SHEFEX – Hypersonic Re-entry Flight Experiment

Vehicle and Subsystem Design, Flight Performance and Prospects

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The purpose of the Sharp Edge Flight Experiment (SHEFEX) was to investigate the aerodynamic behavior and thermal problems of an unconventional shape for re-entry vehicles comprising multi-facetted surfaces with sharp edges. The main object of this experiment was the correlation of numerical analysis with real flight data with respect to the aerodynamic effects and structural concept for the thermal protection system (TPS). The Mobile Rocket Base of the German Aerospace Center (DLR) was responsible for the test flight of SHEFEX on a two-stage unguided solid propellant sounding rocket, which was required to provide a velocity of the order of Mach 7. The SHEFEX vehicle was launched on the 27th of October 2005 from the Andoya Rocket Range, Norway. This paper presents the main design features of the vehicle and subsystems, the flight performance and the current plans for our next hypersonic project.

Nomenclature

ACS	=	attitude control system
AoA	=	angle of attack
CFD	=	computational fluid dynamics
CoP	=	centre of pressure
CoG	=	centre of gravity
DMARS	=	digital miniature attitude reference system
EGSE	=	electrical ground support equipment
GPS	=	global positioning system
PCM	=	pulse code modulation
SHEFEX	=	sharp edge flight experiment
TM	=	telemetry
TV	=	television
TPS	=	thermal protection system
UTC	=	universal time coordinated
WGS84	=	world geodetic system 1984

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I. Introduction

THE purpose of the SHarp Edge Flight EXperiment (SHEFEX) was to investigate the aerodynamic behavior and thermal problems of an unconventional asymmetric shape for re-entry vehicles comprising multi-facetted surfaces with sharp edges. The main object of this experiment was the correlation of numerical analysis with actual flight data with respect to the aerodynamic effects and structural concept for the thermal protection system. The Mobile Rocket Base of the German Aerospace Centre (DLR) was responsible for the test flight of SHEFEX on a two-stage unguided, solid propellant, sounding rocket which was required to provide a velocity of the order of Mach 7 for approximately 30 seconds during the atmospheric re-entry. Stable pointing of the asymmetric experiment forebody was achieved by leaving the spent second stage motor with fins attached to the payload and aligning this body to the flight vector with an attitude control system, prior to re-entry. During the initial design of the vehicle it became apparent that the various flight phases of the trajectory presented a number of conflicting requirements which led to a complete re-design of the vehicle.

To minimize the self induced angle of attack (AoA) during the re-entry phase, considerable ballast in the forward end was required and a new flared tail can and larger fins for the second stage motor were developed. In addition, these fins had to be set to zero incidence to prevent spin up during the re-entry experiment phase. During the ascent, an ogive nose cone covered the experiment and removed the asymmetry induced lift forces. These essential modifications to the re-entry vehicle led to an increase in mass and aerodynamic drag as well as reduced apogee and Mach number. As the flared tail can and enlarged uncanted fins on the re-entry vehicle reduced the flight stability of the complete vehicle during first stage burn, a re-design of the first stage motor hardware became necessary. The fins were changed and a new light weight tail can was developed.

An Attitude Control System (ACS) using cold gas thrusters was developed to remove residual spin during the exoatmospheric ballistic trajectory and perform a large pitch maneuver of almost 180 degrees to provide an optimum re-entry attitude. An inertial platform, which was part of the autonomous service module and ACS, measured the attitude and acceleration motion of the experiment and calculated the instantaneous trajectory. Experiment and service system, attitude, acceleration and housekeeping data were transmitted via two redundant S-Band links. Telecommand provided the facility for transmission of discrete and serial maneuvers and operational commands to the payload ACS from the ground in real time. The separation of the payload from the second stage motor for final recovery was initiated after the experiment phase was completed at an altitude of 15 km, under hypersonic conditions. Without the motor, flare and fins, the highly unstable experimental payload attained a tumbling motion, leading to flat spin and extreme deceleration to subsonic recovery velocity at an altitude of 4 km. A two stage high velocity parachute system with ram air flotation bag was developed for sea recovery of the experiment. The SHEFEX vehicle was launched on the 27th of October 2005 from the Andoya Rocket Range, Norway. This paper presents the main design features of the vehicle and subsystems, the flight performance, the main lessons learned for subsequent hypersonic re-entry tests and the future prospects at DLR for this type of experiment.

II. Vehicle Concept and Experiment

At first sight, the technical requirements for the performance of the SHEFEX mission on a sounding rocket appeared to be relatively straight forward. The achievement of hypersonic velocities up to Mach 7 between the altitudes of 90 to 20 km on the descent, required a ballistic trajectory with an apogee of less than 300 km. The budget limitations suggested the use of a two stage motor combination comprising a Brazilian S-30 motor and a military surplus Improved Orion as first and second stages respectively, a vehicle which we had already successfully launched with both parallel (356 mm) and hammerhead (438 mm) payloads as shown in Fig. 1 to apogees of 450 and 315 km. Initial discussions resulted in an experiment geometry of the order of about 400 mm in diameter and 800 mm in length, which required the hammerhead configuration and suggested an anticipated total payload mass of the order of 200-250 kg. It was obvious that the payload alone would not be stable on re-entry and so it was decided to leave the burnt out second stage with fins attached to the payload. A further requirement was to realign the vehicle with the velocity vector during the exo-atmospheric descent and point the experiment with close to zero AoA and approximately zero spin during the re-entry and experiment phase. The only aspect which seemed to pose obvious problems already from the beginning, was the recovery of the experiment and the major part of the instrumentation which, considering a completion of the experiment phase at 20 km altitude and with an anticipated vertical velocity in the order of 2000 m/sec, made the separation of the payload from the motor and initiation of the parachute recovery sequence somewhat more challenging than usual.

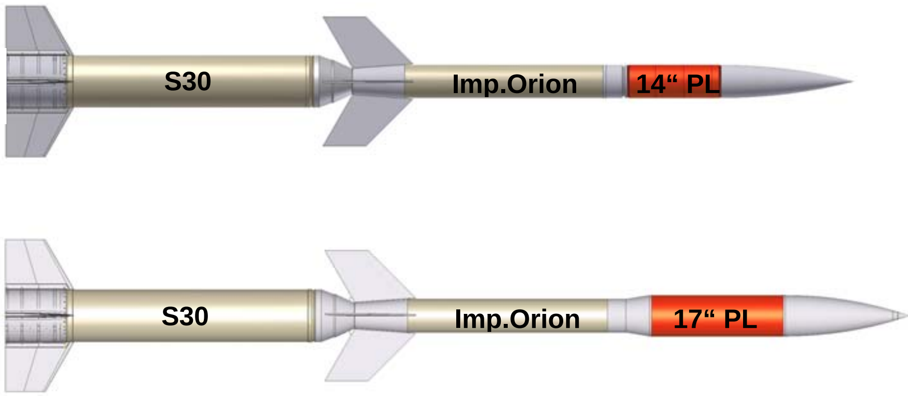


Figure 1. The S-30/Improved Orion Vehicle with 356 and 438 mm Payloads

The final experiment consisted of an asymmetric facetted forebody of 830 mm length and 350 mm diameter at the base, with individual flat panels and sharp edges. An aluminum support structure housed a large number of pressure, temperature and heat flux sensors and to this structure were attached the panels made of ultra high temperature ceramics which, together with flexible ceramic felt insulation, provided the TPS. The use of simple shaped panels reduces the costs of tooling, manufacture and inspection of the TPS and the sharp edges provide improved aerodynamic performance over blunt nosed vehicles at hypersonic velocities. The asymmetric form contained all the characteristic concave and convex chamfer shapes required in a re-entry vehicle and was designed to provide a correlation of the theoretical estimates and wind tunnel measurements of the thermal loads and aerodynamic parameters of such bodies at hypersonic velocities. The facetted experiment was connected to the front of the payload cylinder by an adapter which provided a smooth aerodynamic transition, as shown in Fig. 2.

