

DESIGNING FLIGHT EXPERIMENTS FOR HYPERSONIC FLOW PHYSICS

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Abstract

A major problem of concern for the success of physical-modelling resides in the availability of experimental data for model validation, particularly in the hot hypersonic regime. In the past, validation data have been achieved as secondary product of expensive space-transportation programs. Since in the last ten years there has been almost no successful program due to lack of investment, no new experimental data are available. According, a new trend is emerging for low cost technology validation in flight based on the use of old military rockets. Here it is described the possibilities of sounding rockets to provide a platform for flight experiments in hypersonic conditions as cost-efficient supplement of tests in ground based facilities. The lecture regards the overall philosophy of projects flown on sounding rockets, but mainly the current efforts of DLR on the SHarp Edge Flight EXperiment SHEFEX. The approaches adopted on each of the involved disciplines, i.e. mission system and launcher; aerodynamics, aerothermodynamics and in-flight measurement techniques; structure and thermal protection systems are discussed. The paper is aimed to show that even flying hypersonic is not a routine technology, the strategy proposed is not only a highly efficient way to acquire important knowledge in the physics of hypersonic flows but also an excellent test bed for proving new technological concepts.

1. Introduction

Early in the fifties few small hypersonic wind tunnels had been used almost entirely for fluid mechanics studies but they were unable to simulate neither the high temperatures nor the high Reynolds number of flight. Because of strongly interacting flow fields, viscous interactions with strong shocks, and possible real gas effects, it was generally feared that testing in these limited wind tunnels would not produce valid results. It was expected that the first hypersonic vehicles would reveal large discrepancies between flight and ground test data, as was the case for the X-15, US-Orbiter, Russian-Buran and Japan-HST. Fifty years later and with some progress within the space community in designing, building and flying hypersonic gliding vehicles (**Fig. 1**), the fatal flight of the Columbia in 2003 has shown the severity of the hypersonic environment. Indeed, hypersonic systems are complex, difficult to design and expensive to build due to a lack of physical understanding on the involved flow regimes and a lack of data for design. Although in each decade there is considerable energy spend designing new concepts and developing the technology

base required to support them, no aircraft capable of flying at Mach 10 or higher has been built and the current space transportation systems do not enable routine low-cost operations [1].



Figure 1: ARD and FOTON capsules short after landing (ESA & DLR courtesy).

In order to improve the reliability of accessing space, e.g. problems related to vehicle servicing and refurbishing, must be hardly simplified and the time required for a design cycle has to be drastically reduced. The required technological developments are directly associated to the technological progress of the three disciplines: Aero(thermo)dynamics, Propulsion, Structures and Materials. A close interaction of these three disciplines, as well as the optimal use of all technical potentialities is necessary for a drastic reduction of design times, improved vehicle performance and operations. Several expendable and partially or fully reusable concepts discussed today or already planned require essential improvements over the current vehicles in order to ensure economic viability and to fulfil mission and safety constraints. While computational or “virtual” vehicle design and qualification would be the strategic tool which one day will enable to reduce dramatically the design and development time required for new vehicles, a computational design is based on mathematical models which require verification and validation for establishing credibility. To accomplish its mission, physical modelling strongly requires good data, acquired in ground facilities and in flight, to be used for code validation and when necessary code calibration too. Ground based facilities, the major source of flow data, are very important because they allow a better understanding of the flow physics but in flight-measurement constitutes the only way to obtain data for prediction tools validation and calibration under real conditions and therefore, they are irreplaceable for physical modelling [2-3]. However, in the last 25 years the expectation of progress in hypersonic physics have been strong coupled with new operational systems developments instead of use less complex systems like sounding rockets as test bed for new technological concepts or for simple gathering of flight data which are representative of the real hypersonic environments. Many X-programs, such as Hermes, the X-30, S nger, X-33, X-34 and the X-38 have failed, in some cases short before the flight test that would produce the data, so necessary to advance our understanding of the hypersonic environment and how to design vehicles that would survive in that environment. Those failures occurred partially due to economical reasons and partially due to the technological challenges associated with the projects, being the first cause also close related to the second one.

Sounding rockets may be the way of access to a certain type of “sky based facility”, in analogy to the ground based facilities, concurrent in price also with today’s so-called “industrial wind tunnels”, but in any case, extremely less expensive and less risky than the so-called “X-vehicles”. However, the majority of sounding rockets have been used for experiments which require exoatmospheric trajectories, as for example astronomy and micro-gravity experiments. Micro-gravity conditions are necessary for some technology, material science and biology experiments. Also, sounding rockets have been used for measurements in high atmosphere (mesosphere). As sky-based facility, the experimental phase takes place instead in the exoatmospheric, in the atmospheric down-leg of the trajectory and therefore the layout of the vehicle may be strongly affected by the objectives imposed for the mission-experiment even for similar mission-sequences. Indeed, as mentioned before, hypersonic re-entry is not a routine application even for sounding rockets.

An example of hypersonic experiments on sounding rockets is the NASA Sub-Orbital Aerodynamic Reentry EXperiment SOAREX program [4] to test future flight vehicles. SOAREX-1 was launched in 1998 on a Mk70 Terrier / BlackBrant IX rocket with a payload of 11 separate hypersonic experiments. It reached an apogee of 290 km and ejected each experiment at 4 second intervals. SOAREX-2 was launched in 2002 on a newly developed hybrid rocket to achieve Mach 5 during the re-entry. The payload included a wave rider flying wedge, a hypersonic parachute and a Slotted Compression Ramp Probe SCRAM. Another example of such strategy is the HyShot experiment of the University of Queensland [5]. The goal was to correlate data of a super combustion experiment in shock tunnel and in flight conditions [6]. After a failure of the first flight in 2001, super combustion was achieved in the second flight in 2002. The two-staged rocket consisted of a Terrier as booster and an Orion motor as 2nd stage. For the re-entry the burnt out Orion motor remained attached to the scramjet module to ensure flight stability. At DLR R&D the use of sounding rockets is being adopted defining the SHEFEX project as the pathfinder experiment, a hypersonic experiment that will be flown on a sounding rocket in similar way to the HyShot concept.

