SUPERSONIC RETROPROPULSION FLIGHT TEST CONCEPTS

Ethan A. Post(1), Ian C. Dupzyk(2), Ashley M. Korzun(3), Artem A. Dyakonov(4), Rebekah L. Tanimoto(1), Karl T. Edquist(4)

(1)Jet Propulsion Laboratory, California Institute of Technology, 4800 Oak Grove Drive, Pasadena, CA, 91109, USA, Email: ethan.a.post@jpl.nasa.gov, rebekah.l.tanimoto@jpl.nasa.gov
(2) Science and Technology Corp., NASA Ames Research Center, Moffett Field, CA 94035, USA, Email: ian.c.dupzyk@nasa.gov
(3)Georgia Institute of Technology, 270 Ferst Drive, Atlanta, GA, 30332, USA, Email: akorzun@gatech.edu
(4) NASA Langley Research Center, Hampton, VA, 23681, USA, Email: artem.a.dyakonov@nasa.gov, jeremy.d.shidner@nasa.gov, karl.t.edquist@nasa.gov

ABSTRACT

NASA's Exploration Technology Development and Demonstration Program has proposed plans for a series of three sub-scale flight tests at Earth for supersonic retropropulsion, a candidate decelerator technology for future, high-mass Mars missions. The first flight test in this series is intended to be a proof-of-concept test, demonstrating successful initiation and operation of supersonic retropropulsion at conditions that replicate the relevant physics of the aerodynamic-propulsive interactions expected in flight. Five sub-scale flight-test article concepts, each designed for launch on sounding rockets, have been developed in consideration of this proof-of-concept flight test. Commercial, off-the-shelf components are utilized as much as possible in each concept. The design merits of the concepts are compared along with their predicted performance for a baseline trajectory. The results of a packaging study and performance-based trade studies indicate that a sounding rocket is a viable launch platform for this proof-of-concept test of supersonic retropropulsion.

1. INTRODUCTION

Supersonic retropropulsion (SRP) is an advanced entry, descent, and landing (EDL) supersonic decelerator technology that, if developed, could significantly increase landed mass capabilities at Mars [1, 2]. In the development of future mission concepts, NASA has recognized the need for advanced EDL systems and has included SRP among low-TRL technologies being funded for development. SRP has also been included in the EDL roadmap released by the NASA Office of the Chief Technologist [3]. SRP has been assessed to currently be at Technology Readiness Level (TRL) 2, "Technology concept and/or application formulated." A roadmap has been developed for the maturation of SRP to TRL 6, at which point SRP is likely considered to be sufficiently mature for incorporation into a flight project [4]. Wind-tunnel testing, systems analysis, and computational fluid dynamics simulation efforts are under way within NASA's Exploration Technology Development and Demonstration (ETDD) Program. The work described in this paper represents a focused effort to define Earth-based SRP flight testing concepts for a proof-of-concept flight test (FT 1). These flight test concepts complement ground testing and analytical efforts and will play a critical role in the maturation of SRP to a viable flight system.

Supersonic retropropulsion is characterized by propulsive deceleration and maneuvering at conditions of supersonic flight and is accomplished by directing engine exhaust upstream (ahead of the vehicle). Propulsive-aerodynamic interactions at supersonic conditions are markedly different from those at subsonic conditions [5]. A series of three flight tests within the Earth's atmosphere are currently planned to support SRP development, with two flight tests required to achieve TRL 5 and one additional flight test required to achieve TRL 6 [4]. Prior to FT 1, ground-based testing and computational simulation will have been completed, establishing predictions of FT 1 test article performance and aerodynamic/aerothermal databases. Additional details on the SRP development roadmap can be found in the reference documentation.

