An overview of Reusable Launch Vehicle Technology Demonstrator

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After the successful operationalization of Polar Satellite Launch Vehicle and Geo Synchronous Launch Vehicle, the Indian Space Research Organisation is in the process of developing Reusable Launch Vehicle technologies to achieve low-cost access to space. Towards this programme, a winged body configuration was conceived, which can fly at subsonic, supersonic and hypersonic Mach number regime, re-enter into the earth's atmosphere and simulate the landing manoeuvre. The aerodynamic design, analysis and wind-tunnel testing, aerothermal and structural design, analysis and testing were carried out. Suitable solid motor with slow burn rate propellant was developed. Mission design, guidance and control schemes were implemented. In order to meet the above objectives, certain technologies and infrastructure were developed. The entire subsystems were integrated and a large number of flight measurements were made in the maiden successful flight of Reusable Launch Vehicle Technology Demonstrator in May 2016. The flight measurements and flight performance indicated that the design philosophy, testing schemes and approaches followed are in order, thus providing confidence to proceed to the next logical step in the development of Reusable Launch Vehicle Technologies.

Keywords: Flush air data system, hypersonic flight test, Reusable Launch Vehicle Technology, winged-body configuration.

Introduction

BRINGING down the cost of access to space is a primary goal of space programmes around the world today. The current launch cost is US$ 20,000 per kg of payload, which has to be brought down by half through low-cost access to space. Typically, the on-board propellant accounts for about 85% of the mass and the rest is the structural hardware mass. The payload fraction ranges from 0.4% to 1% of the total mass depending upon the mission and efficiency of the system. Hardware accounts for about 80% of the launch cost, while the cost of the fuel is small. One of the concepts being perceived by many countries is the Reusable Launch Vehicles (RLVs), in which the hardware is planned to be recovered and reused for multiple launches; thus saving its cost[1].

Re-entry technology demonstrator missions have been discussed widely in the literature. ALFLEX, HYFLEX and OREX of JAXA[2], X-43A and X-51A of NASA[3] and IXV of ISA[4] are some examples. Many countries have embarked on RLV programmes since the 1980s. However, a viable technology that is fully reusable and brings down the launch cost is yet to emerge in the global launch market, though this may be changing with the advent of the recent innovative recovery approach and reuse of the booster by the Space Exploration Technologies Corporation (SpaceX).

The Indian Space Research Organisation (ISRO), has been pursuing studies on RLV technologies towards achieving a two-stage-to-orbit (TSTO) launch capability[5–7]. To increase the reliability and bring down the costs, one has to minimize the number of stages. Although single-stage-to-orbit (SSTO) can be the ultimate goal, it does not seem to be feasible with the available technology. TSTO vehicle can be totally expendable or totally/partially recoverable.

ISRO conceived a scale-down wing-body technology demonstrator to acquire and validate the hypersonic vehicle design process as a first step towards full-scale space plane. Hypersonic experiment Technology Demonstrator Vehicle (TDV) was successfully flight tested on 23 May 2016. It has given the RLV programme a big boost; the experimental flight was undertaken chiefly to test the hypersonic re-entry characteristics and capabilities. The Reusable Launch Vehicle Technology Demonstrator (RLV-TD) configuration in this mission is unique, because the double-delta wing body demonstrator was placed on top of the 11 m long, tailor-made slow-burn-rate solid booster. This configuration has not been attempted earlier in the world; similar winged-body flights were tested by the National Aeronautics and Space Administration (NASA) and the European Space Agency (ESA) accommodating the winged body inside a heat shield. The mission was also unique because it was designed to minimize the cost and get maximum data in a single mission. This configuration was known to have high instability, which was managed through the control system design. Four large movable fins in cruciform

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shape were used for ascent phase control. Elevon, rudder and reaction control system were used for the descent phase control. Even though hypersonic flight characterization was the prime objective of the mission, ISRO demonstrated both re-entry as well as landing manoeuvres of the vehicle. As many as 968 flight measurements were done on the demonstrator, which captured different phenomena in various flight regimes, providing enough data and confidence in developing a full-fledged winged RLV technology.

The following key aspects of the design were successfully demonstrated in the flight test: understanding the hypersonic aerothermodynamics of the double-delta winged body, the hypersonic re-entry, integrated autonomous navigation, guidance and control design, design of airframe structure, TPS design, hot structure design and flush air data system (FADS). In addition, comprehensive data generated in flight validated the design and development process and has provided enough confidence for ISRO to take the next leap in the arena of RLV. Necessary infrastructure and test facilities were augmented, extensive simulations were done prior to the flight; there were no anomalies in the entire flight regimes.

This article gives an overview of the vehicle configuration in the next section, followed by the aerodynamic characterization; mission strategy; details on navigation, guidance and control strategies; airframe structure design and testing methodology; aerothermal environment and management; FADS system; solid motor with slow burn rate propellant, and technology development is subsequent sections.

Vehicle configuration

Technology demonstrator vehicle (descent configuration)

Figure 1 shows the configuration of the RLV-TD demonstrator; it has a double-delta wing configuration with twin vertical tails; a wing span of 3.6 m, wing area of 6 sq. m and length of 6.5 m. The key challenges involved in the descent configuration design are as follows: (1) To meet the required lift-to-drag ratio at hypersonic to low subsonic Mach numbers with a wide range of angles of attack, to limit the deceleration levels and wing loading, and achieve sufficient down range as well as cross-range control. (2) Longitudinal, lateral and directional controllability using control surfaces (elevon and rudder) in the high dynamic pressure region. (3) To provide reasonable lift-to-drag ratio at low speeds to limit the sink rate before landing. (4) To minimize leading edge heating. (5) To minimize the centre of pressure movement with change in Mach number. (6) Sizing the fin during ascent and ensuring controllability in all the axes. (7) Flat fuselage surface in the windward side for placement of tiles and possible air-breathing engine integration in future. (8) Sufficient fuselage volume to accommodate reaction control system (RCS) thrusters, avionics packages, batteries, etc.

