor casing): | 813 kg |
| Mass Thermal Protection: | 228 kg |
| Stage Mass empty without motor casing | 2351 kg |
| Stage Mass inert | 3042 kg |
| Propellant loading: | 8996 kg |
| GLOW Stage Mass: | 12058 kg |

Table 3: Mass data of the hypersonic flight demonstrator with Z9 solid rocket motor

The vehicle is stable until Mach 5.15. This is sufficient for both studied options (straight and return flight), L/D is considerably below that of the air-launched demonstrator due to the less slender configuration. Maximum subsonic lift-to-drag stays below 4 and the hypersonic values do not exceed 1.8. (Figure 4)

\[
\begin{array}{c|c|c|c|c|c}
M & L/D & 5 & 10 & 15 & 20 \\
0 & 2 & 2 & 2 & 2 & 2 \\
1 & 2 & 2 & 2 & 2 & 2 \\
2 & 2 & 2 & 2 & 2 & 2 \\
3 & 2 & 2 & 2 & 2 & 2 \\
4 & 2 & 2 & 2 & 2 & 2 \\
5 & 2 & 2 & 2 & 2 & 2 \\
6 & 2 & 2 & 2 & 2 & 2 \\
7 & 2 & 2 & 2 & 2 & 2 \\
\end{array}
\]

\[\text{Angle of attack (degree)}\]

Figure 4: Calculated L/D ratio as function of the Mach number and the angle of attack for the demonstrator with the Z9 motor

The flight demonstrator is fully controllable within the whole flight envelope. The angle of attack limit for trimmed flight is about 18 degrees for supersonic speeds (with margins) and quasi unconstrained in hypersonic flight assuming a maximal flap deflection of -20 degrees [7].

4. FLIGHT PERFORMANCE CALCULATION

The two different demonstrator vehicles’ performances have been determined by complete trajectory simulations from launch to the terminal landing phase. This allows to check the flight dynamic controllability...
along the operational range. All flight simulations are based on actual launch and landing sites, hence verifying the principal availability and suitability of the specific launch mode. Due to the high velocity and the experimental status of the vehicle, only flight above uninhabited areas will be acceptable. Nevertheless, these are purely technical working assumptions and neither scenario has been fully validated on basis of local legal, safety or economical concerns.

4.1. Air-Launched configuration

Since it is not necessary to fly a return loop in the air-launched scenario, the carrier aircraft will be free to separate the demonstrator at any suitable location with the right heading for a straight forward flight to the landing site. This allows for a complete flight over sea with a landing location close to the shore line. Operation of a powered flight demonstrator seems feasible within Europe exclusively under these circumstances.

An eastern heading flight along the coast line of northern Spain is proposed with a landing at the test facilities of CEL (Centre d’Essais des Landes) near Arcachon, France (see Figure 7). Already existing range control and telemetry installations might be reused in this scenario. The demonstrator vehicle with S40 TM motor is used.

4.1.1. Subsonic launch mode

The only operational carrier aircraft for air-launch are NASA Dryden’s B-52 and Orbital Science Corporation’s (OSC) Lockheed L1011-100 for launch of the Pegasus rocket. The latter airplane is the baseline for this study’s subsonic launch mode. After the separation from the carrier aircraft at 10 km altitude and Mach 0.78, it is necessary to fly initially at a large angle of attack in order to gain altitude rapidly. The upper bound of AOA is thereby given by the limit of the normal acceleration \(n_n\leq 3.5\). The lower bound of AOA is given by the upper limit of the dynamic pressure.

The maximum attained Mach number is 5.15 and the maximum altitude is around 55 km (Figure 5). The vehicle stays always in the stable flight regime. A neutral trimming of the demonstrator flight vehicle requires an elevon deflection with a corresponding moment coefficient \(C_m\) of zero. The maximal value of the flap deflection angle of \(-16.5^\circ\) is within the design limit of \(-20^\circ\).

The maximum angle of attack during initial descent flight is \(15^\circ\). The flight Mach number varies between 4 and 4.5 in this phase. The aerodynamic analyses provide a necessary flap deflection of about \(-10^\circ\) to \(-12^\circ\) for these conditions. The maximum values of normal and axial acceleration \(n_n\) (max. \(- 3.5\) g), \(n_x\) (max. \(- 5.8\) g) as well as dynamic pressure (max. \(\sim 50\) kPa) do not exceed the design limits.