These flight tests will be at scales, conditions, and integrated system complexities beyond those achievable in ground test facilities. NASA's Entry, Descent and Landing Systems Analysis team has identified relevant SRP operating conditions for several EDL architectures to land humans and cargo on Mars [1]. While these conditions vary between architectures, they include freestream Mach numbers above 1.7 and thrust coefficients above 8.0. Analytical and experimental efforts have established the primary parameters governing SRP flow physics to be the exit conditions of the retropropulsion nozzle(s), composition of the retropropulsion exhaust flow, and relative areas of the vehicle and retropropulsion nozzle(s) [5]. These parameters are not all directly considered in the development of concepts for FT 1; they are, however, likely to drive concept development for follow-on flight tests.

2. OBJECTIVES AND MISSION REQUIREMENTS

2.1 Objectives

The objective of high-altitude, powered flight testing for SRP is to successfully complete a series of stable and controlled instrumented flight tests, replicating the relevant SRP physics and progressively increasing the level of fidelity of subsystem integration toward a fully-integrated, flight-viable EDL system. A secondary objective is to collect flight test data to support the demonstration of computational fluid dynamics (CFD) prediction capability, the reconstruction of environmental conditions and flight mechanics, and the simulation of environmental inputs to the guidance, navigation, and control (GN&C) subsystem. Significant results from powered, high-altitude flight testing include supersonic initiation characteristics, aerodynamic and aerothermodynamic performance from initiation through nominal operation, and flight dynamics data that can be scaled and applied to a full-scale flight vehicle at Mars.

The objectives of FT 1 are to:

- Demonstrate proof-of-concept for SRP in a flight environment,
- Replicate relevant SRP physics using a minimally integrated system,
- Collect data during flight within acceptable uncertainties to satisfy relevant TRL achievement criteria,
- Demonstrate the ability to design, package, integrate, and test SRP subsystems,
- Reduce the risks associated with increasingly complex follow-on flight tests.

Proof-of-concept is defined to be successful operation (no significant deviations from expected behavior), from initiation through nominal operation, of a “hot” SRP system at conditions that replicate the relevant physics of the aerodynamic-propulsive interaction.

2.2 Mission Requirements

Mission requirements have been defined for this proof-of-concept flight test. These requirements were used to define the trade studies and performance analyses across sounding rocket platforms as described in the sections that follow. Requirements are defined for the Flight Segment and the Ground Segment. The Flight Segment includes the launch system and the test article. The SRP system is a part of the test article. The Ground Segment includes operations, telemetry and tracking, and data processing and analysis.

The mission requirements specify the performance of the launch system and test article, test phase, trajectory, measurements and data analysis, and use of commercial, off-the-shelf components for both the launch system and test article, where possible. The mission requirements can be summarized as:

- Achievement of SRP by directing hot, propulsive jet flow against a supersonic freestream.
- Test phase duration of at least 15 seconds with M_∞ at initiation greater than 2.0 and C_T greater than 5.0 throughout the test phase (C_T is a force coefficient, analogous to drag coefficient).
- Ballistic and stable flight throughout the entire mission trajectory.
- Collection and analysis of data required for post-flight reconstruction, including: atmospheric characterization, 6-DOF test article state, propulsion system performance and state, and in-situ surface pressure and temperature.
- Utilization of existing components for launch system and test article.

3. SYSTEM ANALYSIS

At the beginning of this study, there was some concern as to what platform could serve as the test venue for this first experiment. Performing the test on a sounding rocket would be less expensive than a balloon-launched test article, which is typically used for Mars parachute verification. While the latter can provide a more representative flight environment for the test than the sounding rocket, it is more complicated and expensive. Furthermore, the first flight test is a proof-of-concept test appropriate to a TRL 3-4 test, so targeting a Mars representative flight environment is not required. For these reasons a sounding rocket platform was deemed to be the preferable test platform.

