Hypersonic Flight Experiment ReFEx: Status and Future Perspectives

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The hypersonic flight experiment ReFEx (Reusability Flight Experiment) under development at the German Aerospace Center (DLR) passed the preliminary design review in 2019 and several subsystem CDRs are to be completed this year.

The paper summarizes the latest status of major characteristics of ReFEx, the planned reentry corridor, flight and range safety considerations and assessments on the vehicle's controllability. Examples of the numerical and experimental aerodynamic configuration analyses are shown.

The second part of the paper investigates potential next demonstration steps for winged RLV. Intermediate steps on larger and more powerful launch systems are evaluated with the aim of having a liquid-rocket powered demonstrator stage with multiple-flight reusability ready before the end of this decade. A preliminary version of the technology development roadmap will be presented.

Keywords: hypersonic flight demonstrator, ReFEx, RLV

Subscripts, Abbreviations

AIV	Assembly, Integration, Verification
AOA	Angle of Attack
CAD	Computer Aided Design
CFD	Computational Fluid Dynamics
GLOW	Gross Lift-Off Mass
GNC	Guidance Navigation and Control
IR	Infra-Red
LEO	Low Earth Orbit
LFBB	Liquid Fly-Back Booster
LH2	Liquid Hydrogen
LOX	Liquid Oxygen
MECO	Main Engine Cut Off
RCS	Reaction Control System
RLV	Reusable Launch Vehicle
RP-1	Rocket Propellant (Kerosene)
SSO	Sun Synchronous Orbit
Ti	Titanium
TPS	Thermal Protection System
TRL	Technology Readiness Level
TSTO	Two-Stage-To-Orbit
VO	Virgin Orbit

1 INTRODUCTION

The hypersonic flight experiment ReFEx (Reusability Flight Experiment) is currently under development at the German Aerospace Center DLR. The hypersonic demonstrator is to be launched on a VSB-30 sounding rocket using Brazilian solid motors (Figure 1) with the flight being scheduled for early 2023. The main goals of the ReFEx-project are the demonstration of a controlled autonomous re-entry flight from hypersonic down to subsonic velocity, spanning the typical range of winged RLV-booster stages. Several key technologies required for future reusable winged first stage systems are to be tested in flight [1].

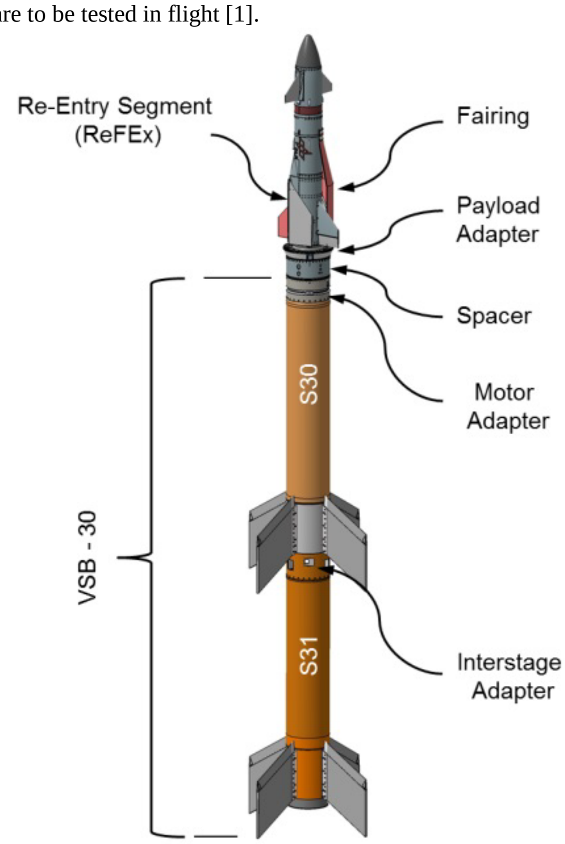

Figure 1: CAD Model of the ReFEx-launch configuration

2 REFEX-CHARACTERISTICS AND DEVELOPMENT STATUS

2.1 Mission Goals

The main mission goals of the ReFEx project are [1, 2]:

- Perform a controlled flight following a re-entry trajectory representative for a winged RLV first stage in the speed range from hypersonic down to subsonic
- Perform a controlled heading change (capability required for future RLV autonomous return to the launch site)
- Flight Test of the autonomous Guidance Navigation and Control (GNC) system
- Perform In Flight Data acquisition using advanced sensors
- Recovery of the Re-Entry Segment for Post Flight Analysis (PFA)

2.2 Configuration lay-out

Figure 1 shows the ReFEx Launch Configuration. The VSB-30 rocket has no thrust vector control capabilities (passively stabilized system). Therefore, the payload segment is required to have an almost rotationally-symmetric shape to enable a safe launch. Hence, part of the Re-Entry Segment (ReFEx) is covered by a fairing during the launch phase. However, the Re-Entry Segment needs to have a typical winged aerodynamic shape for the experimental phase (Figure 2) which is contradicting the launcher symmetry requirement. To meet both requirements the wings of the experimental vehicle were designed foldable and are covered by a fairing for the launch phase [1, 2].

Figure 2 shows the current design status of the Re-Entry Segment in its re-entry configuration with wings unfolded.

The experimental hypersonic vehicle has a length of 2.7 m, a wingspan of 1.1 m, and a mass of approximately 400 kg.

The fuselage is accommodating all subsystems in a densely integrated arrangement [1, 2] (Figure 3). The integrated units are grouped to the following subsystems:

- Guidance Navigation and Control (GNC)
- Avionics (AVS)
- Structure (STR)
- Flight Instrumentation (FIN)

The subsystem definitions are progressing and many of the designs can be frozen and their manufacturing and assembly has started. A more detailed overview of the subsystems can be found in references 1 and 2.

2.3 Mission Architecture

The experimental vehicle shall perform a controlled re-entry flight similar to that of full-scale winged reusable first stages. The flight profile shown in Figure 4 requires a large ground area for testing. The newly opened Koonibba Test Range (KTR) by Southern Launch in conjunction with the Woomera protected area in Australia offer a sufficiently large ground area and were therefore selected as a favored test site for the ReFEx mission.