Aerodynamic configurations were evolved after a large number of iterations with different configurations. Detailed assessment of each configuration and aero-thermodynamic characterization was carried out using in-house developed engineering methods, computational fluid dynamics (CFD) tools and national wind tunnel facilities.

Blunt canted ogive nose cone was chosen for forebody for better longitudinal stability and reduced directional instability level in supersonic and hypersonic regime. The wing consists of a double-delta plan-form with 81°/45°

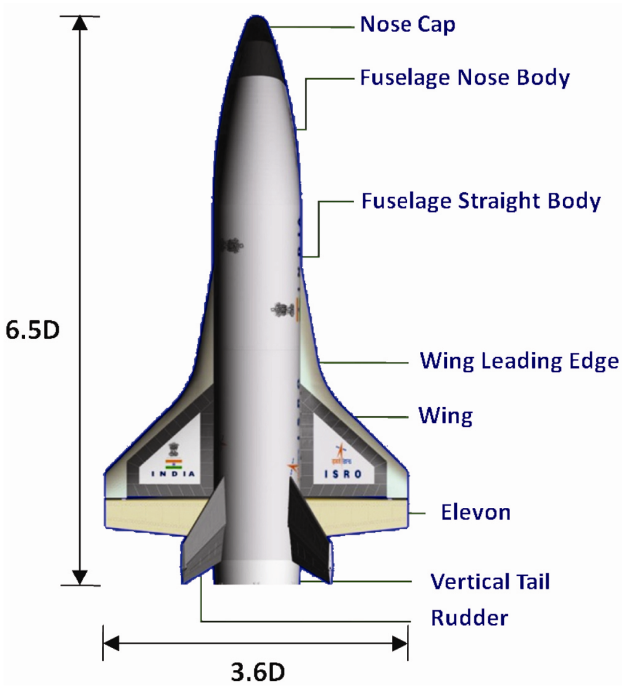


Figure 1. Configuration details of the technology demonstrator.

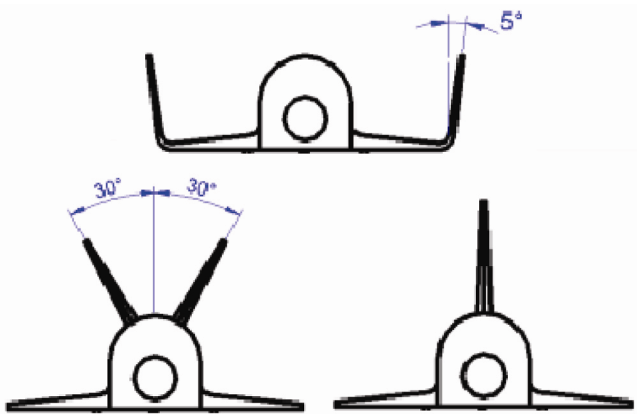


Figure 2. Various stabilizer configurations.

leading edge sweep back angle for minimum movement of aerodynamic centre and delayed shallow stall for entire operating Mach number regimes. The wing has aspect ratio of 2.16, 3° dihedral angle with wing span of 0.55 times the length. Aerofoil was reshaped from cambered-type to reflex-type to reduce the pitch down moment without sacrificing lift requirements. Pitch and roll controls were done through elevons, which combine the function of elevators and ailerons. Various directional stabilizer options, namely winglets, twin vertical tails and single vertical tail were studied (Figure 2). The single vertical tail was found to be directionally unstable at high angles of attack and winglets were not able to accommodate actuators; this increased the complexity of the structural interface design. Twin swept canted vertical tails with double-wedge airfoil section were selected to improve directional stability and controllability at high angles of attack. Rudder was designed to be used as a speed brake and for pitch-trim during supersonic speeds, in addition to the primary role of directional control.

Ascent vehicle configuration

Ascent configuration consists of the TDV and a solid booster, which are mounted in a serial manner. The vehicle height is 2.6 times its length (see Figure 3). There are four fins placed in *X*-configuration at the core base shroud to improve the longitudinal stability and controllability during ascent phase. Each fin is divided into metallic fixed and composite movable parts used for pitch, yaw and roll controls. This configuration is known to have very high instability, which resulted in a large number of design iterations. The configuration was improved after a series of wind tunnel tests for optimal longitudinal stability and to suit achievable actuator bandwidth.

Figure 4 shows a typical HEX-01 mission profile. In this mission, the TDV was boosted to hypersonic Mach number of 5 using solid booster of 9 tonne propellant loading. Spent stage booster was separated at an altitude of 45 km; the TDV performed coasting, followed by hypersonic flight experiment and controlled descent with predefined angle of attack profile as a function of Mach number. Closed loop guidance initiated at Mach number less than 2, guided the vehicle dynamically to reach the specified splash-down point after simulating the approach and landing phases.

Aerodynamic characterization

Extensive aerodynamic characterization for ascent vehicle and descent demonstrator vehicle was carried out using engineering methods, experiment and CFD tools. Engineering methods were used to size the wing, vertical tail, fin and its control surfaces. These methods were used to arrive at the trim capability of control surfaces, hinge moment coefficient and damping derivatives. National wind tunnel facilities were extensively used for RLV-TD characterization. These include 3 m × 2.25 m low subsonic speed continuous wind tunnel at Indian Institute of Technology, Kanpur (National Wind Tunnel Test Facility), 4.27 m × 2.74 m low-speed wind tunnel at Indian Institute of Science, Bengaluru, 1.2 m trisonic wind tunnel at National Aeronautical Limited, Bengaluru, and 0.25 m and 1 m hypersonic wind tunnel at VSSC, Thiruvananthapuram. Around 17 scaled-down models were fabricated and tested in the tunnel to measure the force and moment coefficients, hinge moment coefficients, unsteady pressure, damping derivatives, aeroelastic effects, RCS jet characteristics, stage separation and pressure on FADS port. Also, different flow visualization techniques were employed to understand the flow field on and around the vehicle. Ground wind response studies, aeroelastic and flutter studies were also carried out. Booster stage separation