2. Objectives of the SHEFEX flight experiment

Following the classical 3-steps strategy, i.e. numeric – ground based facilities – flight experiment (**Fig. 2**, [7]), the objective of the experiment is the exploration of a faceted TPS concept and the assessment of the potential of sharp edge configurations. The motivation is neither to perform a re-entry experiment nor to fly at the thermal boundary of today’s high temperature materials but to provide with flight evidences that the temperature peaks at the edges of the ceramic-composite panels are lower than those predicted on the base of a radiation equilibrium hypothesis. Finally and most important, the SHEFEX project is a pathfinder experiment to set up a certain kind of future sky based facility, as irreplaceable way to understand the physics of the hypersonic flow, complementing numerical results and ground based facilities tests. It is a cost-efficient flight experiment by means of which DLR is not only opens the door for hypersonic sharp configuration and for a new way of thinking on TPS designs but also demonstrates the potential of vehicle design based on a multidisciplinary approach. Indeed, improvements in thermal protection system materials have continuously expanded the capabilities and operating envelopes of hypersonic vehicle. However, the implementation of these improvements has been difficult because the

aerospace industry uses a conventional aircraft methodology to aerospace vehicle design that focuses first on mission definition, aerodynamics, structure and last on the thermal protection system. A potential of a multidisciplinary design approach is that designers are not forced to follow such classical scheme but they are free to initiate the design from the most critical or key area and then moves to the other involved disciplines. This is in fact the philosophy adopted in case of SHEFEX, a hypersonic sharp body concept resulted from a thermal protection system perspective.

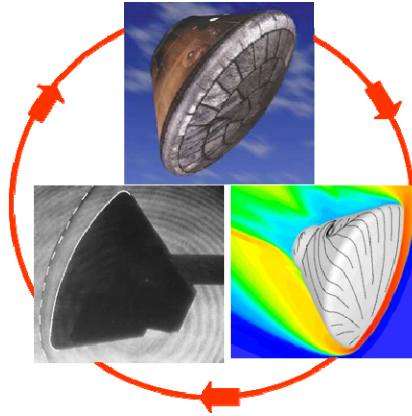


Figure 2: Sources of aerothermodynamic data: ground based facilities (left bottom); flight experiment (top middle); CFD (bottom right).

From an airframe point of view, fabrication, inspection / repair and payload displacement costs are the three major elements determining the TPS cost per unit area of each system [8]. Fabrication represents the cost for purchase and installation of the TPS. Inspection / repair, or maintenance / servicing, represents the cost for re-flight certification. Payload displacement represents the payload mass displaced by the TPS. Increasing the surface area of an aerospace vehicle increases all these three elements of TPS life cycle cost, but from those, payload displacement cost dominated overall since the other two are almost lower by an order of magnitude. An increase in TPS mass displaces payload for the life cycle of the vehicle, and significantly increases the payload delivery cost to and from orbit, since TPS total mass may be equivalent in some cases to the vehicle payload mass. Furthermore, the mass and cost of a thermal protection system (TPS) heavily impact the economic performance of the vehicle. Indeed, vehicle and TPS designs are tightly coupled since a 20 % increase in vehicle dry mass causes a 35 % increase in TPS mass.

While the technology for a reusable TPS is today available, the complexity of current designs consisting of many individually shaped parts makes this very expensive. Particularly, in Germany, a high level of technology for TPS based on ceramic composite has been demonstrated during the MIRKA project [9] and the TETRA / X-38 program [10]. But also some very important aspects for an operational system like maintainability, health monitoring, inspection, repair and economics have been neglected or not investigated in detail. From a servicing point of view, flat panels, having all the same basic shape are the most economical solution since the costs for manufacturing tools can be minimized and inspection devices can be designed more simply. A crude analysis about the economic potential of a faceted TPS system has

been obtained considering a hypothetical situation with a modern aerospace vehicle. The exercise consisted to look first into the aerodynamic impact which would have such kind of vehicle if its front section is covered by planar panels instead of a contoured one (**Fig. 3**). The numerical computation showed no major impact on the aerodynamic behaviour of the vehicle. Panels of 0.8 m maximum length have been considered to account for thermal expansion / contraction problems. To cover the front section of a vehicle like Phoenix almost 50 exact contoured panels are necessary, requiring also 50 manufacturing tools, or 70 flat panels requiring only one manufacturing tool. It turns out that it is possible a saving in manufacturing cost of about 70% accompanied with a strong reduction in development time due to the simplicity of the design. In addition, for the operation of four vehicles, the saving due to TPS inspection, simplicity in replacing and sealing amount another 40 to 50% at the system level. Indeed, from an economical point of view, the shape of the TPS-area of a reusable launch vehicle should be faceted. That change in the TPS concept streams potential enhancements throughout the entire system due to the highly integrated nature of the aerospace vehicles. Particularly, a faceted vehicle will result in an aerodynamic, sharp-edged one.

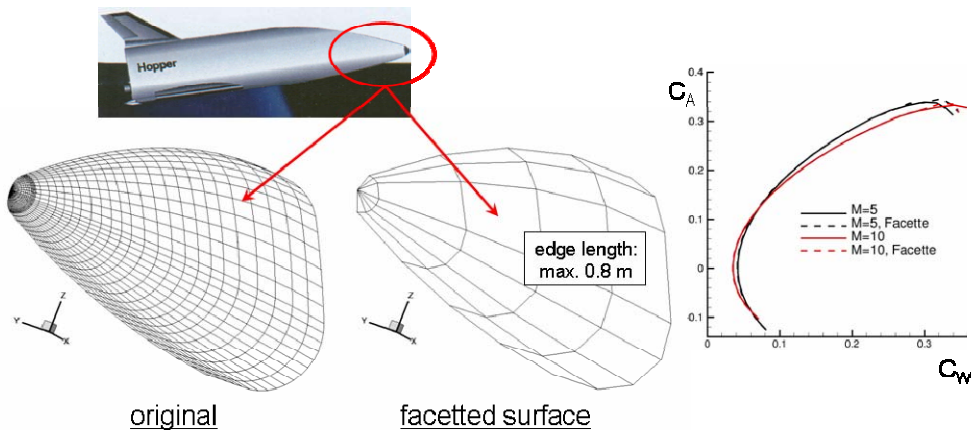


Figure 3: Conceptual evaluation of the faceted concept.