When initially considering a sounding rocket there were two primary concerns raised. The first was whether or not a sounding rocket could provide sufficient test duration at acceptable conditions. Many sounding rocket trajectories are designed to be steep through the atmosphere on ascent and descent to deliver its payload to the highest altitude, some in the hundreds of kilometers [6]. Trajectories such as these are problematic for this type of test, since the preferred test altitude range was found to be 40 km to 50 km from the point of view of dynamic pressure and continuum flow. The result of such a trajectory would be a short duration with a rapidly changing environment. Therefore, the trade study was intended to assess what range of test initiation conditions would result in acceptable conditions, with respect to dynamic pressure and continuum flow, throughout the duration of the test.

The second concern was whether the major system components needed for the test could be packaged into an existing launch vehicle. The majority of common sounding rockets have a 17.26-inch diameter payload fairing and can handle payloads in the 3 m range. This concern was assessed by a packaging study that attempted to package the most stressing configuration of the ones considered.

3.1 Trade Study

The objective of this trade study was to understand the range of thrust that would be required for the retropropulsion rockets and the acceptable range of test initiation conditions that could be tolerated by the test article. The initial flight conditions were defined by Mach number, flight path angle, and altitude. Of these variables, only the flight path angle and the altitude were traded since it was more desirable to understand the effect of the test article thrust on the system rather than deployment Mach number.

For this initial trade study, the metric for success or failure for the test article was derived from the mission requirements and consisted of the test article thrust coefficient. Another metric considered in this first trade study was the change in Mach number from thrust initiation to termination, or Mach delta. At this stage it was perceived that having a greater Mach delta was more beneficial to the test results; however, since this hypothesis was not traceable to any specific quantities that were desired from this test, it was observed but not used as a disqualifying criterion.

The 3-DOF trajectory analysis was performed with POST II using the 1976 standard atmosphere. Figure 1 shows the population of the trade study in a thrust coefficient-Mach delta space. Out of the 1080 cases that were run, 265 of the cases met the criterion for success, which required the minimum thrust coefficient encountered to be greater than 5.0.

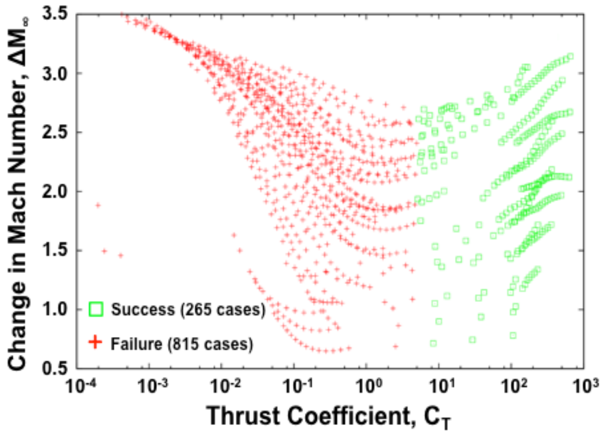

Figure 1. Trade study results indicating successes and failures.

The trade results represent a range of test article thrusts from 1000 N to 5000 N, flight path angles from 0.0 degrees to -80.0 degrees, deployment altitude from 40 km to 70 km altitude, and burn times from 15 to 25 seconds. Due to the range of the test article thrust and burn time, the mass of the test article would be expected to vary substantially. To capture this, the mass of the system was estimated through the use of Mass Estimating Relations (MERs) for a Hypergolic rocket engine. The structure mass and communication mass was assumed to be constant across all cases. The engine mass was determined from a curve fit of existing hypergolic engines ranging in thrust from 100 N to 4000 N, which provided the engine mass as a function of the engine thrust. Finally, the fuel tanks were determined from the thrust, specific impulse of the engine, and the burn duration. In taking this approach, the objective was not to predict the precise mass of the test article but, instead, to capture the trends of the system.

3.2 Packaging Study

The initial trade study indicated that a sounding rocket was a viable option for FT 1 given the constraints initially set. To address the second concern of whether the major components of the test article can be packaged into a standard 17.26-inch-diameter sounding rocket, a packaging study was performed. While the trade study considered a hypergolic rocket system, this was not the only system to be considered. In addition, both monopropellant systems and solid motors were to be considered. For the first case, if the system that had the lowest bulk density and the lowest propulsive efficiency (Isp) could be packaged, then the other concepts should be packageable. As a result, the monopropellant configuration was packaged first, since this configuration would be the lowest bulk density that would translate to the system that would require the largest volume.