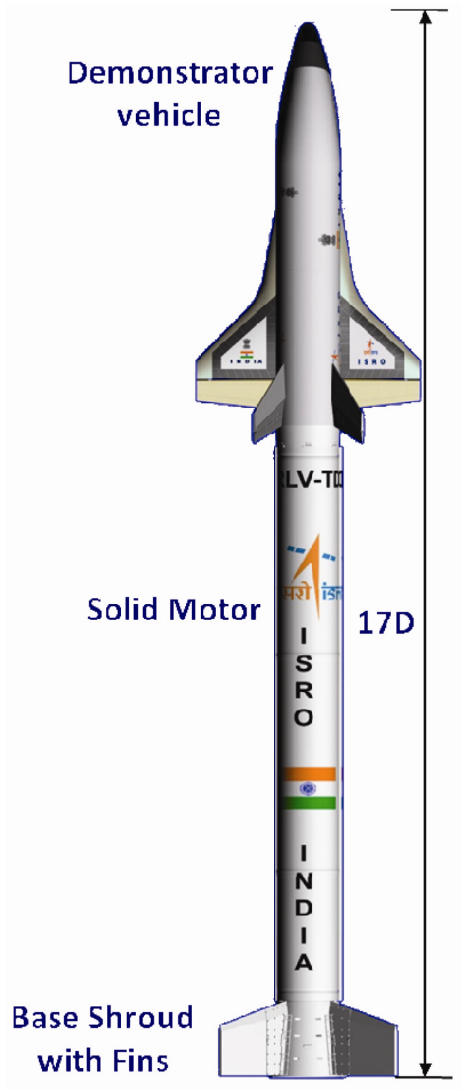


Figure 3. RLV-TD configuration details.

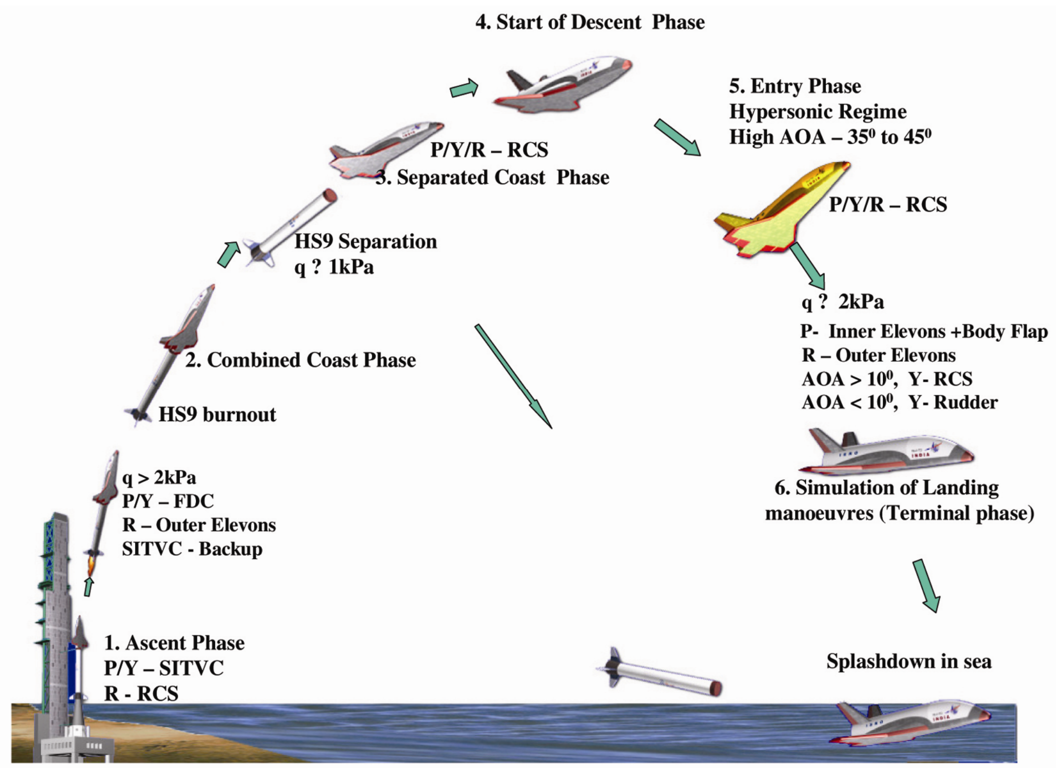


Figure 4. HEX-01 mission profiles.

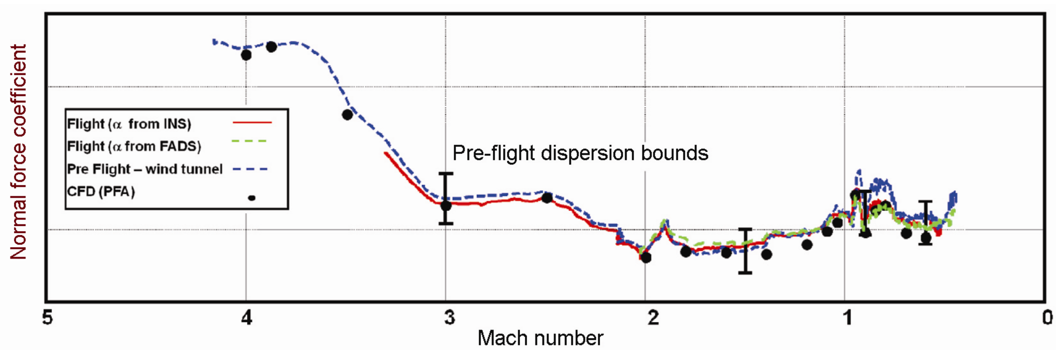


Figure 5. Flight-estimated normal force coefficient of Technology Demonstrator Vehicle compared with computational fluid dynamics and wind tunnel.