From an aerodynamic point of view, the design of hypersonic vehicles is affected by finding a compromise between a vehicle being sharp enough to obtain acceptable aerodynamic and propulsion efficiency and blunt enough to reduce aerodynamic heating. In general sharp configuration vehicles have better aerodynamic performances than hypersonic blunt-nosed ones since sharp body concepts have minimal drag, require lower thrust during ascent and achieve higher cross-range during re-entry. Cross-range improvement affects launch abort, the re-entry windows and the number of locations required for landing sites. Finally, sharp leading edges minimise the number of electrons that jam radio transmissions and that cause communications blackouts during the re-entry of blunt-body vehicles. Small radius leading edges and nose-tips were used in early hypervelocity vehicle concepts to minimize wave drag until further analysis demonstrated that extreme aerothermodynamic heating blunted the available thermal protection system materials. Supersonic aircrafts of the 1940s routinely featured needle noses and tapered wing tips that reduced drag until the beginning of the space age in the 1950s, where it was recognized that materials scientists could not then provide thermal protection systems enabling the design of Earth entry vehicles

with sharp leading edges [11]. The use of blunt-nosed shapes tends to alleviate the aerodynamic heating problem because the heat flux for blunt bodies scales inversely with the square root of the nose radius and in addition, the reduction in heating rate for a blunt body is accompanied by an increase in heat capacity, due to the increased volume. In the past, this concept led the design of many vehicles like Mercury, Gemini, Apollo and also the US-Orbiter. But shape optimization studies in the 1960s and 70s suggested that improved volumetric packaging efficiency and improved aerodynamics would be realized with vehicles using sharp leading edges. Hence, designing a hypersonic vehicle involves a trade off between making the vehicle sharp enough to obtain acceptable aerodynamic and propulsion efficiency and blunt enough to reduce the aerodynamic heating in stagnation regions.

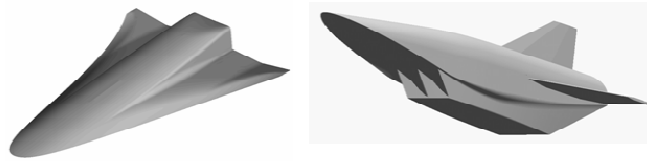


Figure 4: DLR F8 waverider concept (left) and ONERA-DLR JAPHAR concept (right).

During the 1990s, the development of ceramic composite and ultra high temperature materials for thermal protection system applications led to a renewed momentum on aerodynamic sharp edge configurations as shown by the lifting body concept HL-20 and the SHARP project, both of NASA [12-13], the waverider concept F-8 of DLR [14], or the DLR-ONERA JAPHAR project [15]. In particular, DLR is being putting considerable effort to understand the hypersonic physics of sharp-edged configurations (**Fig. 4**). It has been envisaged the potential of ceramic materials to build up sharp configurations thanks the property of thermal conduction along the fibres. First computed results by means of fluid-thermal-structure coupling obtained in the frame of the JAPHAR project [16] showed that the temperature rise at sharp edges would not be as dramatic as expected from a pure radiation adiabatic computation, a finding also reported by NAL [17]. In the frame of the TETRA program, experimental results obtained in arc-jet facilities with ceramic plates modelling TPS in re-entry environments, support those observations [18]. Recently experimental / numerical results obtained in the frame of the IMENS project, a project embedded in the German national program ASTRA, have confirmed those preliminary observations and addressed also the impact of sneak flows [19]. However, the extreme heat impacting the sharp edges can jeopardize the material integrity. Furthermore, at the interfaces of the panels, chamfers are created which interact with the surrounding flow. All these effects have to be investigated in detail with the help of a flight experiment before the overall aerodynamic performance of the sharp concept approach is assessed.

3. Defining the SHEFEX experiment

SHEFEX is a DLR self-contained experiment. It takes advantage of the different and well posed capabilities of the DLR-R&D institutes. The Institute of Aerodynamics and Flow Technology is responsible for the overall conception of the experiment and specific for the aerodynamics of the complete system, the thermal analysis of the faceted body and the instrumentation of the experiment. The Institute of Structures and Design is responsible for the faceted concept and specific for the design, manufacturing

and qualification of the faceted body structure (cold and hot) and also for the integration of the passenger experiments. The Mobile Rocket Base is responsible for the mission definition and execution and specific for the complete design and integration the launcher, data tracking and landing and recovering operations among others. Also, five passenger experiments on TPS qualifications are foreseen in SHEFEX. Four of them are under the industrial responsibility of EADS-ST [20], PLANSEE and MAN, while the last one is due to the Institute of Research on Materials of DLR.

In the following sections the approaches adopted on each of the involved disciplines, i.e. mission system and launcher; aerodynamics, aerothermodynamics and in-flight measurement techniques; structure and thermal protection systems are briefly presented and discussed. For detailed information, the interested reader is referred to the cited literature.

3.1. Launcher and Mission [21]

The flight experiment, schedule for 2005, is a suborbital flight mission to be started from Norway Andoya Rocket Range. The launcher is a two-stage solid propellant sounding rocket system conceived primarily for ballistic microgravity experiments. The launch vehicle consists of a Brazilian S30 motor as first stage and an improved Orion motor as second stage (**Fig. 5**). This motor combination has been launched successfully two times and reached with a hammerhead configuration an altitude of 320 km. Between the SHEFEX experiment and the second stage are two cylindrical modules which house the recovery system, the main electronics, the data acquisition devices, the power supply and the cold gas RCS. Furthermore, the SHEFEX experiment is protected by an ogival nose cone during ascent.



Figure 5: Launcher configuration.

The vehicle is accelerated by the S30 motor for 28 seconds until burnout at an altitude of 17 km. At burnout of the first stage the vehicle has already reached a velocity of 930 m/s, respectively Mach 3.17. The fins of the 1st stage are canted and induce a spin rate of approximately 1 Hz for stabilization. In the coast phase, which lasts for 8 seconds, the 2nd stage separates from the S30 motor. To ensure a non-problematic separation, a passive spring-plunger system is adopted. The ignition of the 2nd stage is 36 seconds after launch. Since the SHEFEX experiment poses no control devices, after burn-out the second stage remains attached to the SHEFEX body till almost the end of the experiment to provide flight stability through its fins. As a non-spinning flight for the down-leg phase is required, the 2nd stage fins may not be deflected. The second stage separates just before deploying the parachute landing system.