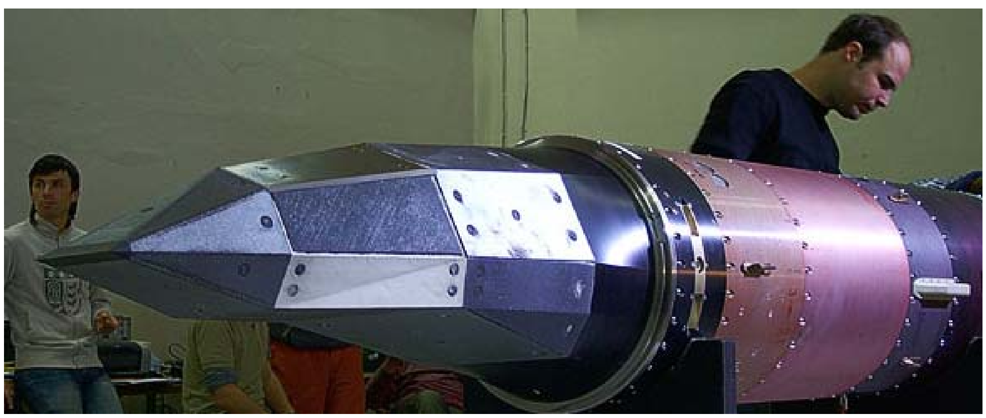


Figure 2. The SHEFEX Forebody

III. Vehicle Design

Driving design factors for both, the re-entry experiment vehicle and the launch vehicle, were the experiment’s requirements. These requirements included a controlled and stable descent flight without any spin-up and at best, zero AoA, especially in the axis containing the experiment asymmetry. To meet these requirements, considerable modifications to the original concept for the re-entry vehicle were found to be essential, which also had a major influence on the ascent behavior, particularly the vehicle’s stability and performance. More extensive investigations were obviously necessary to consider the complete vehicle system from launch to impact and achieve a compromise between experiment goals and technical feasibility.

A. Re-entry Vehicle

Initial aerodynamic investigations including computational fluid dynamics (CFD) analysis of the re-entry stability indicated a major problem in the pitch axis due to experiment asymmetry and associated flight instability.

During descent into denser layers of the atmosphere, the experiment body produces lifting forces which considerably exceed the control capability of the standard four fin combination on the second stage motor or the torque of the attitude control system and leads to an uncontrollable increase in AoA. An extensive aerodynamic study was performed to find an optimum position for both the centre of pressure (CoP) and the centre of gravity (CoG). First steps to increase stability, were modifications in motor and payload cylindrical length by adding spacer modules and slightly enlarging the area for the second stage fins. This strategy was severely limited by the fact that more and bigger fins on the second stage, shift the CoP of the two stage vehicle forward and decrease the static stability margin during first stage burn and in any case the resulting CoP improvement was marginal.

Stabilizing flap mechanisms and movable fins were briefly considered but then discarded due to complexity. Approaches to move the CoG more forward by rearranging the payload modules and installing ballast mass were ineffective as the heavy burnt out second stage motor case does not permit a significant CoG shift. The end result of all of these calculations with respect to minor modifications, was that re-entry stability could only be achieved with a drastic CoG shift of up to 8%, which was not realizable.

A study on the stabilizing effects of flared sections on rockets led to the exchange of the standard boat tail by a conical flare on the Improved Orion. Several different flare cone lengths and angles were considered showing an optimum at the standard tail can length with an opening half angle of 12 degrees. Finally, with adapted and enlarged fins on the flare, stable re-entry behavior in pitch could be realized even when degrading the CoG position slightly with the increased mass of the flare and larger fins at the aft end as shown in Fig. 3.

Figure 3. The SHEFEX Re-entry Vehicle

B. Launch Vehicle

Based on this new layout, the complete interstage section had to be modified as well. The standard Nike to Improved Orion interstage adapter was attached to a newly designed S-30 motor adapter and fairing, providing a smooth stage transition and low mass. Continuing with ascent stability calculations, the design changes on the second stage caused the anticipated problems regarding static margin and performance. As a solution, the standard fins for the S-30 motor were exchanged for the newly developed second stage fin set from the VSB-30 vehicle. Unfortunately, the VSB-30 tail can assembly with three fins still showed a lack of aerodynamic stability and this heavy load bearing version increased lift-off mass considerably. A modification to this tail can was carried out by replacing the former integral construction by a light weight ring and sheet tail can, supporting four VSB-30 second stage fins. Now, sufficient flight stability also for the up-leg was assured. As a positive side effect of the interstage redesign, it turned out to noticeably reduce drag in the supersonic speed regime and hereby compensate the clearly higher vehicle total mass. With this layout, trajectory calculations resulted in an apogee of 270 km guaranteeing the desired re-entry velocities. Fig. 4 illustrates the complete vehicle for the lift-off configuration.

Figure 4. The SHEFEX Complete Vehicle

C. Final Vehicle Configuration

Now with an acceptable two-stage vehicle configuration, the detailed investigations on the resulting descent AoA and associated loads on the vehicle, stability in yaw and roll, stage separation and thermal loads commenced. A final trim to an AoA of close to zero degrees would have required such a large amount of ballast mass as to reduce the vehicle performance to an unacceptable level. As a compromise, a resulting maximum AoA of approximately three degrees for trimmed flight was obtained by integrating a 20 kg experiment adapter in the payload forward section. The corresponding bending loads for this comparatively high AoA were taken as design criteria for component layout including a margin of one degree. Stability in yaw was not considered as critical because the vehicle was fully symmetrical in the yaw plane, with the large Orion fins counteracting any perturbations.

It is usual to offset (cant) the fins of all stages to provide spin stabilization during the ascent phase, however, to ensure a non spinning flight on the down-leg, the second stage fins had to have no offset. The result of pre-flight parameter computation shows the induced spin rate by first stage fin incidence of around 1.0 Hz dropping down abruptly to zero after stage separation. This leads to higher impact dispersion and less stability during the ascent, especially for second stage ignition, and complicates exact trajectory prediction. Even when increasing the first stage fin cant angle up to a maximum, the degrading effect in spin after separation remains. The modified interstage section also required investigations into the stage separation as simple aerodynamic drag separation was no more assured, due to the flare and larger fins and consequent drag on the re-entry vehicle. To guarantee a safe separation before second stage ignition, a small gap between the motors is necessary, which was achieved by a spring driven separation unit consisting of four plungers, installed in the interstage section, pushing against the flared tail can and ensuring a slit at tail off of first stage thrust. Fig. 5 and 6 present the calculated ascent and decent main parameters.

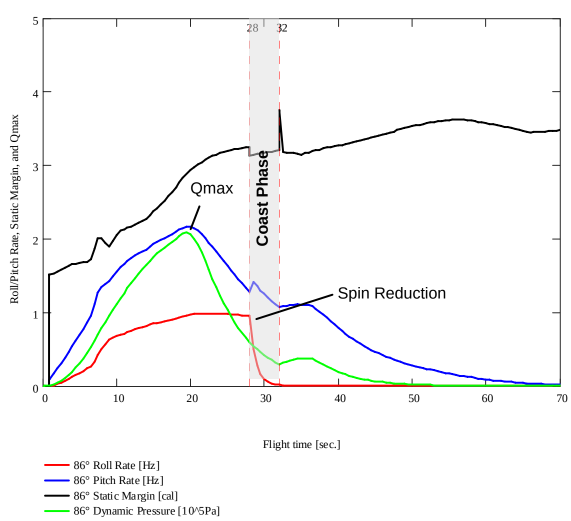


Figure 5. Predicted Ascent Parameters

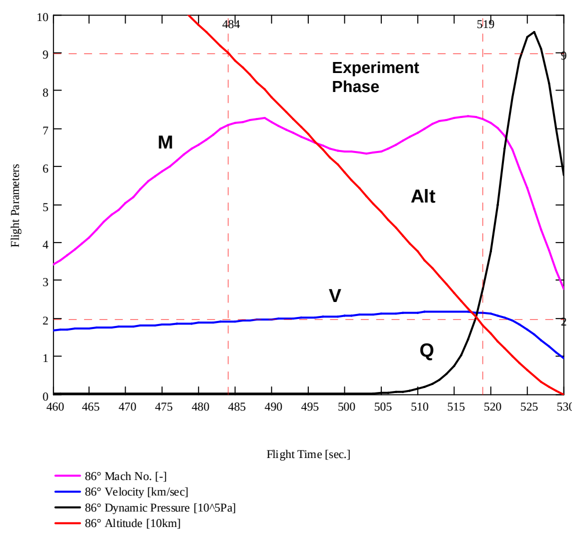


Figure 6. Predicted Re-entry Parameters

Regarding the thermal loads on the vehicle components during the experiment phase, the choice in sophisticated material was limited and protection was realized by applying ablative coating on critical parts such as fin surfaces and particularly leading edges and flared tail can. The final vehicle configuration then represented the evolution of a standard sounding rocket adapted to the specific scientific requirements of hypersonic research vehicles which together with the SHEFEX experiment forebody resulted in the complete experimental vehicle shown in Fig. 4.