studies were carried out using wind tunnel using on space/time march approaches. RCS jet and free stream interactions were simulated in wind tunnel test, pressures were measured at critical regimes and oil flow visualizations obtained to validate the CFD tool for tunnel conditions and the tool used to estimate incremental coefficients for flight events. The total number of wind tunnel tests carried out was 4000 (blowdowns). The in-house PARAS CFD tool was used extensively for understanding the flow behaviour, various configuration design studies, generation of load distribution for structural design, incremental coefficients due to jet-on and simulating flight conditions and flight configurations. Different CFD tools such as CFD^{++}, UNS3D and SU2 were used to complement the above results and gain confidence. In Figure 5, flight-estimated normal force coefficient of TDV is compared with CFD and wind tunnel test results, which exhibits excellent match.

Mission strategy

In conventional launch vehicles, the gravity turn trajectory is achieved at near-zero angle of attack, because the vehicle being symmetric, the normal force acting on it is zero and hence the loads on the vehicle are minimal. However, for this configuration the normal force is not zero because of the cambered wing. It was decided to proceed with zero normal force steering during ascent phase. Open-loop altitude-based quaternion steering was adapted, and optimum pitch and yaw steering program for the ascent phase was computed in order to fly an optimum wind-biased gravity turn trajectory. Extensive simulation studies brought out the amount of excursion in angle of attack during the transonic regime resulting in excessive normal load on the vehicle due to nonlinear behaviour of the aerodynamic parameters, which was not captured well by the control system design. To circumvent this issue, the angle of attack profile during gravity turn was maintained as zero till Mach number of 2, which resulted in better performance of the control system, lower angle of attack excursion at transonic and hence lowered the structural load on the vehicle. During ascent phase, the velocity azimuth at booster separation was constrained to around 90 degrees to minimize the cross-range dispersions during descent phase. During ascent phase the movable fin was used for pitch, yaw and roll control and steering the vehicle in all three planes. At the end of the ascent phase, the TDV propelled by solid booster reached an altitude of 65 km with peak Mach number of around 5. Figure 6 shows the pre-flight and flight achieved trajectory parameters. It can be seen that the flight performance is similar to that of pre-flight prediction.

Descent phase mission started from the peak altitude of 65 km with re-entry Mach number of around 4 and ended at touchdown or splash down in sea. The maximum dynamic pressure of vehicle was limited to 25 kPa and load factor limit of 5 *g*. The allowable angle of attack corridor as function of Mach number was generated considering control capability, arriving at an optimum achievable centre of gravity of the vehicle, trim sharing logic between rudder and elevon for pitch plane and ensuring that the vehicle flew in stable attitude to the extent possible. Figure 7 shows the trim corridor and scheduled angle of attack as a function of Mach number during descent phase. Figure 8 shows the control surface deflection required to trim the vehicle for the schedule angle of

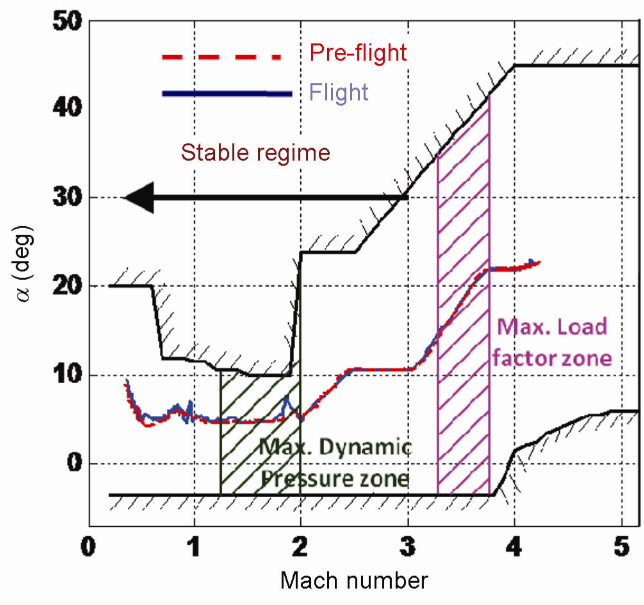


Figure 7. Trim corridor and scheduled angle of attack as a function of Mach number during descent phase.

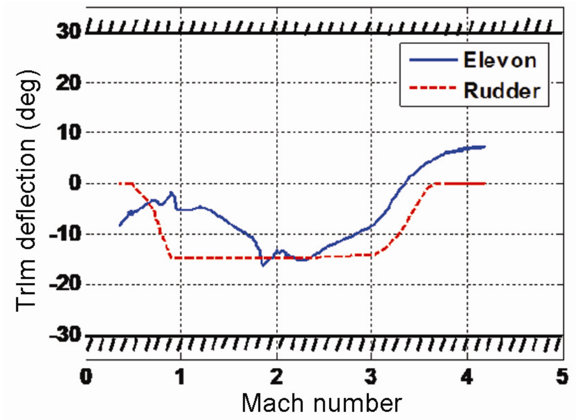


Figure 8. Control surface trim deflection during descent phase.

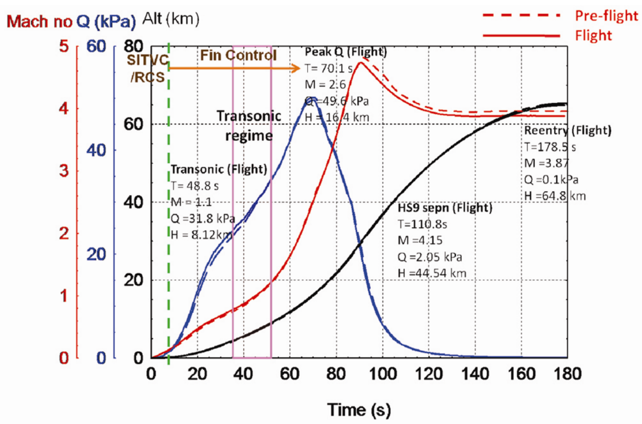


Figure 6. Comparison of pre-flight and flight performance during ascent phase.