After demonstrating that the system could be packaged for a sounding rocket, four additional concepts had packaging configurations designed. The different configurations consisted of a hypergolic system, a blow-down and a pressure-fed monopropellant system, and two different solid motor concepts. The test article masses, specific impulse, and thrust profiles were used to run similar but smaller scale trade studies than the initial study. For this trade, the altitude range was reduced to 40 km – 60km and the flight path angle was varied from 20 degrees – -50 degrees. In addition to the thrust coefficient constraint, these trades were also constrained by the ratio of free-stream, normal shock stagnation pressure (P_{02}) and the rocket nozzle static exit pressure (P_e). If the exit pressure of the jet drops below P_{02}, the flow in the nozzle will be overexpanded and might tend to separate, resulting in a disruption of the physics desired to be modeled.

The results of these trades indicate that a flight path angle of -40 degrees is about the maximum the system can handle before overexpansion and nozzle flow separation could occur. These results feed into the selection of the launch vehicle.

3.3 Launch Vehicle Selection

The initial packaged concept trade study indicated that flight path angles steeper than -40 degrees might cause issues with the desired measurements to be taken, due to short test duration and rapidly changing environments of steep trajectories. The next step was to look at what available sounding rocket vehicles can accommodate this payload. The launch vehicle analysis and recommendation was performed by Wallops Flight Facility (WFF). The constraints given to WFF were the following: 40km – 50km altitude at deployment, flight path angle no steeper than -40 degrees, and velocity between 700 and 1000 m/s. The analysis that was performed indicated that two vehicles would be capable of targeting these conditions. These were the Terrier MK12 – Improved Orion and the Terrier MK12 – Black Brant. Table 1 summarizes the capabilities of each vehicle at 40- and 50-km deployments.

Table 1. Summary of launch vehicle performance.

	Terrier MK12 – Improved Orion		Terrier MK12 – Black Brant	
Altitude (km)	50	40	50	40
Relative Velocity (m/s)	871	974	877	980
Relative Flight Path Angle	-30.3°	-39.4°	-31.0°	-39.9°

The performance of both vehicles is very close; still, the values of the Terrier MK12 – Improved Orion had a slightly more shallow flight path angle and was used for the remainder of this study to baseline the trajectory.

3.4 Configuration Analysis

To compare the different test article concepts, five concepts were simulated on the baseline launch vehicle trajectory and are tabulated below. Both of the monopropellant test article concepts in Table 2 violate the constraints set earlier in the trade studies for thrust coefficient dropping below a value of 5.0. This indicates that the monopropellant concepts are more sensitive to a plunging trajectory and might require deployment at a more shallow flight path angle. In both cases, the thrust coefficient drop is in the last few seconds of the “thrust-on” portion of the test. While a down-selection has not yet been performed, these results reduce the priority of both monopropellant concepts, since the three remaining options can satisfy the requirements. Furthermore, the launch vehicle analysis was simply designed to show that the current target conditions can be achieved by one of these sounding rockets. A more in depth analysis might show that shallower flight path angles might be possible with this vehicle, alleviating the requirements violation for the monopropellant concepts.

Table 2. Comparison of simulated flight test concept performance.