The separation of the demonstrator from the aircraft has to occur at latitude of \(45.11^\circ\) and a longitude of \(-7.73^\circ\). The distance between the separation point and the landing point is equal to about 520 km.

4.2. Supersonic launch mode

A large supersonic carrier aircraft exists, though not operational as a space transportation system: the commercialized, demilitarized version of the Russian strategic bomber Tu-160 SK. It has already been proposed as a space launch system with the two-stage rocket Diana-Burlak of 30 Mg initial mass [9]. The prospected release conditions were Ma 1.7 and an altitude of 13500 m. Assuming the same separation conditions, the maximum attained Mach number of the hypersonic demonstrator is 6.15 and the maximum altitude is around 60 km (Figure 6). The vehicle is stable during most of the mission. Flight in unstable conditions is uncritical, in spite of a maximal achieved flight Mach number of more than 6. This condition is only reached at the end of the propelled ascent phase at an altitude of about 35 km. Aerodynamic forces are therefore small and can easily be countered by moments generated through thrust vector control of the nozzle (before MECO) and use of the RCS afterwards.

![Figure 5](image1.png)  
Figure 5: Evolution of altitude as a function of Mach number for the Air-Launched demonstrator with subsonic initial conditions

The required flap deflections for trimmed flight were determined as already described for the subsonic case.

![Figure 6](image2.png)  
Figure 6: Evolution of altitude as a function of Mach number for the Air-Launched demonstrator with supersonic initial conditions
in paragraph 4.1.1. The maximal value of -17° is within the design limit of -20°.

The angle of attack during initial descent flight is 15°. The flight Mach number is about 5.5 in this phase. The aerodynamic analyses show a necessary flap deflection of about -1° to -2° for these conditions. The maximum values of normal and axial acceleration \( n_z \) (max. \( \sim 3.5 \) g), \( n_x \) (max. \( \sim 6 \) g) as well as dynamic pressure (max. \( \sim 54 \) kPa) remain within the design limits.

4.3. Ground-Launched configuration

A vertical ground-launched scenario requires suitable lift-off as well as landing sites with an intermediate trajectory not endangering populated areas. Obviously, it is much more difficult to find an appropriate combination which fits the vehicle's performance than for air-launched scenarios. A straight forward flight and a closed return flight option are investigated.

No European launch site is selected, but already existing range control and telemetry installations should be reused. The demonstrator vehicle with Zefiro 9 motor is chosen.

4.3.1. Straight forward flight

Within this study, it is assumed that a straight forward flight can be performed from the Woomera launch pad, as the surrounding Australian desert is only extremely sparsely populated. A north-eastern heading is preferred to reach an existing airfield as shown in Figure 8 (geographical co-ordinates: -30.79 of latitude and 140.19° of longitude).

After a vertical climb out and the pitch maneuver, the angle of attack has to be kept near zero to avoid excessive aerodynamic loads. The AOA varies roughly between 0° and 1° in order to gain altitude rapidly. The maximum attained Mach number is 5.35 and the maximum altitude is about 50 km (Figure 9).

The vehicle is stable during most of the flight. The situation is also unproblematic in spite of a maximal Mach number above 5.15. This state is only reached at the end of the propelled ascent phase at an altitude between 35 km and 45 km. Aerodynamic forces are therefore small and can easily be countered by moments generated through thrust vector control of the nozzle (before MECO) and use of RCS.

The required flap deflections for trimmed flight reaches an ultimate value of -10° which is within the design limit of -20°. Sufficient margins exist to reach the prospected airfield with an angle of attack of 10°. The maximum values of normal and axial acceleration \( n_z \) (max. \( \sim 2 \) g₀), \( n_x \) (max. \( \sim 3 \) g₀) as well as dynamic pressure (max.
The distance between the launch pad of Woomera and the landing point is equal to about 475 km (Figure 8).

### 4.3.2. Return flight

As an alternative to the simple straight forward flight solution, a return flight trajectory has been assessed. Kourou is assumed to be a suitable launch place for a return flight scenario, as there is already extensive European launch operation hardware available.