During the ascent and after burnout of the 2nd stage, the vehicle has reduced its spin rate to approximately one half revolutions per second. At approximately 100 km altitude when the aerodynamic drag has reduced to a negligible level, the Attitude Control System (ACS) removes the remaining spin of the vehicle and aligns the vehicle pitch plane with the trajectory plane. The vehicle then rotates in such form that the longitudinal axis is aligned with the instantaneous flight direction, i.e. a re-entry at nominal zero angle of attack. The experiment phase begins at an altitude of 90 km on the down-leg and end at an altitude of 20 km with the separation of the payload from the 2nd stage motor under hypersonic conditions (**Fig. 6**). Thus, a measurement time of approx. 35 sec is available for measurements in hypersonic regime. The selected flight path fits rather well inside the operational range of the high enthalpy ground facilities of DLR, allowing numerical and ground base data correlations with flight measurements (**Fig. 7**). The payload module with the SHEFEX experiment skips to an instable flight and decelerates in an almost flat spin mode to a recovery velocity of approximately 230 - 280 m/s. At an altitude of 4 km the recovery sequence starts with the deployment of a two-stage parachute system to provide a final velocity of approximately 15 m/s and safe landing.

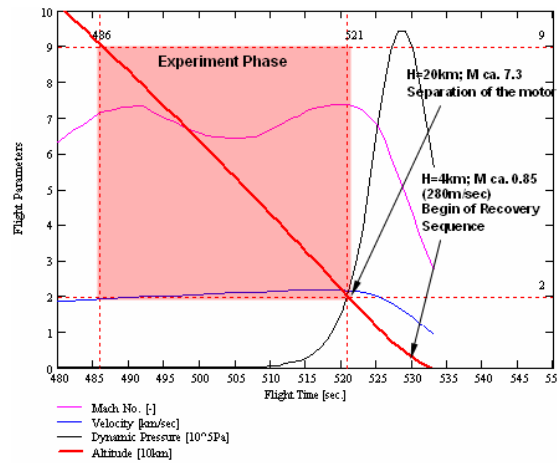


Figure 6: Re-entry parameters for the SHEFEX mission.

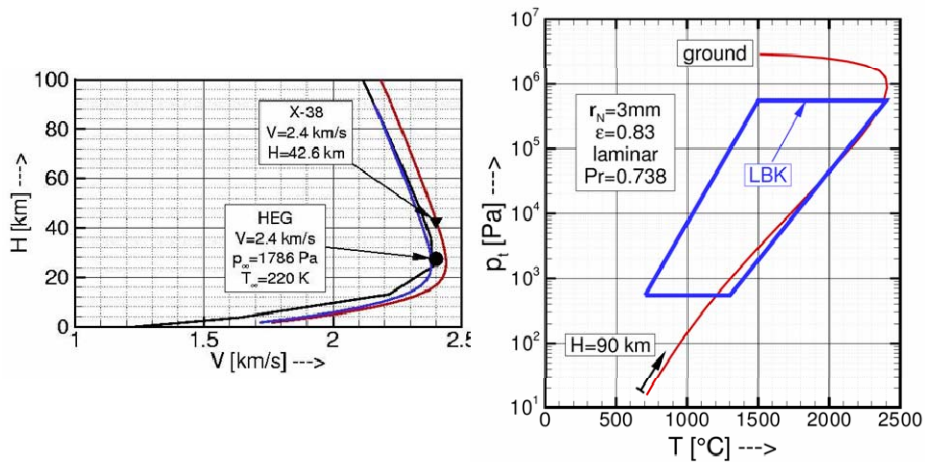


Figure 7: SHEFEX re-entry profile with respect to DLR ground based facilities.

The main sensor for the attitude control is a DMARS-R roll stabilised inertial platform. As low-cost sounding rockets are usually spin stabilized at rates of up to 5 Hz during the boost phase, the use of gyro platforms on such vehicles is limited to those with a roll stabilised or de-spun outer gimbal which decouples the inertial sensors from the high roll rate. The DMARS is an inertial grade device which provides accurate attitude and position which is also an advantage for this mission as not only the re-entry attitude is important but also the accelerations, rates and position which will provide the experimental data on the flight dynamics of the experiment form. The platform comprises in addition to the three gyros and three accelerometers, a computer which generates the navigation data and provides the attitude control signals for the cold gas system. The main microcontroller in the ACS electronics performs data acquisition from the platform, the experiment electronics, GPS receiver, ignition and recovery system and pyrotechnic circuit's computer and the general housekeeping within the ACS and assembles the various data packets in a PCM frame for telemetry. This computer also receives telecommands via dual redundant receivers for correction of manoeuvres or selective enabling of lateral and roll axes control. The control signals from the ACS activate the thrusters in a two level cold gas system with two thrusters each for the two lateral axes and four for the roll axis. The two levels provide high thrust for fast acquisition and low thrust for fine control. The gas system is mounted at the forward end of the vehicle for maximum lateral moment arm. The baseline control strategy is a predefined manoeuvre according to the predicted nominal trajectory. In the event of a trajectory significantly differing from the nominal, the calculation of the flight vector is done from real-time radar data and the transmission of the manoeuvre correction via telecommand. The ACS electronics and service systems are supported by the Electrical Ground Support Equipment (EGSE) which comprises external power and individual switching for the electronics, platform, stable oscillator for slant ranging, two telecommand receivers, two telemetry and one TV transmitters and a radar transponder. To ensure data transmission also in the hypersonic re-entry, a redesign of the existing blade antennae (S-Band, 2.25 GHz) became necessary since former flights with flat spin re-entry showed that degradation of the antenna conductors and insulation occurred because of the high thermal stress and for a stable re-entry the thermal strains on specific points of the vehicle are even higher. The material for the usual blade antenna is a copper-beryllium-alloy (melting point 1030°C). The new antenna has a modified front body increasing the expansion area (**Fig. 8**). The Teflon-insulation of the conductor is changed to Ceramic to improve the thermal insulation. The new blade antenna can also absorb more energy and the heat conduction to the payload modules is improved by a better attachment surface. Tests conducted at the L3K arc-jet facility of DLR show positive results concerning the temperature. A prototype antenna has been tested also in flight on a VSB-30 in October 2004.

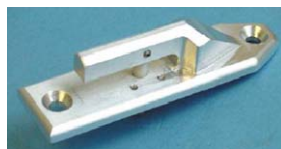


Figure 8: S-Band antenna designed for the SHEFEX experiment.