Figure 4 shows schematically the sequence of the ReFEx mission with preliminary details regarding time and altitudes. For the Launch Phase all actuators which could potentially have an unintended influence are locked. After the ignition of the first stage (S31) the ReFEx Launch Configuration will perform a Lift-Off. The first stage (S31) burns out and is separated at 12 s after ignition. About 20 s after launch, the second stage (S30) will be ignited. The S30 burn-out is at 49 s after launch. During the Launch Phase, the VSB-30 will build up a spin rate to compensate thrust vector inaccuracies. Therefore, the spin rate of the payload shall be reduced afterwards using a Yo-Yo de-spin system at 79 s. The fairing separation occurs at 84 s after launch followed by wing deployment and separation (SEP) from the second stage.

Figure 2: CAD Model of the ReFEX Re-Entry Segment

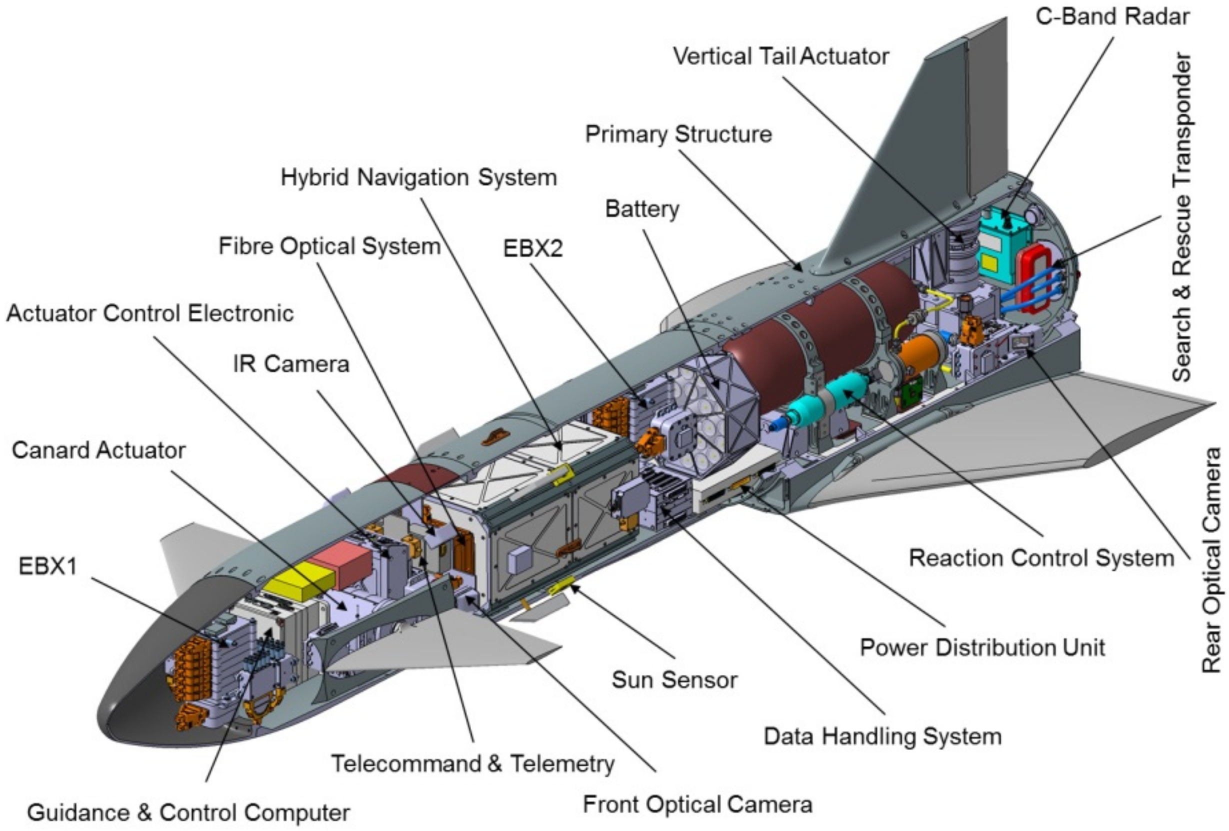

Figure 3: Internal systems accommodation of the Re-Entry Segment

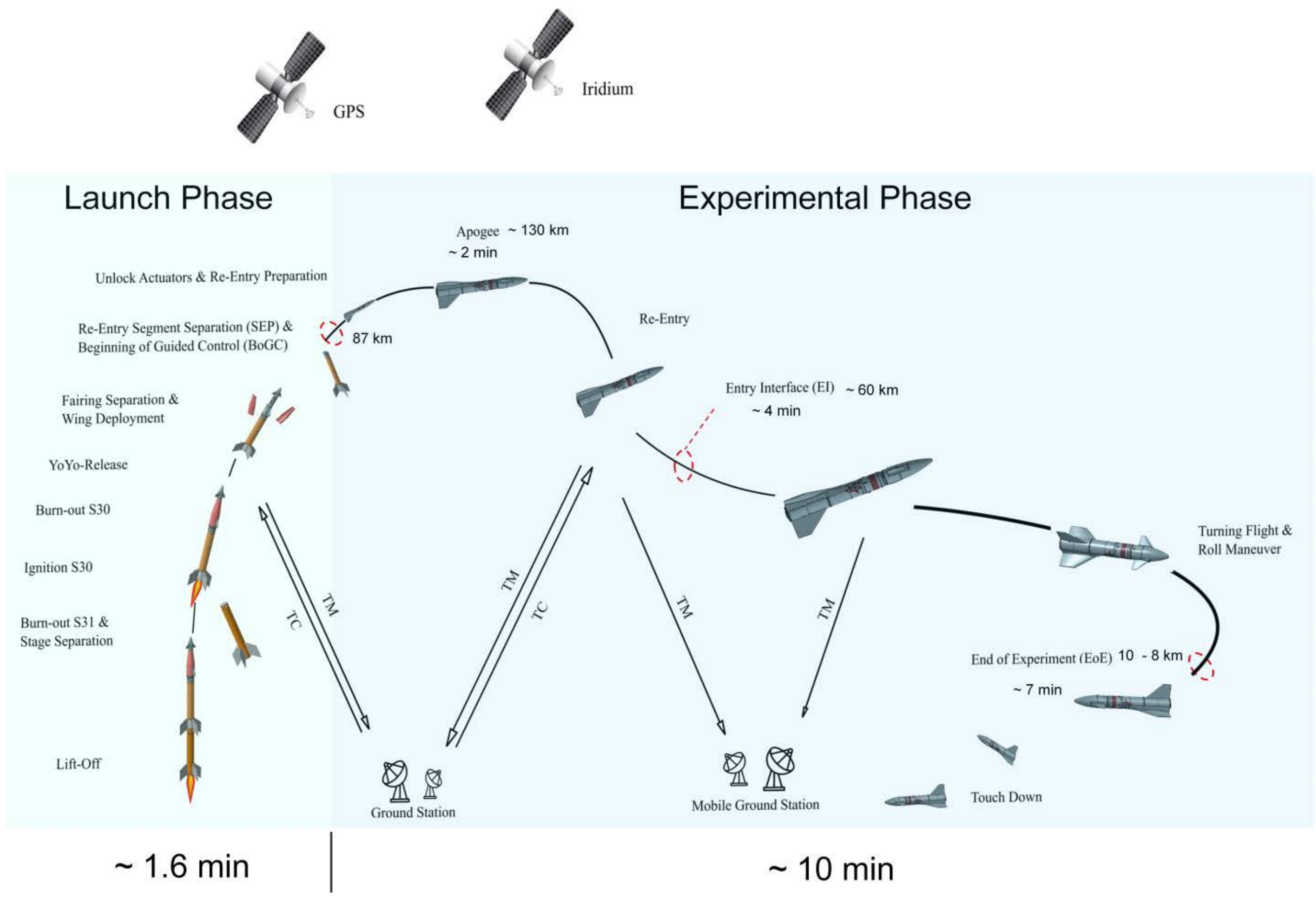

Figure 4: ReFEx Mission Architecture & Major Flight Events

SEP is the start of the Experimental Phase and the Beginning of Guided Control (BoGC) of the vehicle. After separation of the Re-Entry Segment, the control elements (canards, rudder, Reaction Control System (RCS)) will be unlocked. The cold-gas RCS reduces the remaining angular velocity of the Re-Entry Segment and performs required adjustments in the orientation for proper Entry Interface (EI) alignment.

The main parts of the flight demonstration are described in more detail in the following sections 2.4 and 2.5. At an altitude of 8 to 10 km the vehicle has reached the subsonic flight regime and enters a dispersion ellipsoid which defines the End of Experiment (EoE). To reduce the kinetic energy of the Re-Entry Segment prior to touch down a flare maneuver is envisaged at an altitude of approx. 120 m over ground. A controlled landing on ground is *not* part of the experiment [1, 2].

2.4 Aerodynamic analyses

The aerodynamic design of ReFEx derives from the DLR LFBB study [7, 8]. CFD simulations were used to further develop this shape. After several design iterations [9] including flight mechanical analysis [10], an aero shape was found that met all requirements imposed by the launcher and had an aerodynamically stable flight corridor (see Figure 5). Grey areas are statically stable and trimmable flight conditions. The black line depicts the planned flight path, including the roll maneuver at Mach number of 1.5.