A bank angle control is needed additionally to perform the 3D return flight. After the initial vertical climb and the pitch maneuver, it is necessary to fly initially at big angle of attack and bank angle in order to initiate a turn and to dissipate some energy. Otherwise, the demonstrator would fly too far away and a glide back to the launch pad would be out of reach.

The angle of attack is subsequently reduced to values close to 0° in order to limit the maximal altitude. This avoids reaching an excessive distance from the launch pad and assures furthermore that minimal efficiency of the aerodynamic control surfaces is retained throughout the flight. A small angle of attack is however indispensable to continue with a slightly curved flight.

After the propelled phase, the angle of attack has to be increased to reach the level of optimum L/D and thus to achieve greatest range. The bank angle has to be reduced in accordance to obtain a gentle curve of the trajectory for a return to the launch pad. At last, the bank angle is reduced to 0° when a straight heading for a return is achieved. The angle of attack stays obviously at the value corresponding to optimum L/D.

The maximum attained Mach number is only about 3.5 and the maximum altitude is about 20 km (Figure 10). Since the maximal speed for the fly-back case is considerably below Mach 5, the unstable regime is not reached during the whole flight (see paragraph 3.3).

The ultimate flap deflections for trimmed flight remain at -8°, well within the design limit of -20°. The maximum values of normal and axial acceleration n_z (max. ~ 3 g_0), n_x (max. ~ 3 g_0) as well as dynamic pressure (max. ~ 52 kPa) do not violate the pre-defined design limits.

The distance between the launch pad of Kourou and the furthest point of the trajectory is equal to about 62 km (cf. Figure 11).

![Figure 10: Evolution of the altitude as a function of Mach number for the Vertical Launched Demonstrator from Kourou](image)

![Figure 11: Return flight for Kourou launch in geographical coordinates](image)

The maximum achieved flight performances and loads are listed in Table 4. The demonstrator with S-40 motor and supersonic launching conditions experiences the highest loads, while the return flight vehicle the lowest. The stagnation point heat flow is a Fay-Riddell-like approximation. The maximum of 250 kW/m² should enable testing of different advanced TPS candidates.

The relatively low Mach-number of less than 3.5 of the Z9 return configuration is to be noted.

<table>
<thead>
<tr>
<th>Vehicle Type</th>
<th>Launch / Flight Mode</th>
<th>S40</th>
<th>Z9</th>
</tr>
</thead>
<tbody>
<tr>
<td>Altitude, km</td>
<td>Subsonic</td>
<td>54.91</td>
<td>60.14</td>
</tr>
<tr>
<td>Velocity, km/s</td>
<td>Supersonic</td>
<td>1.55</td>
<td>1.91</td>
</tr>
<tr>
<td>Mach Number</td>
<td>Straight Forward</td>
<td>5.13</td>
<td>6.15</td>
</tr>
<tr>
<td>n_z</td>
<td>Return Flight</td>
<td>5.67</td>
<td>6.06</td>
</tr>
<tr>
<td>n_x</td>
<td>3.51</td>
<td>3.51</td>
<td>1.88</td>
</tr>
<tr>
<td>Dynamic Pressure, kPa</td>
<td>49.72</td>
<td>54.31</td>
<td>42.43</td>
</tr>
<tr>
<td>Stagnation Point Heat Rate, kW/m²</td>
<td>188.38</td>
<td>248.92</td>
<td>125.56</td>
</tr>
</tbody>
</table>

Table 4: Maximum achieved flight performance of the investigated hypersonic flight demonstrators
5. PRELIMINARY COST EVALUATION

Presently, no winged reusable flight demonstration vehicle entering the hypersonic regime has ever been developed in Europe. As no indigenous management experience exists, it is worth to have a first look at the US X-15 experimental aircraft. A retrospective analysis of this successful program has been performed in reference [10].

The X-15’s overall flight envelope, inclusive of the Mach number to be achieved and of the maximum altitude, is similar to the vehicles investigated in this paper. In the late 50s and the 60s the development and flight experimentation of the pioneering research aircraft had been a complex program. A total of 199 manned flights were performed. The program stretched over 13 years of development, design and flight operations. Total costs of the X-15 are estimated to have reached $1.4 billion in year 2000 economic conditions [10].