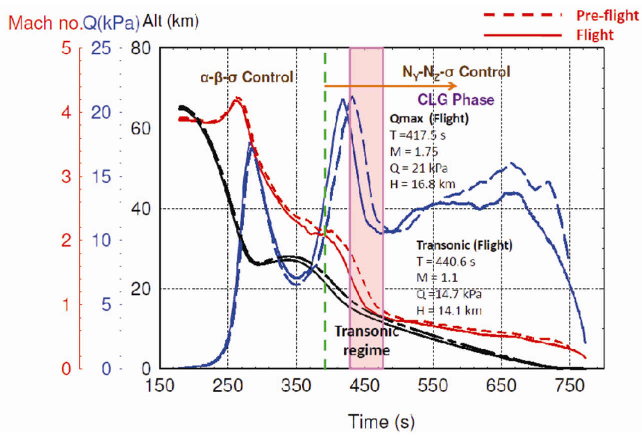


Figure 9. Comparison of pre-flight and flight performance during descent phase.

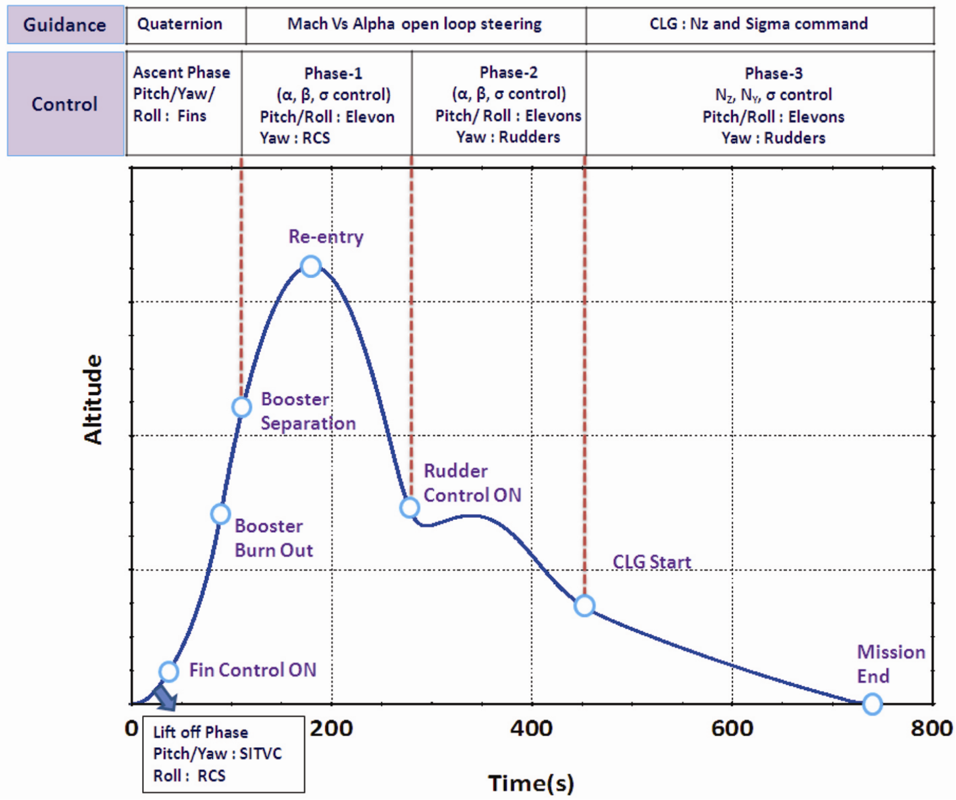


Figure 10. Guidance and control strategies.

attack. Figure 9 shows a comparison of pre-flight and flight performance during descent phase; and trajectory parameters are fairly close to the nominal pre-flight prediction.

Navigation, guidance and control strategies

Navigation

The configuration of the navigation system is decided by the stringent accuracy requirement during landing phase. As this requirement cannot be met by standalone inertial navigation system (INS), GPS-aided navigation system (GAINS) that provides position accuracy of 50 m, velocity accuracy of 0.5 m/s and altitude accuracy of 50 m was used. Radar altimeter (RA) enhances the vertical accuracy to 1 m during the terminal phase of the mission. Air state parameters such as angle of attack, angle of side slip, bank angle, dynamic pressure and Mach number were estimated by navigation from coasting phase till Mach number greater than 2 and measured wind was stored on-board to achieve the required accuracy. Ceramic Servo Accelerometer Package (CSAP) was used for measuring vehicle acceleration with a resolution of 25 μg in all three directions for autopilot control at Mach number less than 2. Figure 10 shows the navigation guidance and control strategy adapted in the flight.

Digital autopilot design

In ascent phase, attitude control in pitch/yaw is provided by secondary injection thrust vector control system. Vehicle rolling moment is controlled by RCS from its lift off until the vehicle dynamic pressures reaches 2 kPa. Fin deflection control system provides pitch, yaw and roll attitude control till booster separation. The main challenge in ascent phase DAP design is that the aerodynamic moment instability coefficient is 20 times more than that experienced in conventional launch vehicles, resulting in a time to double of 150 ms. Since the time to double is less, the control system has to act very fast and with a powerful actuator in lesser time. These challenges were overcome by a state-of-the-art control system design employing high rigid-body control bandwidth, actuator bandwidth and lower sampling time. Velocity-based gain scheduling rather than time-based gain scheduling as done in conventional launch vehicles, was adopted to tackle dynamic pressure variations.