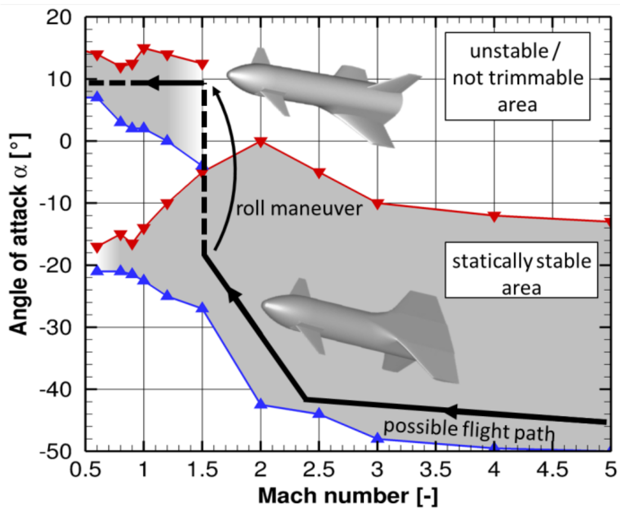

Figure 5: Intended flight envelope of ReFEx from hypersonics down to subsonics

After passing the EI, the Re-Entry Segment onboard control slowly transitions from RCS to aerodynamic control surfaces (canards, rudder). The two systems operate in conjunction until about 50 km at which point all nitrogen within the RCS tanks is consumed. From this point onward, the vehicle is controlled by the aerodynamic surfaces only.

According to CFD analyses, the re-entry vehicle does not show sufficient natural longitudinal and lateral stability in belly-down configuration for high Mach numbers (approx. Ma > 2.5). In order to avoid these unstable flight regimes, the vehicle enters the EI and the Reusable Launch Vehicle (RLV) corridor in belly-up orientation (vertical tail pointing downwards, see Figure 4). In the Mach range of 2 - 1.5 the Re-Entry Segment performs a roll maneuver. After roll, the vehicle remains in belly-down orientation for the remainder of the mission. This maneuver is necessary, because the control effectiveness of the canards in belly-up orientation decreases significantly with decreasing Mach number. Therefore, the re-entry vehicle becomes increasingly unstable and is no longer controllable. The aforementioned roll maneuver ensures sufficient natural stability and controllability of the vehicle throughout the mission. The calculated flow interactions in the high subsonic regime are shown in Figure 6.

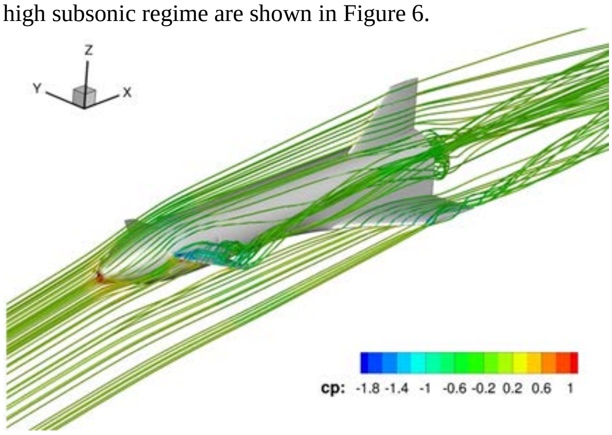

Figure 6: Flow-field interactions of ReFEx at Mach 0.9 after roll maneuver

Typically, for RLV with high Mach number reentries, high (negative) angles of attack can be achieved to decelerate ReFEx as early as possible to minimize thermal and mechanical loads. Given the VSB-30 launcher restrictions, the chosen way was starting the re-entry upside-down to ensure directional stability in this regime. As the Mach number decreases, the controllable AoA range drops and eventually controllability is lost due to flow separation. Therefore, a roll maneuver is conducted and the low supersonic velocities as well as the transonic regime are flown in nominal orientation at moderate (positive) angles of attack (Figure 6).