IV. Payload Service Systems

The payload shown in Fig. 5 consisted of the experiment which was covered by an ejectable ogive nose cone during the ascent and the service systems comprising the ACS, data acquisition and processing, telemetry and telecommand, pyrotechnic initiation and recovery systems. To provide an adequate diameter for the experiment section, a 438 mm diameter, hammerhead configuration was used. The main sections comprised the experiment and adapter which were covered by a spring loaded ejectable nose cone, the cold gas thruster system for the ACS and an extension bay with telemetry, telecommand and radar antennae, a watertight module for the experiment data acquisition system, ACS DMARS platform and electronics, telemetry and telecommand processing, transmitters, receivers, radar transponder, GPS and power, a section for pyrotechnics and recovery system electronics and the recovery system. The payload was connected to the Improved Orion motor via a high velocity separation system with manacle ring and a conical adapter. Two CCD TV cameras were incorporated in the payload. The main camera was mounted in the payload aft conical section and looked backwards to the VS30 and Improved Orion fin assembly. After separation of the payload from the Orion motor and conical section, a second camera mounted in the recovery system was located to view the motor release, heatshield, drogue and main parachute deployment. The selected video signal was fed to one S-band TV transmitter and a solid state video recorder.

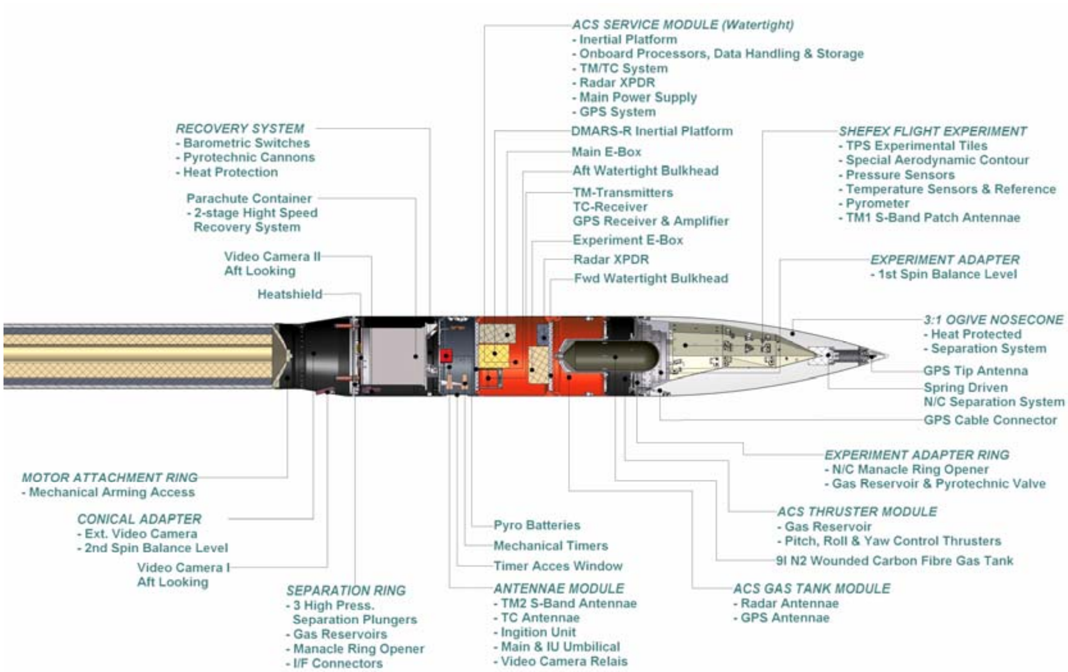


Figure 7. The Payload Layout

A. Attitude Control and Data System

The main task of the ACS was to remove residual spin after leaving the atmosphere and subsequent to ejection of the nose cone, then align the vehicle pitch plane with the trajectory plane and re-point the re-entry vehicle such that the experiment polar axis was aligned with close to zero AoA to the velocity vector during the descent into the atmosphere. The main sensor for the attitude control was a DMARS roll stabilized platform which contains a despun gimbal system which isolates its two dynamically tuned gyros and three accelerometers from the vehicle spin rate during the motor boost phase. The DMARS is an inertial grade sensor which provides not only Euler and Quaternion attitude angles for the ACS, but also accelerations, angular rates and navigation position, which were important for providing scientific data on the flight dynamics of the experiment. The DMARS internal computer, which controlled the platform and drift compensation and performed the navigation calculations and the coordinate transformations, was also used to perform the attitude control calculations and generate the thruster control signals. The main microcontroller in the service system performed data acquisition from the platform, the experiment electronics, GPS receiver and general housekeeping within the ACS as well as from the ignition and recovery system and assembled the various data packets into a PCM telemetry frame. This computer also decoded telecommands for the initiation and correction of maneuvers, selective enabling or disabling of lateral and roll axes control, selection of various DMARS operational modes and also controlled the data and video recorders.

B. Data Acquisition, Telemetry and Telecommand

The PCM telemetry comprised a fixed frame of 180 bytes with 4 bytes synch word and 4 bytes frame counter. The serial data from each of the main sources, each with its own header, identifier and time tag, was placed sequentially in fixed blocks within the frame and unused spaces within blocks were filled with zeros. This simplified the extraction and reconstitution of data packets on the ground as a decommutator was programmed to select and distribute the contents of each block to the appropriate serial interfaces and quick look displays. The length of each block in bytes and the frame rate of slightly greater than 108 frames per second, were designed to provide approximately 20% reserve over the maximum data rate of all sources. The telemetry therefore provided a transparent communication of five serial interfaces with a total rate of 156.25 kbit/s. Experiment data pre-processing was performed by a dedicated electronic box containing four analogue cards, one differential interface and power conditioning card and two microprocessor cards. Signals from the experiment sensors were oversampled, digitally filtered and the resulting data assembled in a serial package and transmitted to the main housekeeping processor via an RS 422 interface at 38.4 kbaud. The DMARS inertial platform produced two data packets. One packet with a rate of 115.2 kbaud comprised the status and output of its sensors and the functions which were required for the pre-launch setup and control of the platform and the attitude control parameters. The other interface with a rate of 38.4 kbaud contained the instantaneous rate, attitude, position and thruster status. The GPS receiver and processor provided a packet at 19.6 kbaud. The acquisition of the various serial data packets, the generation of a housekeeping packet with 38.4 kbaud and the assembly of the PCM frame, were performed by the main control processor. Because of the relatively large slant range, low elevation and possible additional attenuation due to high temperature ablation or possible vaporization of sharp surfaces during the main experiment operational phase, the PCM telemetry was fed to two S-band transmitters. TM 1 shared a set of four hook antennae on the skin with the TV transmitter. TM 2 used a pair of patch antennae mounted under the ceramic tiles. The latter transmitter could also be switched by telecommand to two hook antennae on the skin until the nose cone was jettisoned or in case the patch antennae failed. In addition, the telemetry data was stored on board in a solid state recorder. All eight hook antennae for telemetry and GPS were redesigned with a broadened leading edge to improve their thermal robustness during re-entry.