In descent phase, the control philosophy from the TDV coasting phase to Mach number of 2 was based on air state parameters estimated by navigation. From Mach number less than 2 until splash down, the control philosophy was based on measured accelerations and bank angle from CSAP. Pitch control was achieved by symmetric deflection of elevons, roll control by asymmetric deflection of elevons and yaw control by rudder deflections. The challenges were trim scheduling for a wide range of Mach numbers and angles of attack and two-dimensional gain scheduling based on Mach number and dynamic pressure. Robust control design methods such as Aileron-Rudder interconnect-based gain scheduling, α, β, σ variables were used as commanded variables for higher supersonic and hypersonic regimes. Lateral accelerations were used as command and feedback variables for Mach number less than 2, while 2D based gain scheduling as a function of Mach number and dynamic pressure were implemented to improve the robustness beyond 3σ perturbations.

Guidance scheme

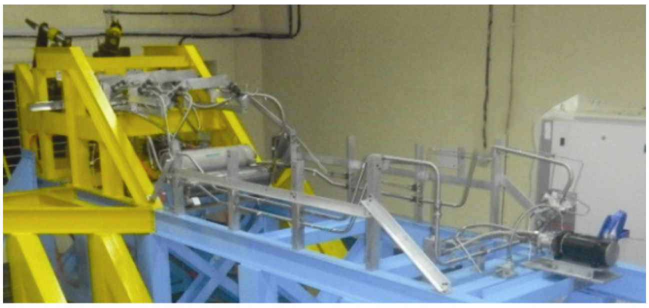
Guidance scheme adopted from lift-off to re-entry was similar to a conventional launch vehicle and an altitude-based open-loop steering was implemented during powered phase. In re-entry phase, the angle of attack was commanded using pre-defined Mach number versus angle of attack profile stored on-board. Below Mach number 2, the vehicle was commanded with normal acceleration commands from the table stored on-board. The manoeuvres in this phase were rate limited to a pre-defined Mach number-based rate limit table, which contains the rate limits for angle of attack, side slip angle, bank angle and normal acceleration commands. In approach and landing phase, based on the energy available, on-board computation done and the closed loop guidance commanded the normal acceleration and bank angle to steer the vehicle to a desired landing point. This included an on-board trajectory design and a path controller to track the planned trajectory. The vertical dynamics was controlled by commanding normal acceleration (N_{ZC}) and the horizontal dynamics was controlled by bank angle (σ_C) command. The guidance problem was formulated assuming that the vertical dynamics is decoupled from the horizontal dynamics. Other features included re-target selection of landing point based on the range to be covered and safe-mode guidance to ensure vehicle safety.

Iron bird test facility for validation

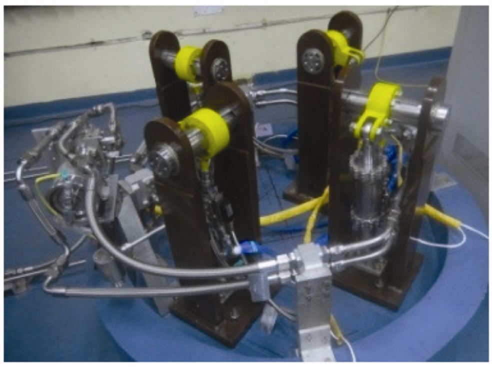
Iron Bird test facility, on-board hydraulic actuation system laid out as in-flight from battery to actuation system (Figure 11), was commissioned to validate integrated NGC software/packages with flight actuators, controlling the simulated aero control surfaces and associated hydraulics system like Li-ion battery-powered BLDC motor, plumbing, etc. The closed-loop tests were performed with QM and FM packages, and validated performance of actuation systems with RF, ramp and step commands. The test segment not only reduced the development risk by testing the fully integrated systems, but also provided a platform for the exhaustive evaluation of the NGC system and subsystem to the maximum possible extent. Extensive simulations brought out the following corrections to enhance the robustness of the NGC systems: (1) modifications of control electronics; (2) replacement of LVDT wire hook joint with splice joint; (3) yaw–roll phase margin improved during descent phase, and (4) roll gain margin during ascent phase. Totally 130 simulations were carried out.

Airframe structure design and testing methodology

The RLV-TD is a hybrid vehicle, which blends the technologies of aircraft and launch vehicles. The TDV is a winged body similar to an aircraft. The booster stage is a typical solid stage of a launch vehicle. Several modern as well as conventional technologies have been adopted for the design of the structural elements of the vehicle. The structural elements can be broadly classified into two types, viz. cold and hot structures. The cold structures operate under mere ambient temperature conditions, whereas the hot structures have to withstand higher thermal loads over and above the flight loads. Both hot and cold structures are brought down to the maximum allowable operating temperatures of their respective materials using suitable thermal protection systems. The vehicle was



ALS configuration for TDV



ALS configuration for base shroud

Figure 11. Iron bird test facility for validation of Navigation Guidance and Control system.

designed for aerodynamic pressure loads at transonic speeds, maximum dynamic pressure and longitudinal acceleration during ascent phase. In addition to the above loads, the demonstrator was also designed for maximum normal and lateral accelerations. Limit and ultimate load was generated by augmenting structural load. AA-2014 material was selected for airframe and wings so that fabrication and assembly was easier. Airframe was designed for ultimate loads. The fuselage was fabricated with skin panels having integral stringers and *D*-shaped bulk heads. Wing was fabricated as multi-cell box-type structure with spars, ribs and skin panels with integrally machined stringers.