NB: the chosen reentry strategy of ReFEx does *not* imply that the same strategy is intended for the next generation of demonstrators (see section 3) or future European RLV. In a larger vehicle, in different launch constraints or in an operational large scale RLV mission such an upside-down maneuver could easily be avoided through different measures which, unfortunately, are incompatible with the current ReFEx due to its size limitations.

An experimental aerodynamic campaign of ReFEx is performed by windtunnel tests. A model is shown in Figure 7 when mounted in the DLR transonic tunnel TMK.

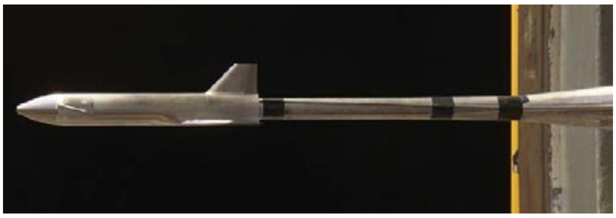

Figure 7: Model of ReFEx in DLR's windtunnel TMK in Cologne

2.5 Planned reentry corridor, flight guidance and range safety

A mission requirement for ReFEx is the demonstration of the successful flight with a re-entry trajectory representative for full-scale winged RLVs. Therefore, such a representative flight trajectory first has to be defined. The re-entry and return flight of any reusable first stage of a launch vehicle features similar events and characteristics. First, following MECO of the RLV's engines, the stage is separated from the core stage/-second stage and follows a ballistic trajectory. Normally, during this ballistic flight phase negligible aerodynamic forces are present due to the high altitude at separation. The stage travels through apogee at suborbital velocity and begins falling back to Earth while gaining velocity. During this phase the aerodynamic forces abruptly build up once the vehicle enters the denser parts of the atmosphere. Typically, the AoA in this phase is as high as possible close to 40° - 45°. At a certain point the stage experiences sufficient aerodynamic forces to start using the aerodynamic control surfaces to control the vehicle. Furthermore, the wing and fuselage generate sufficient lift and drag to slow down the vehicle while maintaining a desired altitude profile to ensure that the aerothermal loads are not exceeding structural limits. Following this phase, where the major part of the deceleration occurs, the re-entry vehicle transitions from supersonic to subsonic velocity and continues its flight as subsonic glider.

Due to the aforementioned reasons, the flight profiles of all winged RLV stages are somehow similar. This allows deriving certain thresholds and boundaries which are valid for any RLV stage. Hence, based on former research at DLR on the LFBB (Y-9) [8] and sub-scaled LFBB launchers (C60) [13], the SpaceLiner concept [14], the French EVEREST concept and RLV-concepts of the ENTRAIN study [4] a so-called *re-entry corridor* was defined by altitudes-Mach number dependent boundaries as shown in Figure 8.

The boundaries of the re-entry corridor (dotted lines in Figure 8) represent two physical constraints to any re-entry flight vehicle. The upper boundary should not be overflown significantly, since the thus induced atmospheric "skipping" (compare simulated C60 trajectory which was not sufficiently adapted) subsequently leads to high peaks of heat flux and is of disadvantage for any thermal protection system. A flight path significantly below the corridor's lower boundary leads to excessive aerodynamic loads, since dynamic pressure and heat flux increase exponentially in the denser part of the atmosphere. Those physical boundaries are shown in Figure 8 as iso-lines for heat flux and dynamic pressure.

In order to fulfil the mission goal of performing a heading change of the vehicle, a prescribed landing site has to be reached within certain accuracy. Since the flight experiment is launched with the unguided VSB-30 booster, the state variables, like altitude, velocity etc. at payload separation, are much higher compared to a guided orbital launch vehicle. This calls for a guidance strategy that is capable of dealing with such strong dispersions while fulfilling the aforementioned mission goals. During the coasting the re-entry trajectory is adjusted to consider the state error after separation. A numeric predictor-corrector method is used in a Newton search to update the angle of attack and bank angle profiles such that the desired terminal conditions are achieved.

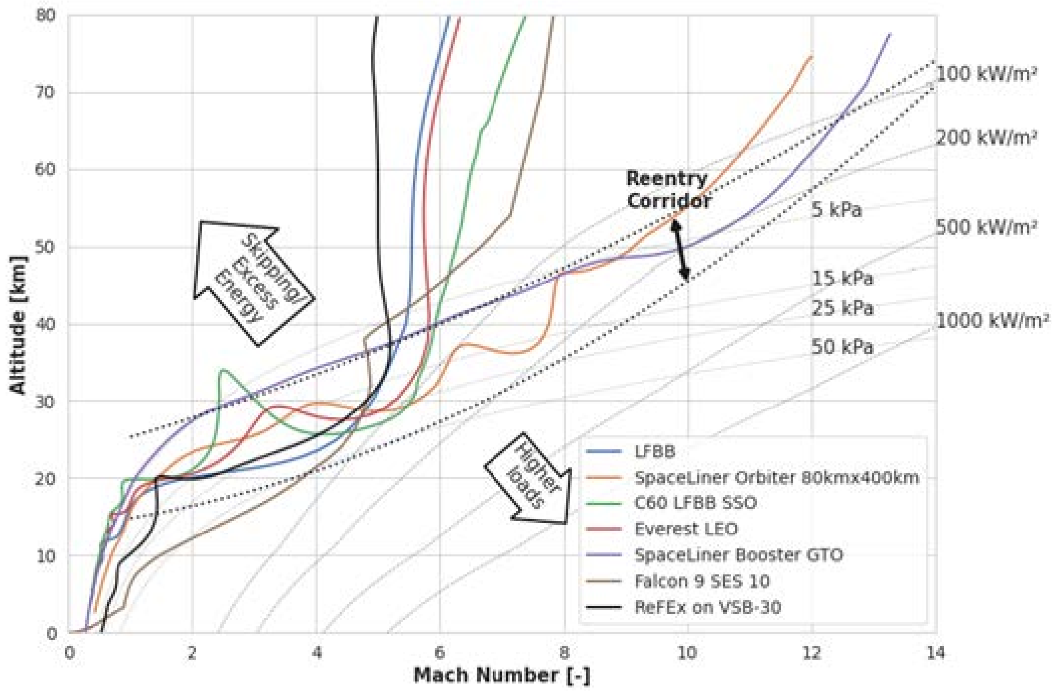

Figure 8: Re-entry corridor and examples of re-entry trajectories of different winged RLV stages including SpaceLiner, LFBB and micro-LFFB concepts (C60), the French EVEREST concept and the Falcon 9 trajectory of the SES 10 mission

Another challenge is the flight control while decelerating from Mach 2 to subsonic flight. As can be seen in figure 5 the vehicle will be statically unstable during parts of this transition which calls for a robustly performant control system that can cope with the expected uncertainties in aerodynamic attitude, the atmosphere and the aerodynamic coefficients. A flush air data system will be used to obtain direct measurements of the pressure distribution on the nose to estimate the aerodynamic angles, dynamic pressure and Mach to alleviate those uncertainties.