Although such figures are not easily transferred to proposed European hypersonic demonstrators, it can be seen that a reusable liquid propelled vehicle in the hypersonic speed range is not a cheap endeavor. Although still affordable, it is important not to underestimate the inherent risk and cost to eventually conclude such a research undertaking by successful flight demonstration.

In a preliminary assessment, major hypersonic demonstrator stage cost items are collected. In a first step this is done for a proposed reusable, liquid powered experimental reference vehicle of comparable Mach-range and air-launch mode. The data is based on experience from already performed projects (e.g. capsule) or comparable vehicles under development. Special development challenges of the propellant feed and pressurization system [8] are taken into account in a rough estimation by implementation of complexity factors. In a DLR-SART estimation of the total program expenditure, inclusive of the operation of 10 air-launched experimental flight missions, the SOCRATES type reference vehicle, not further described here, reaches a total cost of slightly more than 400 M€. Although still preliminary, the obtained price seems to be credible with regard to the knowledge from X-15.

In a second step, the data of the reference stage assessment is adapted to the four different solid rocket propelled demonstrators described in this paper. Obsolescent components like tanks, propellant feed and pressurization system have been abolished, but additional items like nozzle adaptation of the 29 ground launch vehicles have been added. Some costs are scaled by loads, others by component dry mass. Figure 12 shows that the solid demonstrators’ operations costs exceed those of the liquid reference vehicle. This is due to the expenses of a new solid motor for each flight. These costs are higher than the price of the liquid propellants of the SOCRATES reference configuration. Refurbishment cost of the reusable liquid propellant engine is assumed to be low, although this is still to be demonstrated. Ground-launched vehicles lead to much higher operation costs, since they rely on a larger, hence more expensive motor. In the case of straight forward flight ground transportation effort back to the launch site must also be included.

A different perspective emerges when the total amount is compared, which will be spent in hypersonic flight demonstration (Figure 13). All solid motor vehicles are considerably cheaper than the liquid reference stage, reflecting the lower effort in development and manufacturing. The relative cost range is between 52% (subsonic air-launch) to 62% (ground-launch with return flight) of the reference stage. The total saving of solid powered hypersonic flight demonstrators therefore might approach 200 M€.

6. CONCLUSIONS

Hypersonic flight demonstration is identified as an indispensable next step for Europe to develop its very own reusable launch vehicle. In general, two propulsion options exist: liquid rocket engines and solid rocket motors. Although delivering a good performance, the associated technical risk and cost of liquid propulsion should not be underestimated. A dedicated propellant feed and tank pressurization system has to be developed which can not be an adaptation of an existing stage. Therefore, this study focuses on the
Two suitable motors have been selected: The Brazilian S-40 TM for air-launched vehicles and the Italian Zefiro 9 (VEGA third stage) for ground-launched demonstrators.

Two vehicles have been iteratively sized in a preliminary design process taking into account aerodynamic trimmability and flight performance estimation by trajectory simulation. The air-launched demonstrator might be operated off the European coast and can reach a Mach number above 6 in straight forward flight prior to landing on European soil. A ground launch seems to be only feasible outside of Europe. In straight forward flight operation, proposed for Australia, hypersonic conditions can be reached. Simulations of the same vehicle in a return flight mission from Kourou show that performance is severely restricted by the requirement to fly-back to the launch site. In conclusion, acceptable flight conditions for hypersonic research are achievable in all straight forward flight scenarios, but not in the mission requiring return flight.

A preliminary cost estimation reveals that total expenses of all solid propulsion demonstrators are quite similar. Nevertheless, the ground-launched stages become more expensive due to their larger motors. Compared with the liquid engine reference configuration, a significant cost reduction between 38 and 48 % seems feasible.

The RLV-related development approach recommended at the end of this paper is to select an air-launched flight demonstrator with solid motor as baseline. The flight envelope of such a vehicle can be extended step by step up to Mach 6. Should systems studies, to be run in parallel, prefer an operational reusable launcher with significantly higher separation Mach number, the demonstrator’s flight performance could be further enhanced by integrating a liquid propulsion system, as sufficient volume is available inside the fuselage.

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8. REFERENCES


Further updated information concerning the SART space transportation concepts is available at: http://www.dlr.de/SART