The telecommand system comprised an encoder with 24 discrete signals and one serial input at 38.4 kbaud. The discrete signals were set by switches and were used to provide individual operation and activation of thrusters and thrust level control for testing and in case of an emergency, selection of the DMARS platform mode control for command interface and pre-programmed maneuvers, operation of the data and TV recorders and selection of the GPS and TM 2 antennae. The serial input carried the maneuver commands for the ACS which were entered as an Euler angle on the EGSE and converted to quaternions for transmission to the DMARS platform. The radio frequency side of the telecommand comprised two antennae which were thermally protected by high temperature plastic covers, a coupler and two receivers. Using the AGC level, the main processor selected which of the two receivers provided the data. To reduce the duration of the serial data string and the possibility of interruption due to disturbances in the RF link and AGC selection, the serial packet from the EGSE containing the quaternion for the command offset and a message number, was stripped of all additional overhead, packed in the telecommand format together with the message number and after reception and checking by the flight decoder, the original format was regenerated and passed to the DMARS platform. The message number from the EGSE was retransmitted in the main DMARS telemetry packet and displayed on the EGSE in case a serial command was lost in transmission, which could have occurred in the event of the antenna pattern passing through a critical minimum during the reception of a serial string. In case of a failure of the telecommand link during flight, the flight processor was loaded with a default maneuver sequence prior to launch. In the event of a failure of data reception for 10 seconds, this sequence would have caused the ACS to perform the required maneuvers automatically, based on the requirements for a nominal trajectory.

C. Electrical Ground Support Equipment

The ACS and service systems were supported by Electrical Ground Support Equipment (EGSE) which comprised the pre-launch power switching and control for the electronics, DMARS platform, stable oscillator for slant ranging, two telemetry and one TV transmitters, two telecommand receivers and a radar transponder. The EGSE also contained a microcomputer controlled decommutator which provided selection and distribution of data packets to five personal computers for data management, quick-look display and archiving. A further feature of the EGSE provided simulation of payload dynamics and control of a rate table for closed loop DMARS platform control tests. A schematic diagram of the data links between the ground and flight systems is shown in Fig. 8. Pre-launch control parameters were loaded either via an umbilical interface or telecommand and data was received via umbilical or TM.

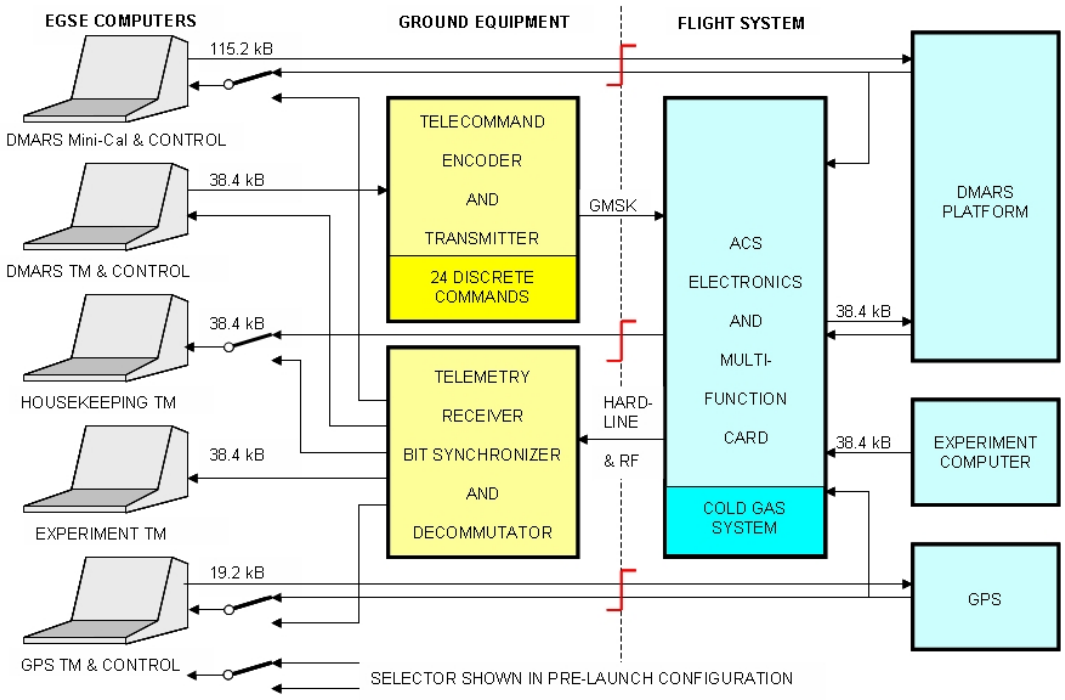


Figure 8. Schematic Data Links between Ground and Flight Systems

D. The Recovery System

The recovery operation was considered to be a high risk task. It consisted of separation of the hammerhead payload from the Improved Orion motor at an altitude of less than 20 km on the descent trajectory at a velocity of more than 2000 m/sec. Various methods of initiation were considered including a signal from the DMARS inertial platform navigation computer or telecommand activation based on real time radar trajectory data, but finally the simplest solution was to use two redundant barometric pressure switches attached to the recovery system pressure measurement manifold, operating at 15 km altitude. The separation system was a pyrotechnically actuated, high pressure gas manacle ring release and plunger system which was initiated by the pressure switches. The aerodynamic forces on the asymmetric experiment forebody ensured that the payload would quickly separate from the Improved Orion motor and as it was inherently unstable, it would tumble and decelerate to a velocity of less than 250 m/sec.

By the time it reached an altitude of 3.5 km the recovery sequence was to be initiated by a barometric switch which activates heat shield ejection and deploys a specially designed, high velocity drogue parachute. This ribbon parachute contained a de-reefing delay of 5 seconds and was designed to decelerate the payload to a decent velocity of 60 m/sec. After a further 15 seconds, a tri-conical parachute was to be deployed in a reefed condition and then also de-reefed after 5 seconds, leading to a final sink velocity of 15 m/sec, before sea impact. The main parachute contained a ram air filled 270 liter flotation bag for buoyancy and a recovery beacon providing a floating time of the payload of at least 48 hours.

V. Launch Campaign

The S-30 motor was delivered directly by aircraft from Brazil to Andoya island, Norway, two weeks before the start of the campaign. The launch campaign was planned for 10 days on range and the activities included the assembly of the motors and the payload, installation of the EGSE, final calibration and test of the payload sub-systems and experiment, flight simulation tests, installation on the launcher, a test countdown, flight readiness review and the hot countdown. The original launch date was set for 28th October but because of the chance of bad weather, the campaign schedule was planned with one spare day. This was fortunate as the weather forecast for the 28th October and several days after, was extremely bad so we were able to be ready for a launch on the 27th October. The 3 hour countdown was commenced at 6:00 hours local time but numerous holds were caused by problems with the TM ground stations and the presence of fishing boats in the first stage impact area. The countdown was somewhat more complex than usual because of the necessity to perform the self alignment of the DMARS inertial platform and enter all the parameters for the flight control and maneuver. Eventually the countdown progressed below T-30 minutes only to be stopped at T-3 minutes by a safety problem in the launch area. With less than 30 minutes daylight remaining (necessary for the TV cameras) and the prospect of no launch possibility due to bad weather for several days, an abbreviated countdown procedure was quickly generated and the vehicle launched. The communication and tracking systems in flight are shown schematically in Fig. 9 and the main mission control station in Fig. 10.