Another important aspect of the mission is the flight and range safety. The vehicle has to fulfil all local safety requirements in order to be certified for launch. In general, the Australian safety regulations require a casualty expectancy, which is the number of third-party casualties per launch, lower than 10^{-4}. To show that the vehicle is capable of fulfilling those requirements, a study on flight safety was initiated at DLR [6]. A vast number of Monte-Carlo simulations for different failure case assumptions were conducted and impact points of the vehicle were used to derive a probability distribution for which a typical result is presented in Figure 9.

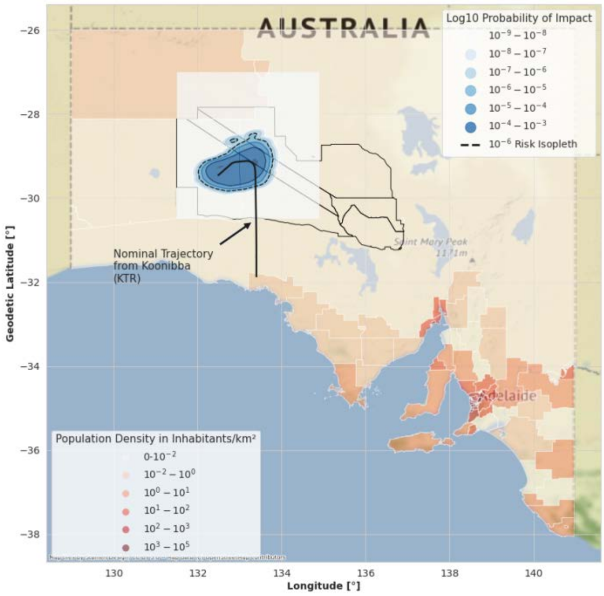

Figure 9: Calculated ground impact probability of ReFEx to support range safety

2.6 Development status, flight target

The ReFEx project is currently in between PDR and CDR phase. One of the main challenges was and still is the strong coupling of many subsystems with the aerodynamic shape of the vehicle. This situation was aggravated by the fact that ReFEx, being an aerodynamically controlled vehicle, is launched on an unguided sounding rocket and was maximized in size so not all aerodynamically active surfaces can be covered by a fairing [1, 2].

In order to reduce some of the coupled design iterations a basic aerodynamic design was frozen at the so-called aerodynamic configuration freeze (ACF) in January of 2019. At this point the hardware and flight software development were somewhat de-coupled. The reasoning was that a larger aerodynamic shape (i.e. more control surface area) would not be possible with the means available. This then defined a maximum envelope for the internal systems in particular the actuators for the system. The performance of these actuators was then maximized for the given volume. This allowed the design of the subsystems hardware to be further detailed, while the flight control software completed its iteration loops with an ever increasing CFD aerodynamic database. The performance of the vehicle now hinges on the capabilities of the flight control software making maximum use of the hardware available. This of course increases the risk in terms of mission requirements fulfillment, but allows a somewhat parallel development and speeds up the design process.

The next milestones coming up for ReFEx are the subsystem CDR season at the end of 2020 followed by a System CDR in beginning of 2021. During this time a structural model AIV campaign will be conducted to make informed decisions about potentially necessary design changes ahead of CDR.

For ReFEx only a protoflight model (PFM) will be built following the structural model campaign. This again is meant to streamline operations, reduce the number of models and adhere to tight budgetary and schedule deadlines. The PMF final integration campaign will commence in Q2 of 2022, with delivery of the final product to the launch range scheduled in the first half of 2023 and a launch a few weeks after delivery.

The flight will be followed by extensive analysis and a lessons learned session with the conclusion of the project at the end of 2023.

3 NEXT DEMONSTRATION STEPS AND POTENTIAL RLV DEVELOPMENT ROADMAP

DLR’s research in future launcher concepts aims for the initial operational capability of a next generation heavy launcher with reusable first or booster stage by the mid-2030s [3, 4, 5]. This target requires having all necessary technologies for reusability at a TRL between 5 and 6 at the end of this decade.

Obviously, the ReFEx flight experiment as described above addressing merely a few critical technologies for controlled hypersonic reentry is not sufficient to reach the TRL target. Additional demonstration steps are necessary at a relatively fast pace and are required to follow a clear demonstration logic.

DLR is currently investigating the potential demonstration steps and options which are to be evaluated afterwards on their feasibility and affordability. Some of these demonstrator options are published here for the first time.

Note: None of the hypersonic flight demonstrators described in the following sections have been finally selected or have attained any development funding yet. Additional options exist which will be further elaborated in the future. Flight demonstration beyond ReFEx is likely to be organized in European research and industrial collaboration which will also have some impact on the final selection process.

3.1 Accelerated demonstrator options

Near term opportunities for flying hypersonic demonstrators of limited size can be identified if an acceleration stage is to be used similar to the ReFEx-approach. This stage however, might be larger and consequently more performant than the one described above in section 2, allowing the integration of a heavier demonstrator with less aerodynamic restrictions be-cause it should be carried under fairing.

A small propulsion system might be included to further increase the winged stage’s performance, though without ground launch capability. Recovery of such a vehicle in its full integrity is preferred which would enable reflight of the same hardware.

3.1.1 VO LauncherOne

A possible successor of ReFEx could be launched atop the Virgin Orbit LauncherOne two-stage vehicle, which is capable of carrying small payloads up to 300 kg to SSO and 500 kg to equatorial LEO and is shown in Figure 10. This air-launched vehicle consists of two LOX/RP-1 stages. In the case of launching a suborbital flight demonstrator, the second stage could be replaced by an inert mass simulator, and the demonstrator could be accommodated in the standard fairing of LauncherOne. Such a configuration was proposed by Virgin Orbit during mutual collaboration on future ReFEx demon-strator mission requirements.