In order to validate the design methodology, extensive structural tests were carried out, namely critical joints tests, panel level test and component level tests. As a final confirmation, integrated airframe test was carried out by assembling the fuselage with the inter-stage in cantilever mode, subjecting the panels to aerodynamic pressure distribution and applying inertia force and shear loads on the wings and fuselage using whiffle tree mechanism so as to reduce the number of loading jacks (Figure 12). Flight airframe with all the subsystems was qualified for an overall acoustic level of 155 dB in an 1100 m^3 acoustic chamber. Measured strains from the flight at critical locations revealed that the overall design and testing methodologies were satisfactory.

Aerothermal environment and management

The demonstrator experiences maximum specific energy of 1.8 MJ/kg, which is benign compared to the orbital re-entry vehicle, an order of 36 MJ/kg. Hence, the demonstrator experiences significantly lesser thermal environment in the HEX-01 mission compared to the orbital re-entry vehicle. Even for this mission, airframe material exceeds the allowable temperature limit of AA-2014. Hence structures should be restricted to limit the thermal load by protecting them with an appropriate thermal protection system (TPS). Waterproofed silica tile and flexible insulation were designed to protect the windward and leeward regions. Hot structure systems and materials were identified for the components which were likely to experience higher thermal and pressure loads, viz. nose cap, wing leading edges, vertical tail and control surfaces. Figure 13 shows the thermal environment and materials used for various regions of the TDV.

In order to validate the hot structure design, structures were qualified by simulating combined effect of structural and thermal load for the entire flight duration at the thermo-structural test facility. Figure 14 shows a typical thermo-structural test set-up for carbon/carbon nose cap.

TPS were successfully qualified for various tests, namely water repellency, kinetic heating simulation for flight duration, shock and vibration levels, thermo-vacuum condition and simulation of shear flow under thermal environment. Temperature measurements on hot structures and TPS from flight indicated that design of hot structures methodology was satisfactory.

Flush air data sensing system

FADS forms a mission critical subsystem in future re-entry vehicles. This system was not used in the active loop during flight. FADS makes use of surface pressure measurements from the nose cap of the vehicle for deriving the air data parameters such as angle of attack, angle of side slip, Mach number, etc. These parameters find use in the flight control and guidance systems, and also assist in the overall mission management. The system functioned from Mach number of less than 2 at a corresponding altitude of 20 km, where the accuracy of navigation-based estimation of air states parameters was reduced due to predominant role of wind speeds. To perform this

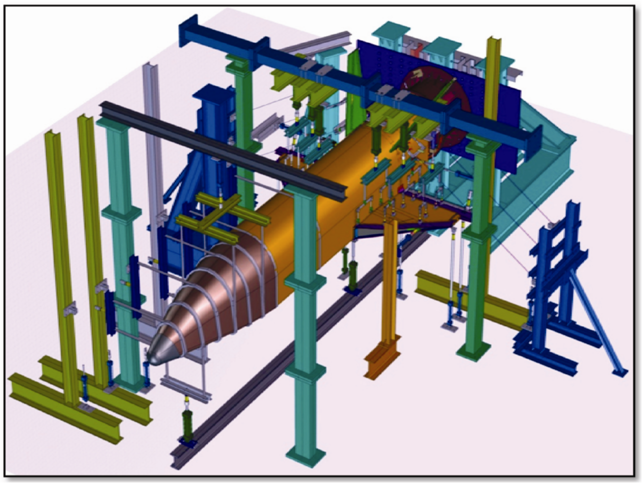


Figure 12. Schematic illustration of airframe integrated test.

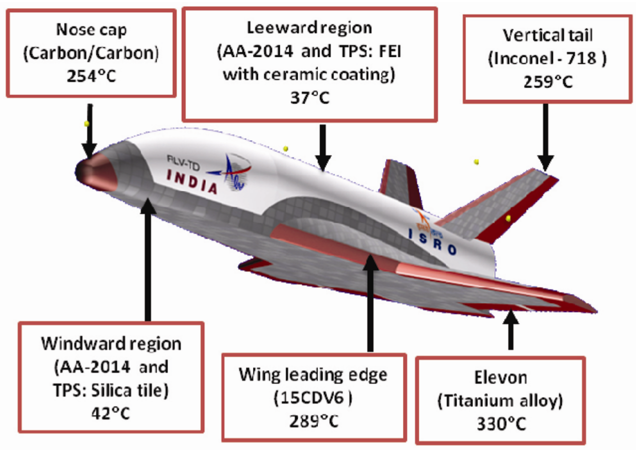


Figure 13. Thermal environment and selection of materials for various regions.

estimation, air data states must be related to the surface pressure on ports located on the nose cap. The aerodynamic model is postulated as a compromise between a simple potential flow and modified Newtonian flow theory for blunt objects in hypersonic flow. The end-to-end system was qualified in this flight with the accuracy of angles within 1° and Mach number within ± 0.1. The main design considerations like criteria for selection of pressure ports, pneumatic tubing, pressure sensors and algorithm were validated in flight.

Development of solid motor with slow burn rate

The TDV has to be delivered with a maximum Mach number of 5. The available solid propellant motor at ISRO has a burn rate of 5 mm/s with an action time of 50 sec, which leads to ascent phase dynamic pressure of nearly 150 kPa; this demands an unrealistic structure and control system, which is practically not feasible. Hence, an optimum and achievable thrust profile was conceived with constraints of maximum thrust limit to 250 kN and an action time of 90 sec, which lead to the permissible dynamic pressure. The above thrust profile was achieved by developing a slow burn rate solid propellant and grain design of motor. Solid propellant formulation contains hydroxyl terminated polybutadiene (HTPB) as a binder, ammonium perchlorate (AP) as oxidizer and aluminum powder as fuel without burning rate catalyst. Slow burning rate is achieved by coarse AP particles at its highest proportion (4 : 1) against the fine AP; oxamide was used as burning rate retardant. Head-end and middle-end segments were provided with cylindrical ports, while the nozzle end segment had deep-slotted star port grain configuration were conceived to increase the initial phase of thrust of the motor. The cylindrical shell of the motor was made of 15CDV6 with shell thickness of 2 mm. Static test of the newly developed motor was conducted and subsequently grain design of motor and igniter design were modified. Second static test of the motor with the above modifications was conducted. The results indicated that the design was in order, based on which the flight motor was casted. Flight exhibited that the performance of the motor was close to nominal prediction.