Concept	Propellant Type	Burn Time (sec)	Thrust (N), max/min	$C_{T,min}$	$P_{02,max} / P_e$ (< 1.0)	ΔM
1	Hydrazine (pressure-fed)	35.0	3100 / 3100	4.2	1.170	0.40
2	Hydrazine (Blow-Down)	24.0	3100 / 800	2.0	0.950	0.02

3	N_2O_4/MMH (pressure-fed)	30.0	4003 / 4003	8.0	0.680	0.85
4	Solid (STAR 13B)	15.6	9643 / 6007	75.0	0.104	1.40
5	Solid (STAR 15G)	36.4	12460 / 1744	80.0	0.144	2.10

4. FLIGHT TEST CONCEPTS OVERVIEW

One of the goals of SRP flight test development was to identify a platform that is capable of delivering the flight test article to the required conditions. A sounding rocket is the preferred test platform, due to a long history of flight-proven reliability and lower cost, as compared to other high-altitude delivery options, such as a rocket-assisted balloon launch. For these reasons and on the basis of the supporting analysis discussed in the previous section, all of the concepts discussed here utilize a sounding rocket platform. The ability to package a flight test article that satisfies mission requirements was also assessed, as the packaging of the test article hardware within the constraints of a sounding rocket is a known challenge. Moreover, the packaging assessment helped to identify the overall dimensions of the flight test article and the most practical arrangements of the propulsion system hardware and other components.

The type of propulsion system is the main distinguishing feature between the five flight test concepts. Monopropellant, bipropellant, and solid-rocket motor-based propulsion systems were sized for different flight test concepts to identify the benefits and challenges of the different alternatives. All propulsion system concepts use a single, centrally located nozzle.

Each of the five flight test concepts has four elements: (1) an attitude control system (ACS), (2) telemetry, (3) a flight computer and an inertial measurement unit (IMU), and (4) propulsion and instrumentation. The ACS, telemetry, and flight computer and IMU elements were assumed to be common for each concept; the propulsion and instrumentation elements are different. Figure 2 shows the generalized flight test concept with each element identified.

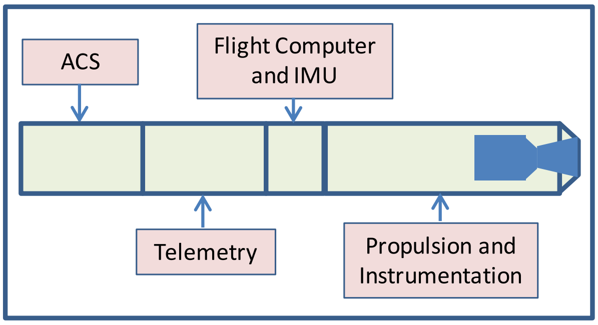

Figure 2: Generalized flight test article concept broken down by element.

The attitude control system was included in each of the concepts to maintain the stability of the test article by correcting for any potential SRP thrust vector misalignment and aerodynamic disturbance torques. A trade between passive and active control has not yet been completed; however, a NASA Sounding Rocket Operations Contract Inertial ACS (NIACS) was assumed for preliminary mass and volume estimates. The propulsion and instrumentation element contains the SRP propulsion hardware, including the engine, the propellant and pressurant tanks, and other miscellaneous propulsion components. The instrumentation includes a ring-like arrangement of pressure transducers, as well as thermocouples that gather aerothermal data. Each concept has a conical forebody, truncated to accommodate the centrally-located retropropulsion nozzle. The skin structure of the sounding rocket was assumed to be 0.125-inch-thick aluminum. Differences in the propulsion and instrumentation elements resulted in varying lengths of the skin between concepts. A side–by-side comparison of the five concepts is shown in Figure 3.

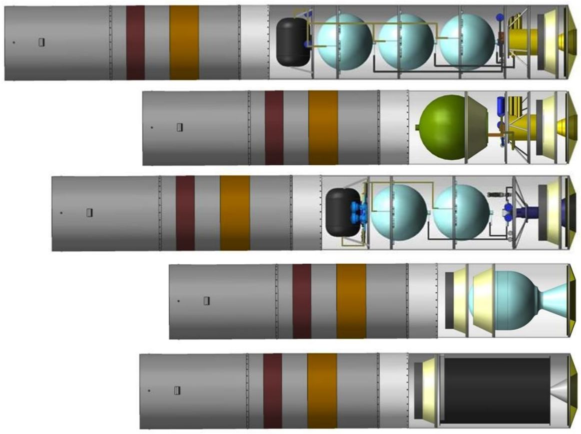

Figure 3. Five flight test concepts shown to scale.