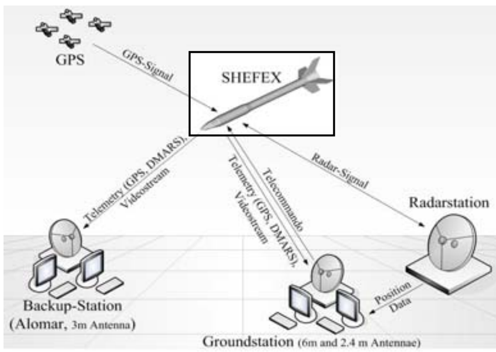


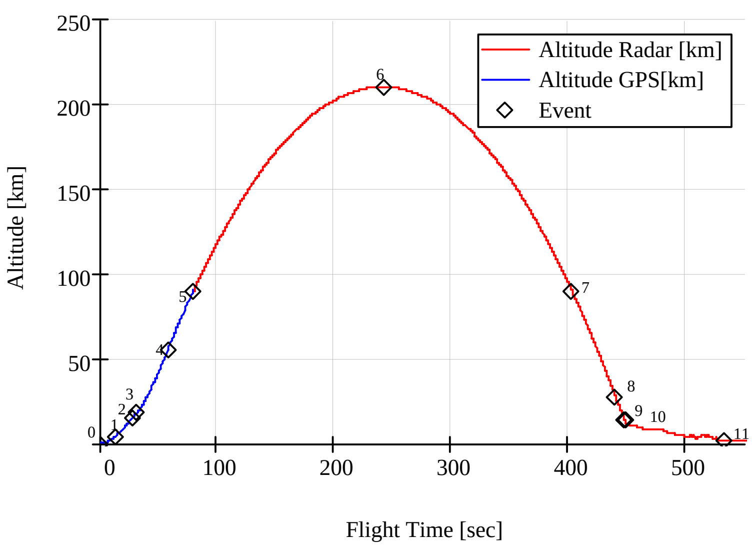
Figure 9. The Communication System

Figure 10. The Mission Control Station

VI. Flight Performance Analysis

The SHEFEX lift-off time T0 is indicated by the umbilical extraction from the vehicle at 13h 45min 30.4sec UTC. The exact definition of the lift-off time is important for the time correlation between the different data sources such as GPS, Radar and housekeeping. The measured altitudes are based on the WGS84 ellipsoid.

The apogee was radar-tracked to 210.6 km which differed by minus 24 km from the latest pre-flight prediction. Explanations for this lower performance are the lower launcher elevation with 82.7 degrees, higher than expected motor masses of 25 kg and uncertainties in the drag calculation for the complete vehicle. As a consequence, the lower elevation led also to a higher ground range of 190 km. The actual launch azimuth was 327.4 degrees resulting in an actual heading angle of 310 degrees. The total flight time from lift-off until loss of telemetry before splash down into the Norwegian Sea was 533 seconds comprising 45 seconds of experimental phase between 90 km and payload separation at an altitude of 13.8 km. Fig. 11 summarizes the events during flight with the corresponding radar/GPS derived trajectory.



No.	Flight Time [sec]	Event
0	0.0	Lift-off, Umbilical Extr.
1	14.4	Max. Roll-Rate 1.83Hz
2	28.7	Stage Separation
3	32.1	Ignition 2nd stage
4	58.3	Burn-Out 2nd Stage
5	80.1	Nose Cone Ejection
6	242.0	Apogee 210.6km
7	403.1	Start Experimental Phase
8	440.0	Max. Velocity Mach 6.4
9	448.3	P/L Separation
10	448.7	Max. Deceleration >50g
11	533.5	Loss of TM 2.1km

Figure 11. The Radar/GPS Flight Trajectory and Events

A. Ascent Phase

The initial maximum acceleration of 7 g during the S-30 burn phase occurred immediately after lift-off decreasing to 6 g shortly prior to tail-off. At T+14.35 seconds the vehicle spin rate reached its maximum value of 1.83 Hz. Maximum dynamic pressure loads of 165 kPa (3,450 lb/ft²) can be detected at T+20.50 seconds, peak velocity is 862 m/sec. The stage separation started at T+27.90 seconds with the disconnection of the motor pressure interface connector and ended physically with the loss of the roll transfer at T+28.70 seconds. At this point a roll rate of 1.61 Hz could be measured including a slight precession motion, decreasing immediately as predicted to zero within 1 seconds after separation. Contrary to the calculations, the roll rate changed its direction and started building up to a maximum spin of -0.9 Hz. First explanations for this unexpected behavior indicate that the structural stiffness of the second stage tail can, transferring the torsion moments from first to second stage was possibly inadequate. Also, possible tolerances in the second stage fin settings have been considered. These ideas are the basis for further investigations. The second stage ignited at T+32.10 seconds and reached a maximum acceleration of 15 g at T+36.15 seconds. The tail-off of the Improved Orion motor started at T+53.98 seconds and burn-out followed at T+58.3 seconds. The maximum ascent velocity was logged with Mach 5.6 at T+56.35 seconds. At T+80.10 seconds the protective nose cone was ejected. Fig. 12 and 13 illustrate the measured ascent and decent flight parameters for altitude, main acceleration, main velocity and roll frequency. The performance of both motors was nominal.

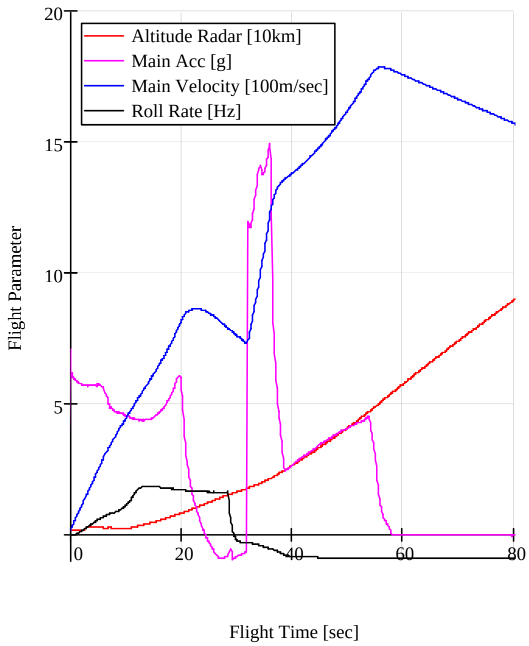


Figure 12. The Ascent Parameter

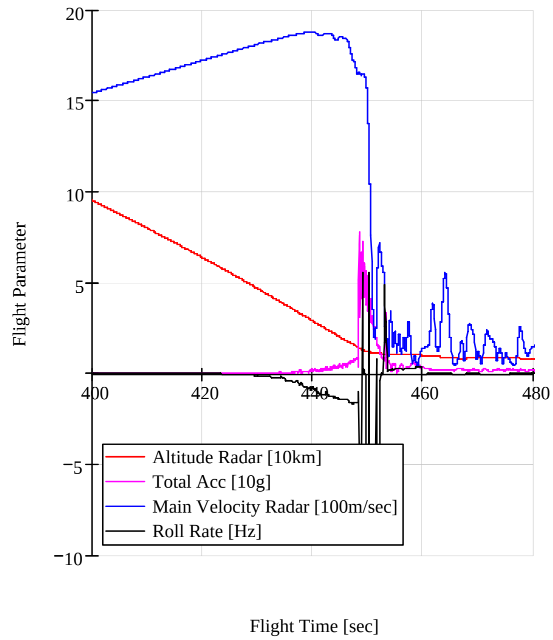


Figure 13. The Descent Parameter

B. Exoatmospheric and Experiment Phase

The ACS roll control thrusters eliminated all roll motion during the 70 seconds following nose cone ejection but consumed a significant amount of gas reserve. The pitch and yaw movements of the vehicle were controlled manually to set the calculated re-entry attitude. Due to the excessive consumption of cold gas for the unexpected despin, a slow tumble motion was still remaining when the vehicle approached the atmosphere. At little less than 100 km of altitude when first fin stabilization effects became obvious, the re-entry vehicle started aligning itself to the velocity vector. The beginning of the experiment phase, defined by 90 km altitude, started at T+404.95 seconds. First atmospheric effects on the acceleration sensors could be observed at around T+410 seconds, corresponding to 80 km. The pitch and yaw angle also started now to oscillate around a mean value and the roll rate was increasing again with atmospheric influence starting to build up. The vehicle finally established a stable flight attitude with a decreasing ellipse-shaped precession around the flight vector induced by the lift forces and visualized in Fig. 14 and 15. The final flight AoA in altitudes below 30 km complies well with the prediction of 2.5 to 3.0 degrees. The unintended spin complicates the analysis of the experiment data.