3.2. Aero(thermo)dynamics and Vehicle Layout [22, 23]

Driven by the faceted concept, two main criteria have been used to define the shape of SHEFEX. It should have as much as possible faceted panels (totally 31) and it should represent as many as possible configuration details of space vehicles, like concave and convex chamfers and a sharp unswept leading edge. The experiment shall enable a detailed investigation of the flow effects and their structural answer during flight. The shape and size of the SHEFEX vehicle is limited by the inner volume of the nose cone fairing of the launcher system. Thus the length of the vehicle does not exceed 1000 mm and the diameter is limited to 500 mm. Having a sharp leading edge the whole surface is created by flat panels with exception of the so called adapter ring, which provides the interface between the chamfered cross section of SHEFEX and the circular launcher structure (**Fig. 9**).



Figure 9: SHEFEX faceted experiment shape.

The aerodynamic and aerothermodynamic studies are performed based on calculations with the DLR Euler and Navier-Stokes Code TAU [24] applying unstructured grids. Additionally, for general parameter studies the DLR surface inclination method HOTSORE [25] is applied. As shown in several comparisons with wind tunnel experiments [26-27] both codes are well established for the aerodynamic layout and analysis of complex space transportation systems. In all calculations perfect gas was assumed which is the worst case assumption from an engineering point of view. The main purpose of the aerothermodynamic analysis of the SHEFEX forebody was the assessment of the expected heat loads and a first analysis of the flow field. At the beginning an average flight Mach number of $M=7.5$ was assumed. Hence, the results represent the upper limit of the expected thermal loads. The calculations cover the complete Reynolds number range of $70.000 < Re < 3.500.000$. Dependent on the altitude the boundary layer is assumed to be fully laminar or fully turbulent. The wall was assumed to behave radiation adiabatic ($\epsilon=0.83$). The analysis of the surface values of pressure, temperature and heat flux are basis for the choice of the flight instrumentation and the positioning of the sensors. **Figures 10 and 11** illustrate the influence of the boundary layer state on the flow field and the surface temperature. Although the forebody is very simple a complex and interesting flow field is obtained. In the backward part of the lower surface which is intended to simulate conditions of a deflected control surface parallel to the hinge line a separation is obtained which clearly decreases in case of a turbulent boundary layer.

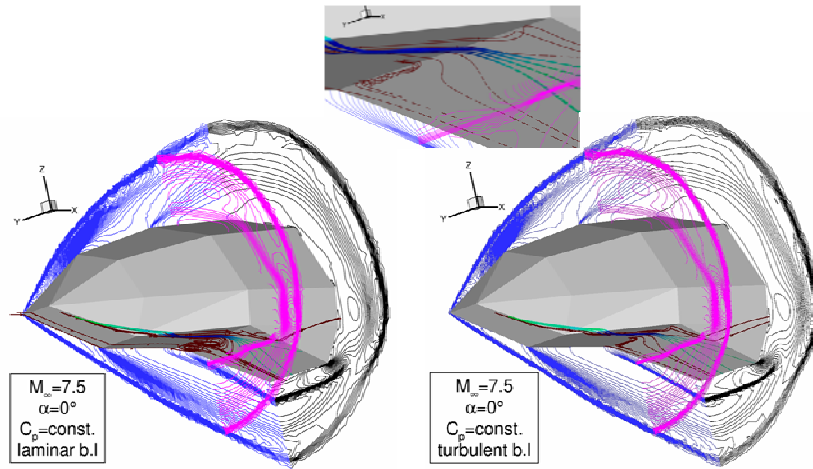


Figure 10: Influence of the boundary layer state on the surface flow patterns at $H=42.6$ km.

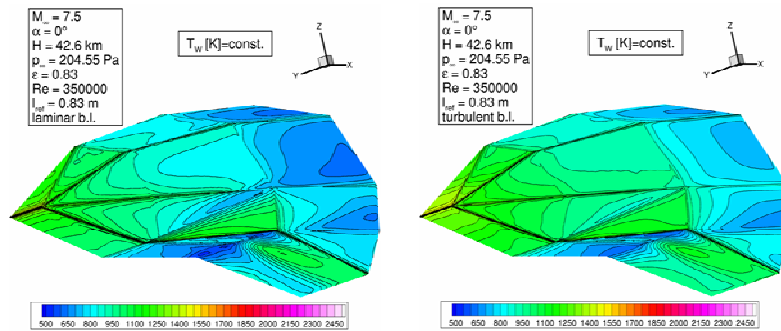


Figure 11: Influence of the boundary layer state on the surface temperature at $H=42.6$ km.

Additionally, the coloured streamlines show, that along the highly swept corner of the second forebody-segment a small vortex is obtained (see enlarged view). This vortex hits on the backward inclined panel and induces a local hotspot as shown in Fig. 11. To capture this phenomenon the number of sensors is increased in these regions. The surface temperatures indicate that except of the un-swept leading edge of the forebody never a critical temperature with view to the resilience of the ceramic panels is reached. The shown temperatures are a measure for the upper temperature limit because they represent the never obtained steady state conditions without taking into account the structural heat conductivity. One critical point of the configuration is the step appearing between forebody and the first cylindrical segment (**Fig. 12**). Here, heat loads in the order of the ones at the un-swept leading edge are expected. In any case structural failures due to thermal loads or shock-shock interactions have to be prevented because they would end in the destruction of the vehicle. Therefore, special emphasis is taken on the analysis of this region. The complex interaction between the separation induced shock and the forebody shocks is shown in Fig. 12. By comparison of the lines of constant Mach number and the pressure coefficients the slip line which might be critical with view to the heat loads can be detected. **Figure 13** shows that the separated hot

boundary layer flow is redirected back into the separation bubble. This results in very high radiation adiabatic temperatures but it has to be taken into account that these results are steady state solutions and that the structural conductivity is neglected. A not fully coupled finite element calculation during the structural layout indicated that a material with high thermal conductivity, like copper, will be able to withstand the heat loads without total structural damage. If the high temperature area between slip line and shock is considered it is obvious that thermal loads coming from this region will not affect the surface flow for the flow conditions expected to appear during the SHEFEX experiment.