The propulsion and instrumentation elements are shown for the five flight test concepts in Figure 4 through Figure 8. All flight test concepts are presented in a minimum length configuration. Each test article could feasibly be elongated in the aft direction with additional structure if detailed design determines a test article with a finer aspect ratio to be more desirable for aerodynamic stability. Each flight test concept has space allocated aft of the forebody for any ballast required to adjust the center of gravity (CG) of the test article.

4.1 Pressure-fed Monopropellant Concept

This concept, shown in Figure 4, uses an Aerojet MR-80B hydrazine engine capable of producing 3100 N of thrust. Three 13-inch spherical diaphragm tanks are loaded with 49 kg of propellant to provide a 35 s burn duration. Ample clearance between the tanks and the sounding rocket skin inner wall is provided to accommodate propellant lines. An approximately constant thrust level is maintained using a "bang-bang" type pressurization system. A composite overwrapped pressure vessel contains the helium pressurant.

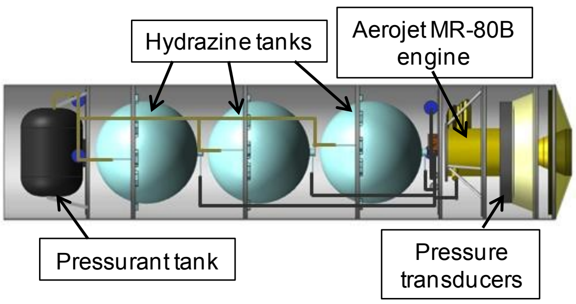

Figure 4. Pressure-fed monopropellant concept.

4.2 Blow-down Monopropellant Concept

This concept, shown in Figure 5, uses the same MR-80B hydrazine engine as the pressure-fed concept in Section 4.1. A single 15-inch spherical-diaphragm tank is loaded with 22 kg of propellant. The absence of a pressurization system allows the propulsion system to be shorter than the similar pressure-fed concept described in Section 4.1. Due to the blow-down design, the thrust decays from an initial 3100 N to 1041 N over 24 s. The lower average thrust and shorter burn duration results in a narrower velocity range and lower thrust coefficient during the test phase, compared to the pressure-fed monopropellant concept.

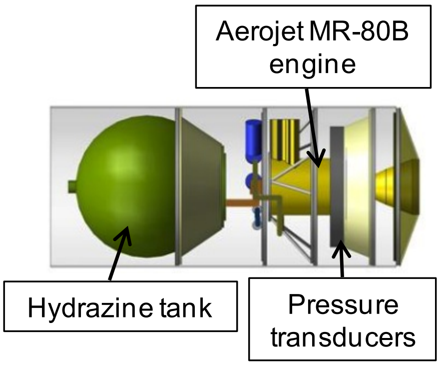

Figure 5. Blow-down monopropellant concept.

4.3 Pressure-fed Bipropellant Concept

This concept, shown in Figure 6, uses an Aerojet R-40B bipropellant engine. The nozzle is truncated in the configuration shown to keep the flow from becoming overexpanded. Two 13-inch propellant management devices are used to store the monomethyl hydrazine (MMH) fuel and nitrogen tetroxide (N_2O_4) oxidizer. The thrust level is maintained using a regulated pressurization system and the burn duration is 30 s. A composite overwrapped pressure vessel contains the helium pressurant.

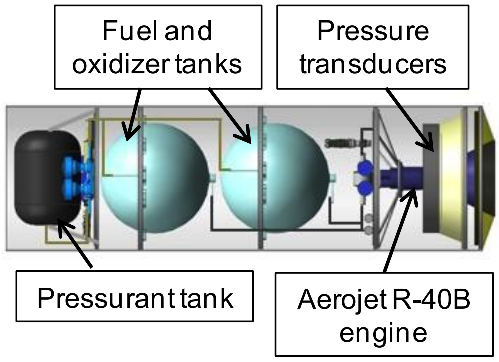

Figure 6. Pressure-fed bipropellant concept.