Figure 10: LauncherOne released from carrier aircraft "Cosmic Girl" (Picture by Virgin Orbit)

Whereas ReFEx reaches a re-entry velocity of slightly above Mach 5 (Figure 8), the successor mission should reach a higher velocity. Hence, possible injection parameters were investigated based on the assumption that the second stage is replaced by a mass dummy with the wet mass of the second stage and a payload mass of 200 kg to 600 kg. The latter value is estimated as the threshold mass that LauncherOne can carry without any modifications to the structure. The launch was assumed to occur over the North Sea/North Atlantic Ocean as shown in Figure 11. Thus, the "Cosmic Girl" 747 carrier aircraft could take off from any airport in the North Sea region, and could launch ReFEx in a direction parallel to the northern Norway coast line where it might be recovered close to Spitsbergen or in a different trajectory close to Andøya.

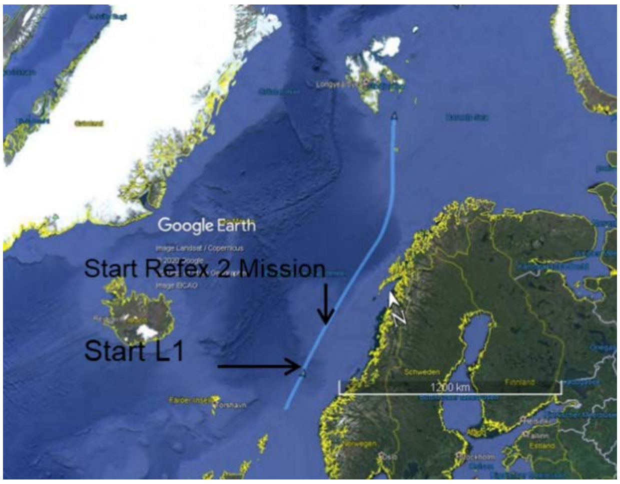

Figure 11: Exemplary launch of ReFEx successor on LauncherOne

Considering the fairing dimensions of LauncherOne, the current ReFEx vehicle would easily fit inside the fairing with extended wings. However, an increase in wing span without the use of foldable wings similar to the current version is not possible. Hence for this preliminary analysis, a generic demonstrator geometry was defined with foldable wings and thus a possible higher wing span and better aerodynamic behavior than the current version (see Figure 12, left). The investigations have shown again that a better aerodynamic performance is of advantage for the envisioned mission. However, the presented ReFEx successor layout is not a final design but rather a preliminary proposal of a potential geometry.

Possible re-entry trajectories were calculated in 3-DOF for a mass of 600 kg. The peak heat flux during re-entry is intended to be minimized. The analyses show that any LauncherOne based successor mission of ReFEx is expected to experience much higher heat loads than the current ReFEx mission, due to the significantly increased performance of LauncherOne compared to VSB-30. Furthermore, a mission with the current ReFEx geometry and relatively high 600 kg mass assumed tends to undershoot the re-entry corridor and would experience severe heat fluxes. This would require a stronger and heavier TPS of the ReFEx successor. A different geometry with a larger wing span and better aerodynamic performance can significantly reduce the re-entry peak heat flux: from 2.7 MW/m² (current ReFEx geometry on LauncherOne) to 1.2 MW/m².

The preliminary performance of such a demonstrator will be shown in section 3.3.

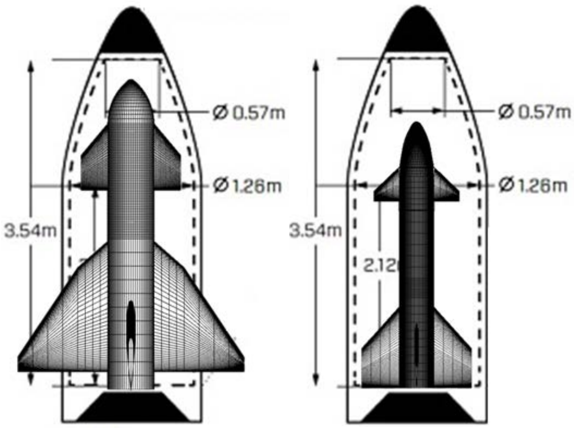

Figure 12: LauncherOne fairing with generic ReFEx successor (left) and current ReFEx geometry (right)

3.1.2 THEMIS

THEMIS is a flight demonstration for a non-winged reusable first stage to be landed vertically and currently under development at ArianeWorks. It is planned to be of similar or identical size of a full-scale booster stage and to be propelled by up to three Prometheus engines. As such it will provide the power and volume to accommodate a larger version of a winged flight demonstrator with an extended flight envelope. The winged flight demonstrator on top of THEMIS is assumed to be launched from Kourou CSG.

First assessments were conducted within DLR to assess the feasibility and interest in using THEMIS as an accelerator stage for an enlarged winged flight demonstrator. The wingspan of the flight demonstrator was set to be less than the outer diameter of the THEMIS rocket in order to avoid a hammerhead fairing that should encapsulate the winged body. For the feasibility study the geometry of the ENTRAIN VTHL first stage concept [4] was downscaled to fit under a potential THEMIS (e.g. Vega-C-class) fairing with some minor shape improvements.

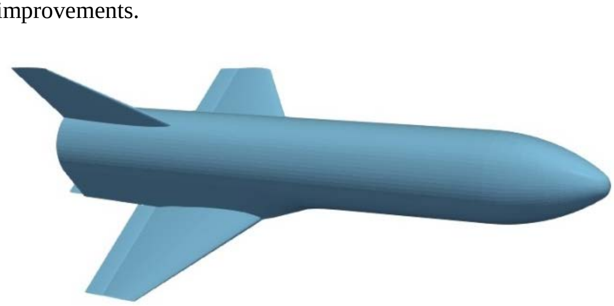

Figure 13: Potential aerodynamic shape for second generation hypersonic flight demonstrator on top of THEMIS

The selected wingspan of 3.04 m should allow sufficient spatial margin to fit under a THEMIS fairing. The fuselage diameter is set to 0.806 m and the overall length of the fuselage is 5.652 m. This geometry offers a subsonic trimmed L/D ranging from 4.0 to 4.3 and a supersonic trimmed L/D between 1.4 and 1.5 depending on Mach number. With an estimated descent mass of 1175 kg for the descent vehicle without any additional accelerator module and an estimated separation velocity from THEMIS of nearly 2.9 km/s at an altitude of 62 km the demonstrator is at Mach 8 when entering the flight range for RLV applications and hence represents an improvement with respect to REFEX.