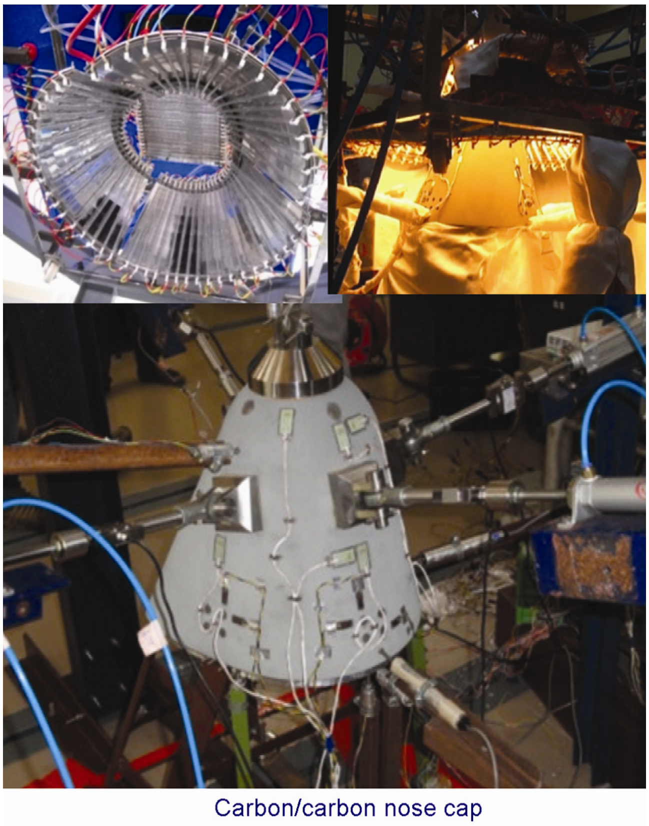


Figure 14. Thermo-structural test of carbon/carbon nose cap.

Technology development

Super alloys development

Elevon component of TDV demanded a material of high specific strength and good mechanical properties at elevated temperatures; hence titanium alloy (Ti4Al6V) was chosen. Though Ti alloy forging process is well-established, realization of large-sized (80 × 1400 mm) elevon from Ti alloy has not been attempted earlier in the country. There are challenges is maintaining chemical homogeneity, mechanical properties and microstructure for realization of large size. Hence, new thermo-mechanical processing cycles were devised to handle the above challenges. In the present work, 830 mm diameter ingots for forging were made through double vacuum arc remelting (VAR) process using pure raw materials, subjected to conditioning so as to remove the oxidized surface, preheated just above the β-transus temperature and forged to 650 mm diameter bars as billet stock. Forged stock was again forged in a 1500T press below the β-transus at a slow rate involving suitable upsetting/drawing operations followed by suitable heat treatment. The above process plan was specifically made to meet the requirements of microstructure, mechanical properties and ultrasonic quality of large-sized forging. Figure 15 shows the large-sized elevon realized from Ti alloy.

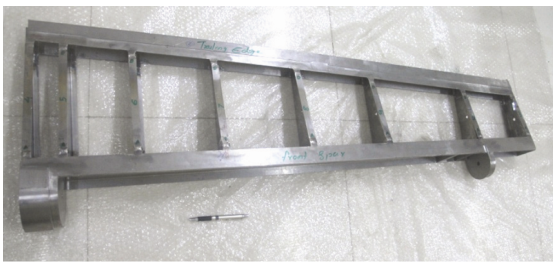


Figure 15. Elevon torsion box made of titanium alloy.

Vertical tail leading edge experiences very high thermal and pressure load; hence inconel 718 alloy was selected as a hot structure material. Development of large size of 1.42 m forging process was carried out in three stages, namely upset forging, drawing down and finish forging. The solution annealing as well as ageing were fine-tuned to meet the austenitic fine-grained structure and mechanical properties. The realized forgings were further fabricated into vertical tail leading edge component.

Ceramic coating

Flexible external insulation (FEI) is identified as a TPS candidate for the leeward region of TDV, with highly curved surfaces and lower heat load. FEI configuration has silica cloth layer on either side with cerablanket felt sandwiched and stitched using quartz thread, which leads to uneveness and waviness on the external surfaces. Ceramic coating on FEI is needed to enhance the emissivity and reduce solar absorptivity. This coating acts as a thermal barrier and minimizes the unevenness in the external surface. Silica binder, silica powder with size of 40 μm and ceramic balls were added and the mix was milled. This slurry was charged into a spray gun and sprayed over FEI on the TDV under room conditions. Flight results indicated that the ceramic coating performance was satisfactory.

Summary

This article provides an overview paper of the hypersonic flight experiment of wing-body technology demonstrator. Towards low cost access to space, ISRO conceived a wing-body RLV. The aerodynamics, aerothermal, FADS, hot structure, NGC system and associated subsystems were designed, developed, integrated for a successful mission. From the flight measurements valuable lessons were learnt; data indicate that the approach followed, design margins, etc. are satisfactory.

Subsequent to the above mission, it is proposed to carryout a TDV as an under slung mass in a helicopter to an altitude of 3.5 km and drop it to demonstrate the autonomous landing capability. During the unpowered descent phase, the vehicle will accelerate and start picking up velocity. On-board guidance will be initiated and the vehicle will be made to align with the pre-defined approach and landing glide slope, and followed by a long flare, touchdown on the runway and roll out to standstill. The next step is to demonstrate orbital re-entry mission of the wing-body vehicle.

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doi: 10.18520/cs/v114/i01/38-47