4.4 Solid Motor Concept I (STAR 13B SRM)

This concept, shown in Figure 7, uses an ATK STAR 13B solid rocket motor (SRM). Overall, this is a simpler design than the liquid propellant-based concepts due to the lack of tubing, valves, filters, and pressure-control hardware associated with those concepts. The simplicity of this design makes this concept the most compactly packaged.

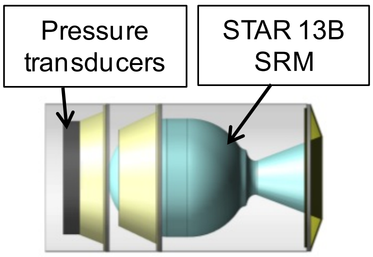

Figure 7. Solid motor concept I, with STAR 13B SRM.

4.5 Solid Motor Concept II (STAR 15G SRM)

This concept, shown in Figure 8, uses an ATK STAR 15B SRM. The main difference between this concept and the STAR 13B concept is that the average thrust is significantly higher for the STAR 15G and the burn duration is longer, as shown in Table 2.

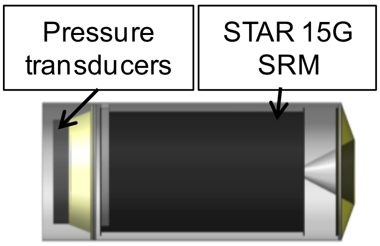

Figure 8. Solid motor concept II, with STAR 15G SRM.

While each concept described above can be packaged in a sounding rocket, their physical dimensions and mass distributions differ. The lengths and mass characteristics of all five concepts are summarized in Table 3 for both the pre and post SRP burn cases.

These masses were estimated from vendor-supplied data, ground-up design, and rough sizing estimates. The lengths given in Table 3 are measured from the forward end to the aft end of the sounding rocket skin. The pre-burn axial CG and post-burn CG locations are the CG locations before SRP initiation and after SRP burnout, respectively, measured from the forward end of the sounding rocket skin. No ballast has been placed to adjust the CG location. The masses are current best estimates.

Table 3. Lengths and mass characteristics of FT 1 concepts.

Concept	Propulsion Type	Pre-burn Mass (kg)	Post-burn Mass (kg)	Length (m)	Pre-burn axial CG Location, (m)	Post-burn axial CG Location, (m)
1	Monoprop (pressure-fed)	234	184	3.4	1.6	1.8
2	Monoprop (blow-down)	194	172	2.5	1.3	1.4
3	Biprop (pressure-fed)	220	178	3.0	1.4	1.6
4	Solid (STAR 13B)	186	144	2.4	1.1	1.3
5	Solid (STAR 15G)	234	154	2.5	1.1	1.4

5. CONCEPT OF OPERATIONS

The flight test concepts described in Section 4 were each designed for a 17.26-inch-diameter sounding rocket. The launch vehicle and flight test article configurations at each separation event are shown in Figure 9. A sounding rocket launch profile was determined that enabled the flight test article to meet the desired initiation conditions described in Section 3.3.

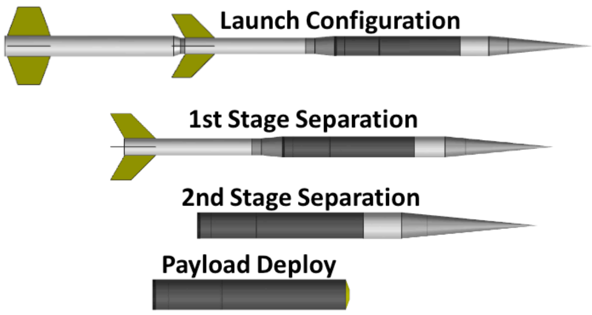

Figure 9. Flight test article configurations at launch and each separation event.