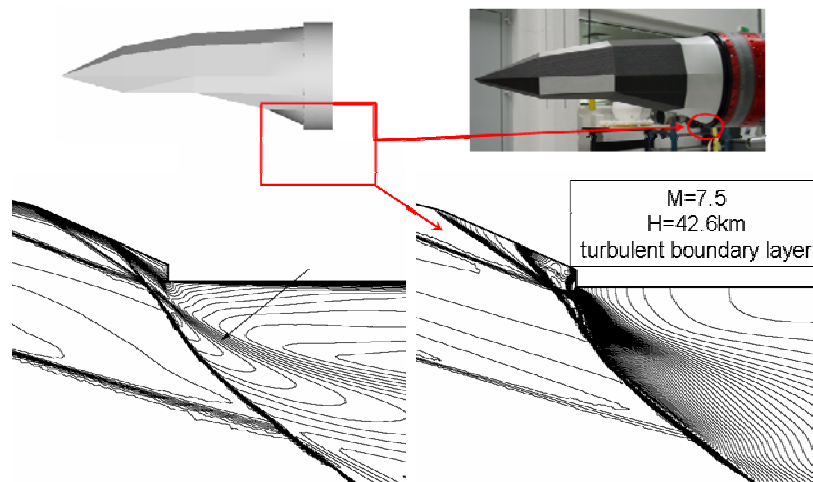


Figure 12: Flow topology in symmetry plane at the interface ring.

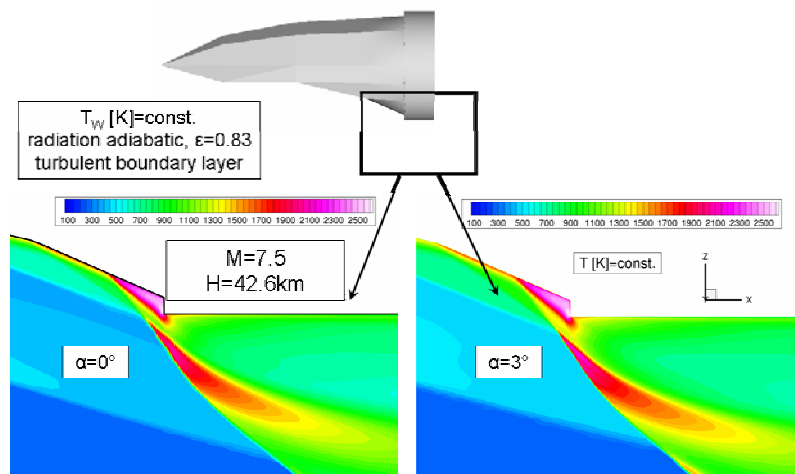


Figure 13: Influence of the angle of attack on the impinging temperature.

The layout of the reentry configuration is dominated by the lift of the asymmetric payload which drastically reduces the static margin. Indeed, the asymmetric shape of the SHEFEX payload and the

resulting lift and pitching moment lead to a non-zero trim angle. The main goal of the aerodynamic layout is the definition of a configuration which is stable with view to the longitudinal as well as the lateral motion. In order to reduce the drag as well as the structural loads the angle of attack has to be as close as possible to the originally projected angle of $AoA < 1^\circ$. Complicating the conditions for the layout is the fact that the fins of the second stage must not be deflected and that the adaptation of the overall center of gravity for the re-entry configuration is very demanding due to the high mass of the burnt-out engine. A first analysis of several fin arrangements with view to the static margin pointed out that only a four fin configuration with two horizontal and two vertical fins would enable to define a realistic reentry configuration. Additionally, the fins are defined as large as possible however the fin size is restricted by the ascent stability. Therefore, a variety of aerodynamic solutions to shift the center of pressure backward have been assessed while the only realistic aerodynamic change is the introduction of a flare in the back of the vehicle. A sensitivity study allowed realizing the best compromise between additional drag and increase of the static margin. In combination with several structural changes to shift the center of gravity forward a stable configuration is realized. The resulting layout of the vehicle exhibits enlarged fins for the second stage adapted to the new flare. This change together with its fully symmetric arrangement, which is required to prevent spin during descent, forces also a redesign of the first stage fins and the corresponding tailcan since the modifications of the re-entry vehicle reduce the vehicle stability during ascent. Therefore, also for the first stage a new tailcan and fins are developed (**Fig. 14**). The structural layout of the flare and its structural optimization concerning mass and stiffness is performed applying ANSYS FEM analyses.



Figure 14: Adopted launcher layout (compare with Fig. 5).

An important chapter of the experiment is also the post-flight analysis. The flight data will be rebuilt by means of the high enthalpy wind tunnel of DLR. The flow conditions to be simulated in the facility fit very well in the flight path of SHEFEX. A wind tunnel model 1:2 scaled, instrumented with 72 sensors (40 thermocouples and 32 pressure sensors) is planned. The test matrix accounts for AoA and side slip sensitivity studies. In addition, a strong numerical rebuilding is foreseen taking advantage of the DLR CFD IMENS tool [28] for fluid-thermal structure coupling analysis.

3.3. Structure and TPS [29, 30]

A low cost easy to manufacture modular test platform has been the criteria follow to design the faceted body. It consists on a primary structure cover by TPS panels easily attached to it. All in-flight measurement techniques are integrated into the TPS and housed in the primary structure. This strategy allows test panels of different materials and/or surface coatings; direct access to the primary structure and the easy replacement of type and/or distribution of sensors. That is, each panel may be used as a self contained experimental platform.

The basic substructure consists of an aluminium frame created by stiff booms and spars. The free space is closed by flat aluminium panels, which are also used for mounting the TPS panels and experiments. Inside the frame, some measurement equipment is integrated. These items are a thermocouple connection and compensation box, pressure transducers and a pyrometer (**Fig. 15**).

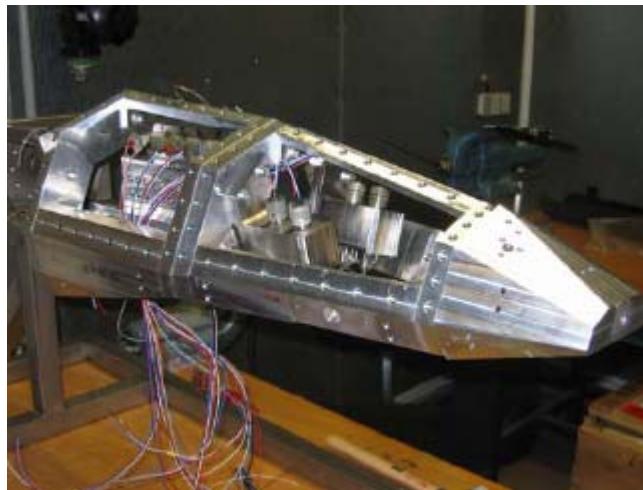


Figure 15: Main aluminium sub-structure including measurement equipment.