The preliminary performance of such a demonstrator will be shown in section 3.3.

3.2 Autonomously powered demonstrator options

In a subsequent step, a larger, ground-launched autonomously powered hypersonic demonstrator with multiple reuses will be needed to reach a TRL between 5 and 6 for reusability. The vehicle will have a high performance rocket main propulsion system to achieve the necessary Δ-v for hypersonic flight conditions in a single stage. Overall size and complexity of the system is well beyond the demonstrator options described in the previous section 3.1 and should be understood to be based on their successful demonstration in flight.

The concept might make use of an existing full-scale engine of around 1000 kN initial thrust. However, a vehicle of reduced scale employing smaller upper stage engines might become more attractive in order to save cost while offering similar demonstration capabilities.

Several of such type winged hypersonic demonstrators have been preliminarily defined by DLR [12, 13] and performance has subsequently been evaluated. A few of the promising concepts are described here.

A feasible European rocket engine for this role is the Vinci which has been qualified to be used in the Ariane 6 upper stage [15]. It is designed as a closed expander cycle with the hydrogen flow heated in the regenerative circuit driving the turbines in series and subsequently injected into the main combustion chamber to be burned with oxygen. Vinci uses a large nozzle expansion ratio of 175 as an upper stage engine optimized for operation in vacuum conditions. In case the Vinci should be used as a propulsion system of a ground-launched flight demonstrator, the nozzle expansion ratio needs to be strongly reduced. All ceramic parts of the nozzle are to be removed and only the small regeneratively-cooled part remains which has an expansion ratio of 22. Actually, the engine is capable of operating in this arrangement under sea-level conditions as has been demonstrated several times during tests [12, 16].

An expansion ratio of 22 is feasible but not optimal for Vinci sea-level operation because the nozzle exit pressure is relatively low around 0.3 bar. Further, the nozzle exit angle is quite large increasing divergence losses. Nevertheless, analyses show that the cryogenic Vinci is capable of providing good performance for the demonstration task. A major advantage of selecting Vinci would be that a modern engine could be used which is under production and available at an affordable price. Although Vinci is not designed for multiple reusability per se, without questions the engine can be used safely a few times in the role of powering a reusability demonstrator.

All rocket-engine powered winged flight demonstrators have been assumed to be launched from Kourou CSG.

3.2.1 Fly Back Configuration

The demonstrator is equipped with a single Vinci engine providing approximately 137 kN lift-off thrust. A single turbofan Larzac 04-C20 (see [12]) should be sufficient to allow for a powered airbreathing flyback as it has been previously foreseen with the LFBB. The preliminary dimensions of such a stage are shown in Figure 14. With an overall length of more than 16 m and span of 8 m this demonstrator is significantly larger than the accelerated vehicles. On the other hand it is considerably smaller than the proposed THEMIS.

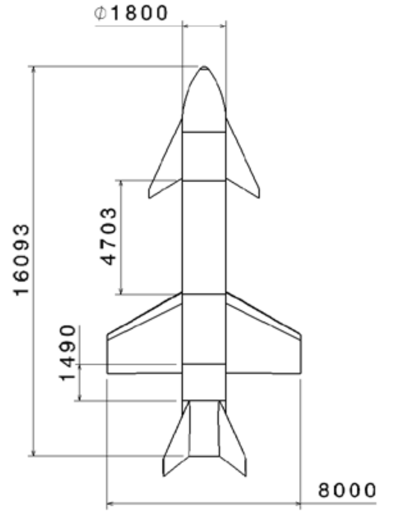

Figure 14: Sketch of Winged Demonstrator Configuration with major dimensions [12]

Lift-off weight is estimated at 11125 kg with a dry mass of 4190 kg. An ascent, descent and fly-back loop as might be flown over the Atlantic off the coast of French Guiana is shown in Figure 15. The requirement of autonomously bringing the stage back is restricting the flight performance. Approximately a maximum speed of 1.4 km/s (Mach 4.5) is reached in 38 km altitude at MECO-condition.

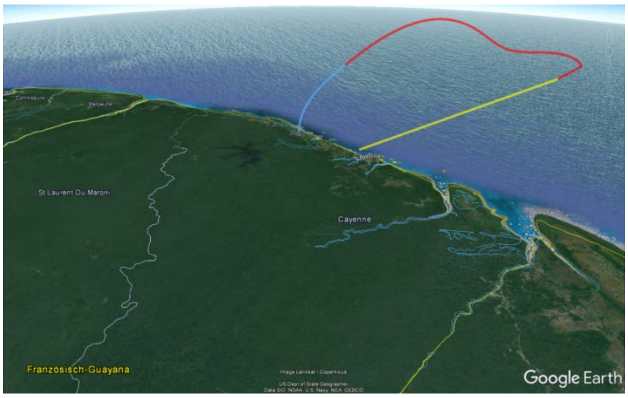

Figure 15: Ascent (blue), ballistic and descent (red) and fly-back (yellow) trajectory segments of winged fly-back demonstrator [12]

3.2.2 Downrange Glide Configuration

The downrange glide configuration does not consider any active return flight of the demonstrator to the launch site. As the flight for safety reasons can only be performed over uninhabited areas, the mission is planned in easterly direction over the Atlantic Ocean. A suitable downrange landing site in the sea is not available. In order not to lose the reusable vehicle, the patented in-air-capturing procedure (see detailed descriptions in [18, 19]) is proposed for safe recovery of the demonstrator stage. After towing the stage back to a suitable landing field, the winged stage is released for autonomous touch-down on a runway. The patented “in-air-capturing” (IAC) technology is now flight tested on lab-scale level [19] and is part of DLR’s development roadmap for future RLV.

Lift-off weight is estimated at 9986 kg with a dry mass of 3725 kg [12]. The dimensions of this type are identical to those shown in Figure 14. The same 6100 kg propellant loading is available for the ascent mission. Due to the better mass ratio and increased lift-off acceleration of the lighter vehicle, the mostly similar demonstrator achieves velocities beyond 2 km/s [12] as is shown in section 3.3.

A similar downrange glide configuration with 2 Vinci engines would allow for higher propellant loading and result in increased flight performance. Such a demonstrator option is currently under investigation.