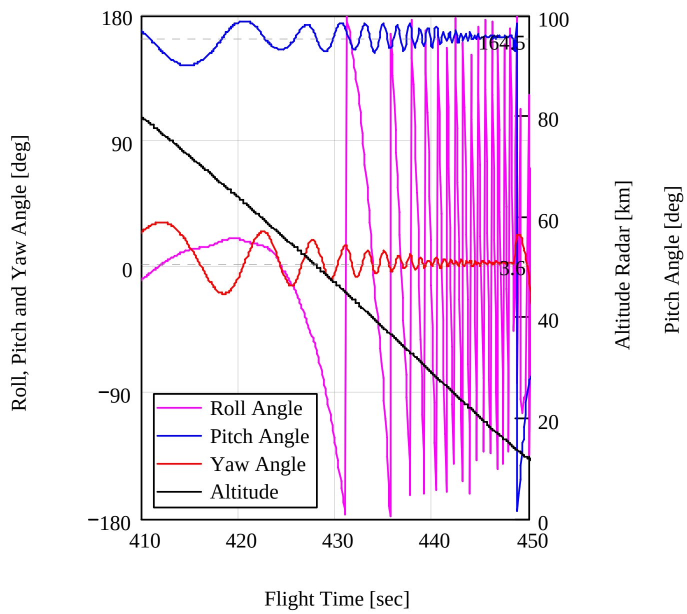


Figure 14. The Re-entry Motion

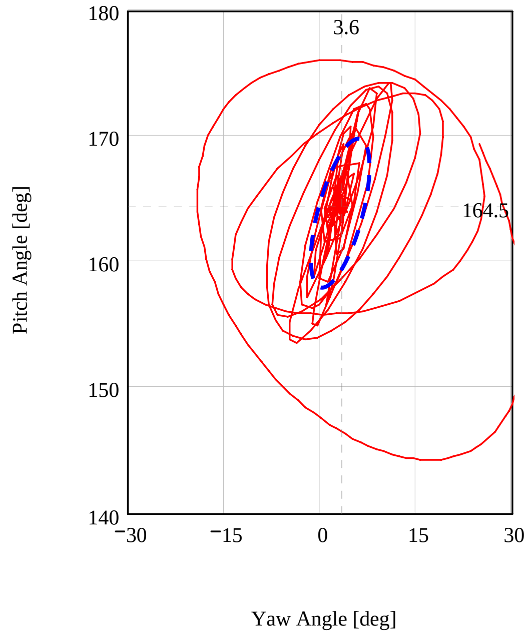


Figure 15. The Pitch over Yaw Motion

The maximum re-entry velocity of Mach 6.4 was achieved at T+440 seconds at 28 km altitude. From then on the drastically increasing dynamic pressure started to decelerate the vehicle significantly. When passing an altitude of 25 km the onboard camera showed severe thermal degradation of the fin leading edges resulting in a sine shape deformation and glowing, see Fig. 16. In the following period, shortly prior to the motor separation at T+445.50 seconds, three shocks within one second could be recorded as displayed in Fig. 17. In combination with the sudden increase in temperature of the sensors inside the experiment, damage to the TPS has to be assumed leading to hot gas flow into the forebody structure. Additionally, the video images showed a smoke plume. This interesting event is still subject to further analysis. The TPS was in any case not designed for high flight velocities at altitudes lower than 20 km corresponding to dynamic pressure loads exceeding 600 kPa (12,530 lb/ft²). The temperature, pressure and heat flux sensors transmitted data during the complete experiment phase until the payload separation and even after.

Figure 16. Visible Evidence of Thermal Loads on the 2nd Stage Fins

At T+448.34 seconds in an altitude of 13.8 km the payload separation was initiated nominally by barometric switches. Without the stabilizing motor and fins the payload started immediately to turn and tumble with decelerations exceeding 50g (saturation of sensors) for the following 3 seconds. Only 0.2 seconds after separation, the recovery sequence was initiated, starting with the ejection of the heat shield and drogue parachute deployment. As flight velocity at this time was around Mach 5.6 this resulted in immediate loss of both parachute stages which could also be observed on the onboard TV. By turning from stable to instable attitude, the recovery system's manifold static pressure ports, used to feed the barometric switches, were oriented fully into the air stream. The measured peak pressure was 820 mbar including also the dynamic component. Hence, as the threshold for the recovery activation barometric switch was 550 mbar, the sequence was initiated, illustrated in Fig. 18. Without any deceleration device, the payload showed smooth flat spin motion on further descent, decreasing in velocity down to about 100 m/sec. At T+533.50 seconds in 2.1 km the TM signal was lost due to horizon blanking.

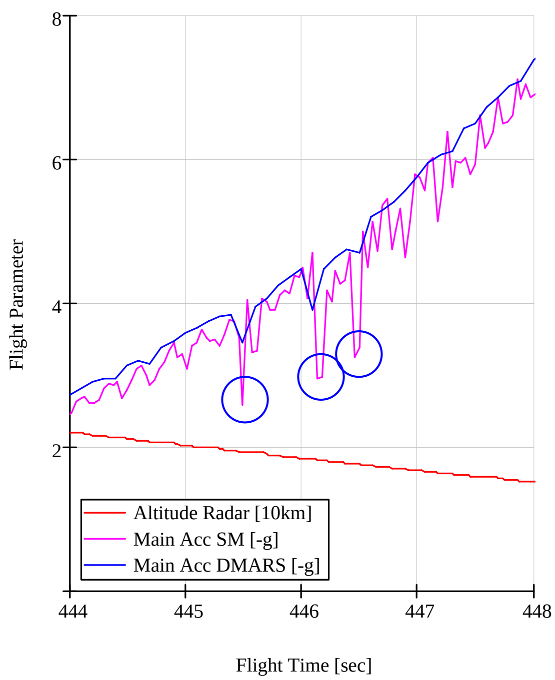


Figure 17. The Shocks during Experiment Phase

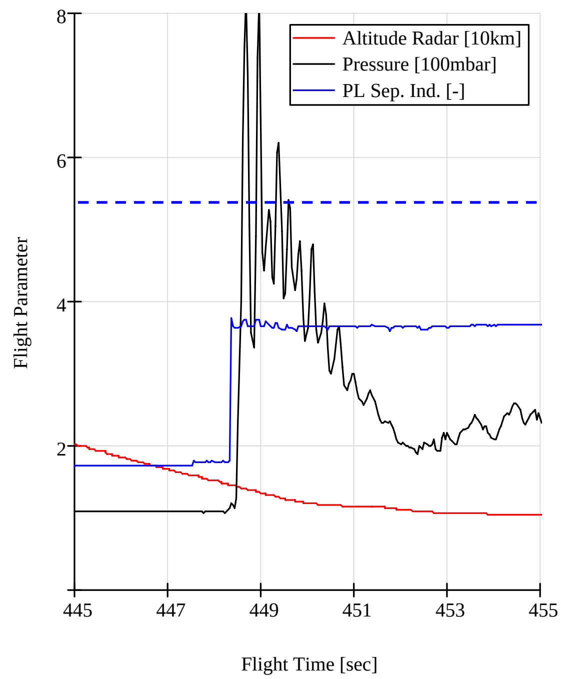


Figure 18. The Separation Sequence

VII. Prospects and Preliminary Concept

The success of the first SHEFEX scientific mission and the experience gained with many of the specific problems of hypersonic research vehicles, has resulted in an enthusiastic support from the participating groups to continue this program but with a number of additional and challenging technological targets. The next experimental flight will focus on hypersonic guidance with moving canard fins in conjunction with a thermal protection system. Contrary to the first flight, the facetted experiment forebody will be symmetrical. For the flight experiment the boundary conditions demand at least a velocity in the order of Mach 7 on the downleg part of the trajectory. Flight control maneuvers will take place in the lower layers of the atmosphere. The experiment phase should end at an altitude of 20 km with subsequent recovery. As a possible launch vehicle, first focus was set on the VSB-30 which was developed in cooperation between DLR Germany and IAE/CTA Brazil as a replacement for the Skylark 7 and is a motor combination consisting of a S-31 as booster and the S-30 as a second stage and can carry a 400 kg payload to an apogee of 250 km. The adapted version first considered for the subsequent mission is shown in Fig. 19.