The main thermal protection system is designed using CMC panels manufactured via DLR's liquid silicon infiltration process. The basic TPS system is based on a segmented concept developed within an ESA Program. The main element of the concept is a fibre ceramic cover plate, supported in all directions by a so called central post and flexible stand offs at the corners (**Fig. 16**). Thus, the thermal expansion will be not suppressed. Beneath the cover plate a lightweight fibrous ceramic insulation felt is inserted. A key element of this TPS concept is a ceramic fastener, used for the connection of the panel to the CMC stand off and central post. Indeed, all facets of the second and third segment are created by a single CMC panel. However, the facets of the first segment are too small for single panels, thus, this part is designed and manufactured as one integral nose cone, but also connected using flexible stand offs and screw rivets. A novel CMC fastener consisting of a bolt with a conical or spherically shaped head and a split threaded shank has been developed at DLR (**Fig. 17**). At first, the fastener is screwed in with no deformation. When the torque moment reaches a particular level, the shank of the fastener, which is in the nut, is pressed together until the inner cone or the slot limits further deformation. By drawing the inner cone with a pin

axially, the shank of the fastener is spread back resulting in a pretension in both radial and axial directions as both flanges of thread and nut are pressed together. Thus the torque moment during fastening is low. This is another positive effect, because the torsion strength of CMC materials is usually low due to the low interlaminar shear strength of the material. Thus, the screw rivet design provides and combines nearly the same functionality as a common screw and a blind rivet, but the internal stress distribution is more adequate due to CMC material properties. To loosen this element it is only necessary to press down the conical locking element. Within the SHEFEX project, the detailed shape of the fastener has been improved and mechanical strength could be significantly enhanced. In total, a number of 52 screw rivets are necessary for the assembly of the flight unit. Enough to get experience in practical use considering cost effective manufacturing, fixation, quality assurance and loosening procedures after flight.

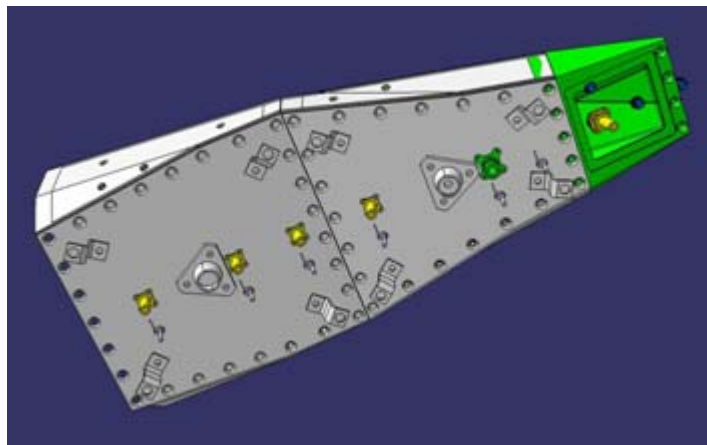


Figure 16: Central post and stand-off arrangement for the TPS attachment.

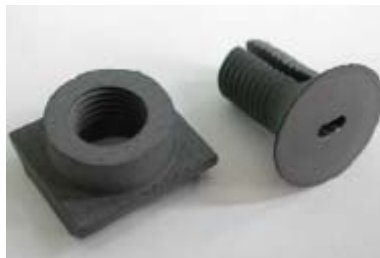


Figure 17: CMC screw rivet for panel attachment.

The second key element for the TPS is the seal design at the interfaces between the panels. During the nose cap development for X-38 a rigid and flexible seal combination was developed. During a lot of ground tests within plasma wind tunnel test facilities, the performance and reliability could be demonstrated. However, the manufacturing effort was rather high to adopt the rigid seal contour close to the panel shape. Since the shape of SHEFEX includes a certain number of different angles at the interface chamfers, a lot of different single parts would be necessary if a new approach is not used (**Fig. 18**). During the last years an oxide ceramic based CMC material was developed at DLR. The so called WHIPOX

material can be used as an oxidation stable alternative to carbon based CMC's. However, temperature stability is limited and a special coating is necessary to improve emissivity and catalytic behavior. Nevertheless, this material provides a flexible intermediate state during the manufacturing process. Thus, it is possible to shape a component (in case of SHEFEX the rigid seal) during assembly. Hardening will occur during operation. Using this property, it is possible to shape and cut all required seal components from one uniform WHIPOX tape. A typical seal interface was tested within a plasma wind tunnel and handling procedure, seal performance and temperature stability could be demonstrated successfully.

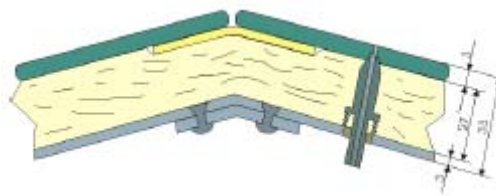


Figure 18: Schematic view of the interface seal and insulation package.

The third key technology is the design of the sharp leading edge, which will be exposed to very high heat fluxes. Currently, two different approaches are under development. The first and simpler solution is to use DLR's C/C-SiC material which has demonstrated a very moderate erosion behavior at extreme heat loads in the past. The idea is to design an expendable leading edge with a quick exchange connection. This concept will be used for the first flight of the SHEFEX vehicle.

Finally the experiment will allow the comparison of manufacturing techniques for C/SiC. In addition two tests are devoted to flexible ceramic and two to metallic TPS. Indeed, in the opposite of the overall TPS design, the facets of the second segment are created by some passenger experiments. Adjacent to the C/C-SiC bottom plate, a cured WHIPOX panel is integrated on each side of the vehicle. The remaining facets of this segment are closed by four different TPS samples provided by EADS. Two ceramic based and two metallic based concepts. Both metallic TPS experiments are delivered by Plansee, Austria. Finally, an additional experiment from MT Aerospace is integrated within the third segment; however, it is characterized only by a different material of the CMC panel.

3.4. In-flight Measurement Technique

The in-flight measurement sensors foreseen in SHEFEX allow to acquire aero(thermo)dynamic flow parameters as well as to qualify modern measurement techniques. A micro-pyrometer and several pressure, heat flux and temperature sensors are integrated in the faceted body, amounting totally 64 measurement points (**Fig. 19-20**). To a calibration phase of each type of sensor in electronic laboratory it follows a qualification phase in arc-jet facilities of the sensor mounting concept. Once the electronic box is housed in the SHEFEX aluminium inner structure, a pre-integration vibration test is performed as a reference for the complete vibration test carried out at IABG/EADS Company in Ottobrunn. Finally two functional tests, one using electrical- and one with physical-signals are done.