3.3 Technical characteristics and performance comparison

A size, mass and performance comparison of the previously outlined future demonstrator options is summarized in Table 1 and associated to the ReFEx-configuration planned for its 2023 mission. The future accelerated demonstrators are still in a similar class of size and even mass as ReFEx. The ground-launched vehicles increase by at least an order of magnitude in mass, however, a large portion of the mass being the propellant.

Table 1: Size and mass comparison of future winged demonstrator options

	overall length [m]	span [m]	initial mass [kg]
ReFEx 2023	2.7	1.1	400
Accelerated demonstrators			
LauncherOne-air-launch	3.5	2.5	600
THEMIS-CSG-launched	5.65	3.04	1175
Ground-launched powered demonstrators from CSG			
LFBB, Vinci	16.1	8	11125
IAC, Vinci	16.1	8	9986

Figure 16 depicts the Mach-altitude dependencies of the different demonstrator options during their experimental phase. Therefore, the orange line of the ground-launched Vinci-powered vehicle has two branches. The ascent flight starts in 0 m at Mach 0 and accelerates up to approximately Mach 6 in 50 km before the ballistic down-range glide phase is initiated. This simulation ends in subsonic flight at around 8 km for the “in-air-capturing”-maneuver. A Mach-number of above 10 in the relevant aerothermal regime between 40 km and 50 km can be reached using the LauncherOne-accelerated option. Based on the current assumptions for THEMIS, the flight demonstration might reach at least Mach 7 to 8 in 40 km.

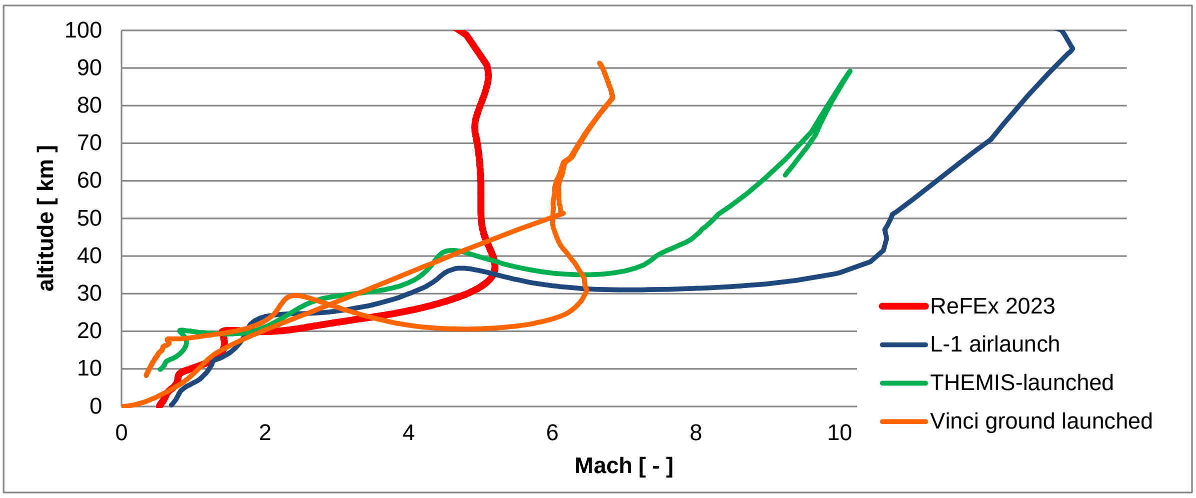

Figure 16: Re-entry (and ascent flight) characteristics of future hypersonic winged demonstrator options

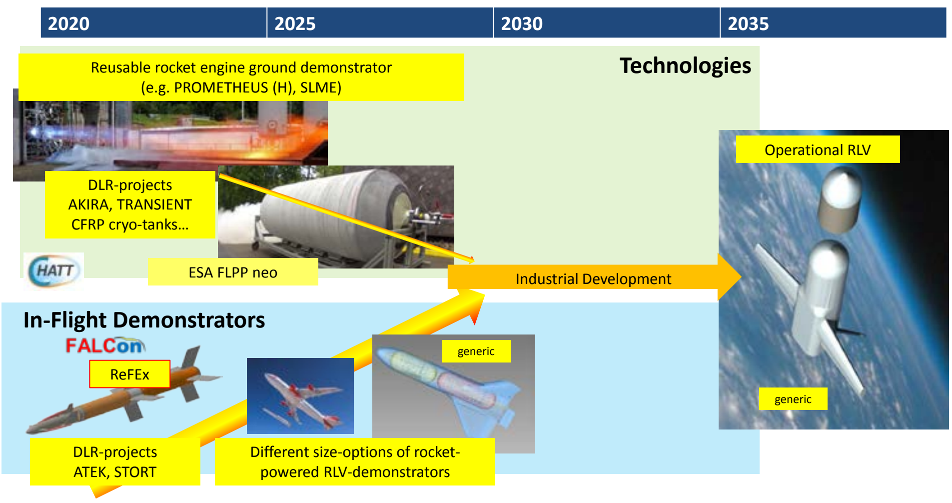

Figure 17: Winged RLV development roadmap

3.4 Potential Roadmap for Demonstrators to RLV Development

Target date for the Initial Operational Capability (IOC) of a completely new, partially reusable European launch vehicle is around 2035. Further assuming at least 5 years for development and qualification of the RLV, a TRL of 6 should be achieved by 2030 for multiple, representative, reusable hypersonic flight demonstrations.

Figure 17 shows some key-elements of a roadmap for winged RLV development which is compliant with the IOC of such a new European launcher in 2035. Technology development, including ground demonstrators will be needed as well as in-flight demonstrations of different size and speed range. Some of the projects funded by DLR, ESA or the EC are listed in the roadmap; however, the presentation of ongoing initiatives might not be complete.

Note, the next steps after the ReFEx flight in 2023 need to follow each other closely synchronized and in carefully planned sequence to meet the development target. The multiple flight-demonstration options, of which a few have been presented in this section, will be evaluated and oriented towards a clear development logic.

4 CONCLUSION

DLR is preparing the next generation of partially reusable European launchers. A combination of systematic system concept studies, technology development and flight demonstration is used for this purpose.

The ReFEx hypersonic flight demonstrator currently under development will master a typical reentry corridor of RLV from hypersonic down to subsonic flow conditions. The ReFEx flight is scheduled for early 2023.

Analyses of subsequent, more ambitious flight demonstration options started in-line with a launcher development roadmap. The next generation of demonstrators is not yet frozen and might be realized in European and international cooperation.

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