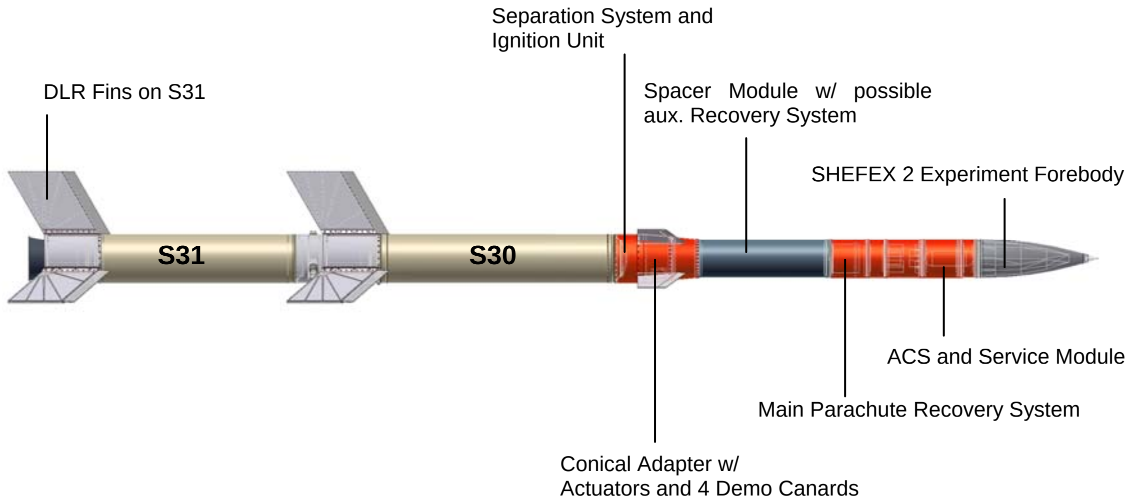


Figure 19. The Modified VSB-30 Vehicle for Hypersonic Research

It was obvious that increasing the vertical velocity would shorten the experiment time, which particularly for a canard guidance and aerodynamic thermal loading experiment, should be as long as possible. The problem was how, with the available moderate cost vehicles, we could provide a flatter trajectory and where we could obtain a long enough ground range, with the possibility to send commands to and receive telemetry data from the payload in real time without interruption throughout the whole flight and particularly during the re-entry phase. The problems of long range recovery of the payload, especially from a water impact are also significant. A solution was found with a different vehicle and the same launch site, albeit with an interesting extension of the impact area.

The vehicle now foreseen, is a Brazilian VS-40 which was designed as a vacuum test bed for the S-44 apogee motor of the Brazilian satellite vehicle VLS. The first stage S-40, which is also used as the 3rd stage of the VLS, is in this case passively spin stabilized with canted fins and accelerates for 62 seconds. After a coast phase and separation, the spinning S-44 with payload is ignited exo-atmospheric and provides a 60 seconds thrust phase, and with a payload of around 350 kg can reach an apogee of the order of 800 km and a launch elevation of 79 degrees. The motor diameter for the S-40 is 1007 mm and for the S-44 1009 mm respectively.

The question was whether we could re-point the second stage before ignition and use most of its energy to provide a greater horizontal velocity. As we need an exo-atmospheric three axis ACS to provide an acceptable initial AoA for the canard experiment phase after second stage burn-out, despin, fairing ejection and payload separation, the solution was to provide in addition a two axis spinning ACS mode after first stage separation. This is used to adjust the spin rate and re-point the second stage motor and payload before ignition. The precession maneuver from the attitude resulting from the first stage burn to the required ignition attitude of the second stage, can be performed with additional cold gas thrusters in the ACS. Due to the thrust limits of the cold gas system and high inertia of the vehicle, the re-pointing and correlation of attitude with the required flight vector before ignition of the second stage, will be of the order of 2 minutes. An active control of the S-44 requires the addition of a command destruct system. After burn-out of the second stage, the payload and motor will be despun with a yo-yo system, the fairings covering the re-entry vehicle stabilization fins ejected and the payload controlled in all three axes to set the re-entry attitude. The current evolution for the flight vehicle including several modifications of the standard configuration is shown in Fig. 20.

As launch site, Andoya Rocket Range is proposed, the same as for the first SHEFEX test flight but including also the facilities at Svalbard island, 800 km to the north, which provide tracking and receiving stations and infrastructure which will be used by our mobile telemetry and telecommand stations and also has access to two optical fiber links from there to the launch site at Andoya, which are essential for uninterrupted communication with the vehicle and payload over the full trajectory. A further considerable advantage for this mission is that the tracking facilities at Svalbard are on a plateau 300 meters above sea level, which will provide us with line of sight communication during the whole of the experiment phase. The final advantage is that if we move the impact point north of Svalbard and launch in late spring, we can land on the polar ice cap, which significantly increases the chances of recovery. The map in Fig. 21 shows a possible ground track for the future mission.