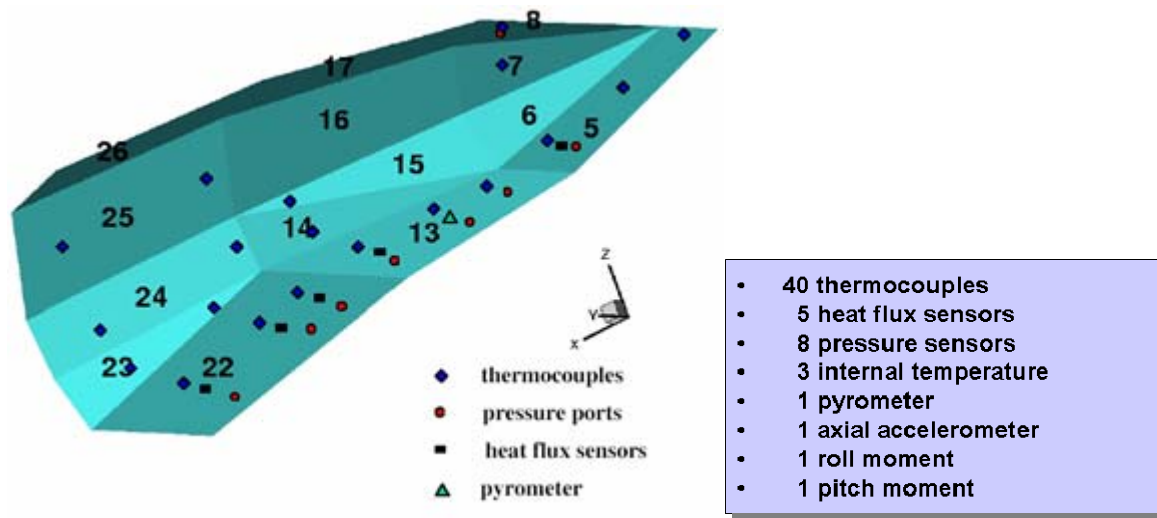


Figure 19: Position and number of instruments

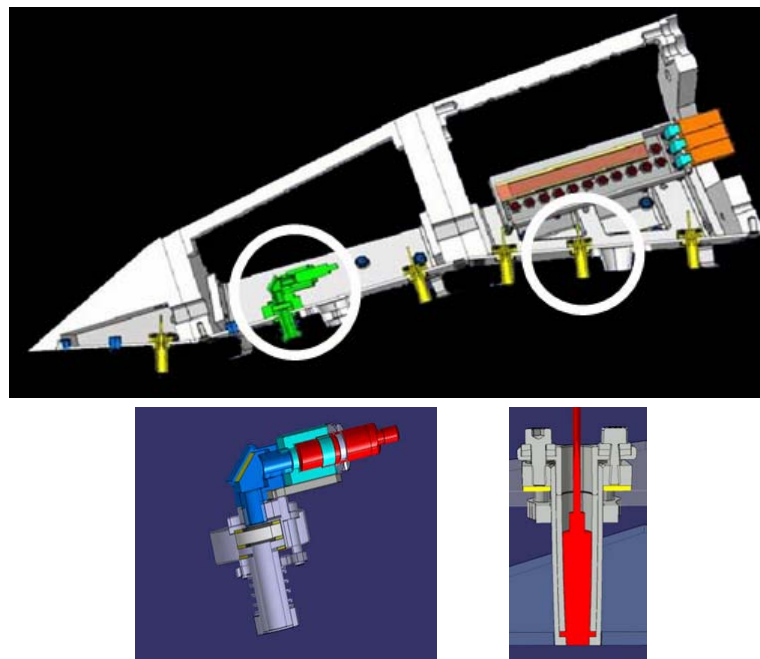


Figure 20: Pyrometer and heat flux sensors layout

4. Summary

A major problem of concern for the physical-modelling success resides in the availability of experimental data for model validation, particularly in the hot hypersonic regime. In the past, validation data have been achieved as secondary product of expensive space-transportation programs. Since in the last ten years there has been almost no successful program due to lack of investment, no new experimental data are available. According, a new trend is emerging for low cost technology validation in flight based on the use of old military rockets. On the basis of the DLR flight experiment SHEFEX, here has been examined the

possibilities of sounding rockets to provide a platform for flight experiments in hypersonic conditions as cost-efficient supplement of tests in ground based facilities.

The SHEFEX experiment is not a new re-entry vehicle but a hypersonic experiment on a new concept for future aerospace vehicles which should have the potential to reduce cost and servicing efforts for the TPS and at the same time enhance vehicle performance due to the hypersonic exploitation of the aerodynamic sharp configuration concept. The main purpose of the flight experiment is the validation of analytical predictions, cross checking the ground testing data and acquisition of flight testing on some structural design features like sharp leading edges and seal system for the development of future space transportation systems. Designed following an unconventional way, is SHEFEX one of the first concepts resulting from a multidisciplinary approach. Finally, SHEFEX is a pathfinder project of an emerging strategy on a kind of sky based facility, as economical way to acquire knowledge in the physics of hypersonic fly.

Sounding rockets may be the way of access to a certain type of “sky based facility”, in analogy to the ground based facilities, concurrent in price also with today’s so-called “industrial wind tunnels”, but in any case, extremely less expensive and less risky than the so-called “X-vehicles”. However, the majority of sounding rockets have been used for experiments which require exoatmospheric trajectories for micro-gravity conditions. As sky-based facility, the experimental phase takes place in the atmospheric down-leg of the trajectory and therefore the layout of the vehicle may be strongly affected by the objectives imposed for the mission-experiment even for similar mission-sequences, as is the case of SHEFEX. Indeed, the aerodynamic layout is dominated by the large amount of lift production of the asymmetric forebody which drastically reduces the static margin of the vehicle. A parameter study to increase the static margin by adaptation of the center of gravity leads to a final configuration with enlarged fins and flare in the back of the re-entry configuration. The changes allow a comfortable static margin and an average flight Mach number of $M=7$. Although the experiment is relatively simple at first view, the results of the aerodynamic requirements end in a nearly complete structural redesign of the original selected sounding rocket layout. Indeed, hypersonic environments are severe enough that even for systems based on sounding rockets, they do not enable routine operations.

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