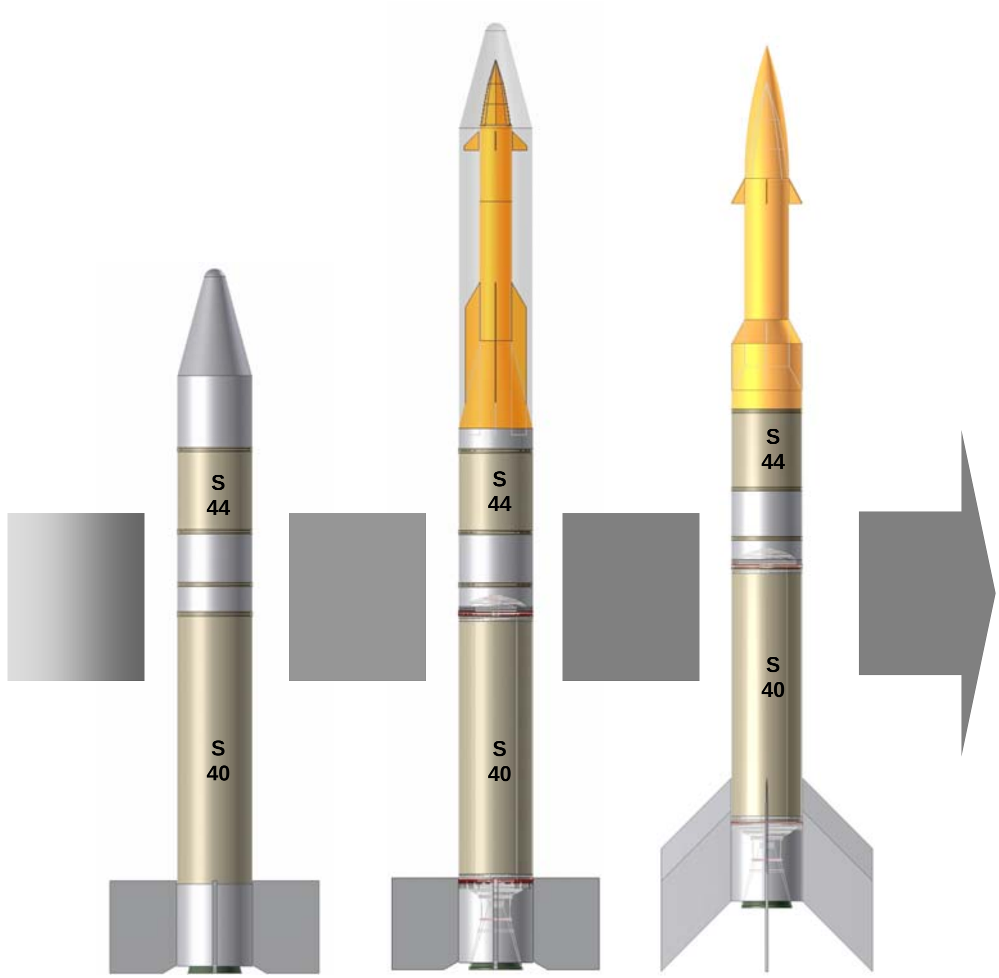


Figure 20. The Evolution Process of the VS-40 Vehicle for Hypersonic Research

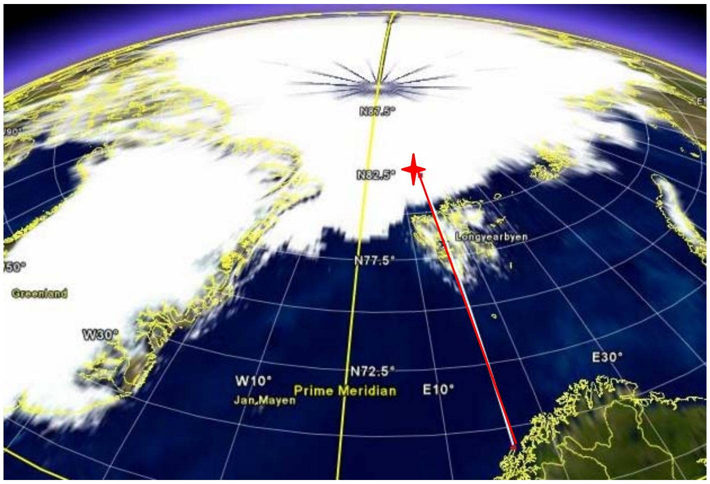


Figure 21. The Preliminary Ground Track

The preliminary trajectory calculations result in a down range in the order of 1200 km which requires that the re-pointing maneuver is not to a horizontal attitude, but in the order of 10 to 20 degrees elevation. The following figures display the preliminary flight performance. The maximum altitude for this scenario is approximately 180 km, the ground range is predicted as 1250 km. The experimental phase on the downleg from 100 km down to 20 km comprises 62 seconds of experimental time providing velocities in the region of 3470 m/sec and 2520 m/sec respectively and a Mach regime between 12.6 and 10.5. The re-entry flight path angle at 100 km is approximately 25 degrees. The current mission scenario as well as the predicted experiment phase performance are shown in Fig. 22 and Fig. 23 respectively.

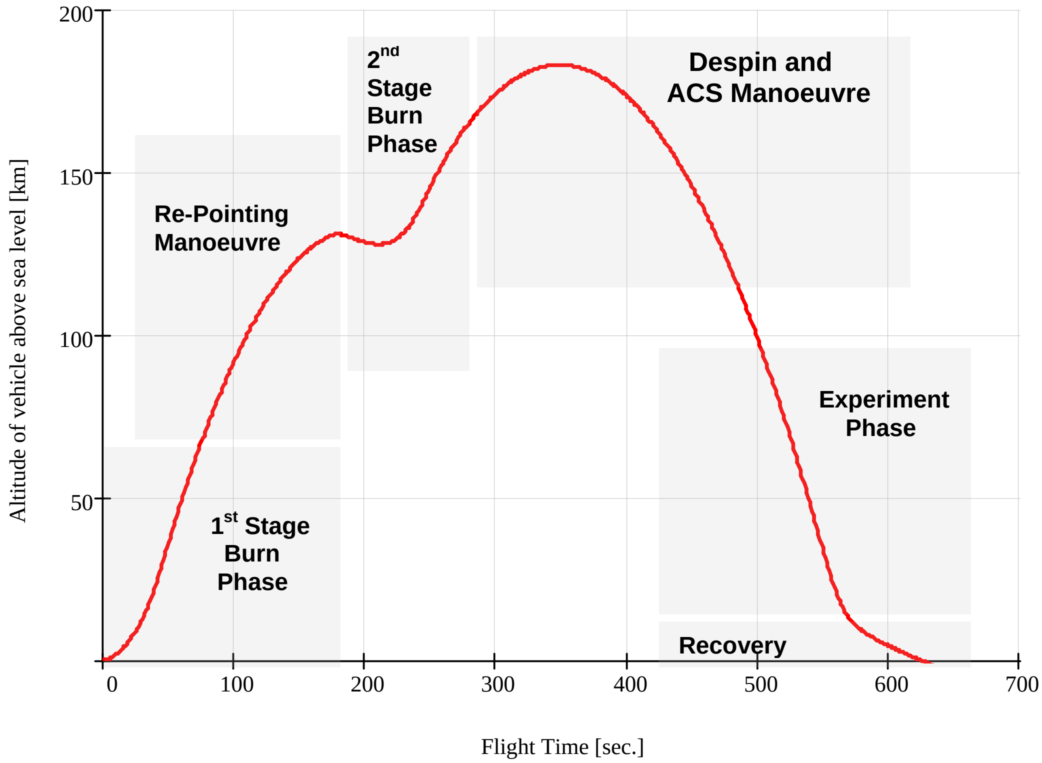


Figure 22. The Preliminary Mission Scenario

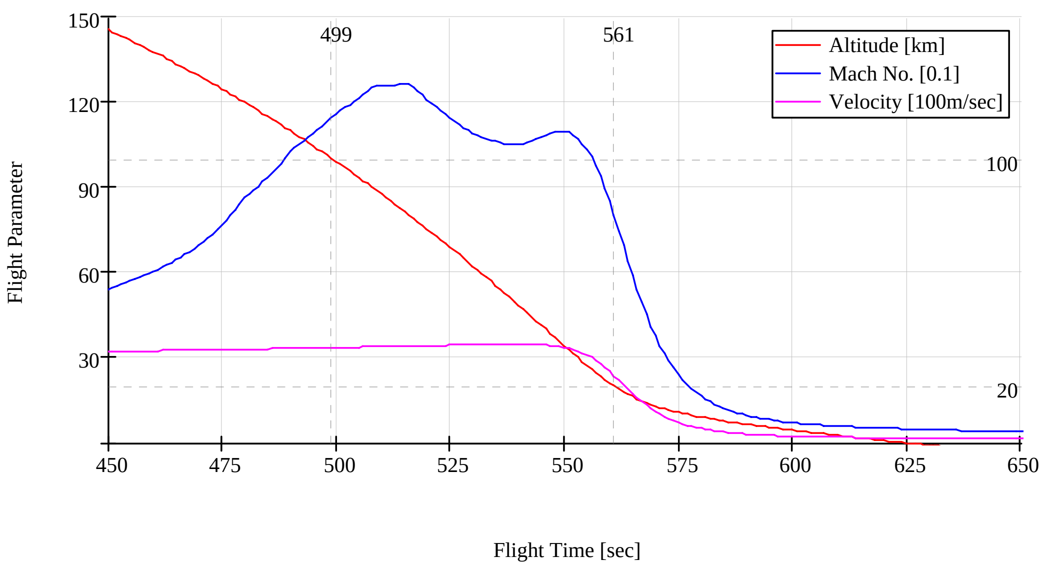


Figure 23. The Preliminary Re-entry Parameter

VIII. Conclusion

At the start of our involvement with the SHEFEX test vehicle four years ago, we assumed that apart from the obviously difficult recovery operation, the performance of a hypersonic experiment in the upper atmosphere at the end of a ballistic flight would pose no great problems. We have now obtained first hand experience of the construction of vehicles which from the standpoint of aerodynamics, structures, thermal protection, payload and ground support subsystems and attitude control, bear little resemblance to our usual sounding rockets for exo-atmospheric research. We now look forward to applying this experience in the next stages of the SHEFEX hypersonic re-entry research program.

IX. Acknowledgments

Apart from the number of colleagues at DLR Mobile Rocket Base and the other participating DLR institutes, who contributed to the design, development and successful launch of the SHEFEX vehicle and experiment, a special acknowledgement is due to the team at Inertial Science who produced the DMARS platforms and provided considerable support with the adaptation to the SHEFEX requirements. The colleagues from the Brazilian Aerospace Institute IAE provided campaign support during the vehicle preparation and launch. Our colleagues from SSC Esrange provided radar and telecommand support during campaign activities under the SSC/DLR EuroLaunch cooperation agreement.

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