As shown in Figure 10, the sounding rocket first stage burns out 5 s after launch, at 1.8-km altitude. Second stage ignition is delayed for 25 s to suppress the trajectory. The second stage burns out at 29 km, 25 s after its ignition. After separating from the launch vehicle, the payload coasts to an apogee altitude of 60 km. The nose cone and forward ACS then separate from the payload. The payload descends to an altitude of 50 km and begins recording and transmitting data, marking the start of the test phase. After a 5-s delay, SRP is initiated. SRP terminates at burn out. After a 5-s delay, data recording and transmission ends, marking the end of the test phase. The flight test article ends the flight test by impacting the ground or splashing down into the ocean. There is currently no plan to recover the vehicle.

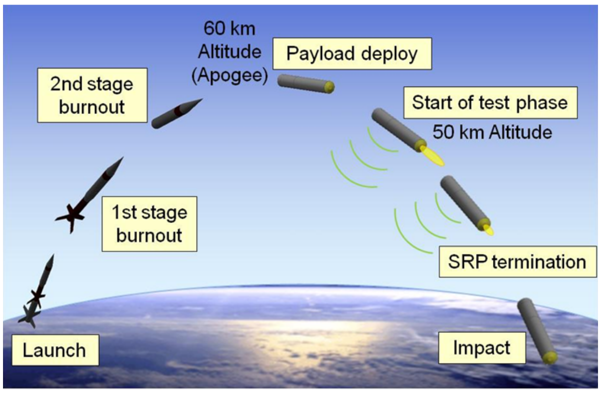

Figure 10. Concept of operations for FT 1.

6. CONCEPT EVALUATION

Several viable test architectures are the primary results of this study with the propulsion system type as the principle discriminator between the architectures. Evaluation of these concepts toward a down-selection requires an understanding of significant performance and cost factors. This study does not present a ranked set of concepts, mainly because the analysis is in early phase. Instead, recommendations are made with respect to the performance and most relevant cost factors, with emphasis on the latter.

Two out of five FT 1 mission requirements focus on the performance of the test article. Requiring a ballistic and stable flight through the entire mission trajectory places constraints on the test article's aerodynamic characteristics and, possibly, the inclusion of a passive or active stability device. Requiring a minimum duration of the test phase, the initiation Mach number, and the thrust capability relates the test article's ballistic coefficient, thrust coefficient, and trajectory parameters to the overall objective to replicate relevant SRP physics. With this in mind, all architectures were found to be capable of decelerating through a range of velocities sufficient to satisfy the proof-of-concept aspect of the FT 1 objectives. Differences in achievable ΔM_∞, while notable, are less critical in the opinion of the authors because SRP flow physics should not change qualitatively between $M_\infty = 2$ and, for example, $M_\infty = 3.5$. A flight demonstration at $M_\infty = 2$ is just as valid as one at $M_\infty = 3.5$. It is desirable to end the test phase below $M_\infty = 1$ as a direct result of the challenges of testing the transition through transonic conditions in ground test facilities.

Each of the considered architectures generates sufficient thrust over an acceptable duration. In some cases, for steeper trajectories with an MR-80B engine (see Section 4.1, 4.2), the thrust coefficient might drop below 5.0 near the end of the test phase. However, the majority of the test phase is at conditions that are satisfactory to the mission requirements. In the end, any of the three propulsion system types can be made to work within the FT 1 architecture to deliver desired performance.

Aside from performance characteristics, there are other factors that should be considered in the down-selection of propulsion system options. The availability of a given engine plays a key role in the schedule and cost of the flight test project. Questions such as, "Is it readily available?", "If not – when was the last time this engine was manufactured?", and/or, "Are any of the personnel involved with the last manufactured item available for the manufacturing of another system?" should be asked. Answers to the above questions may inform the focus of future studies. For example, the Aerojet R-40B engine has been out of production since the mid-1980s; new manufacture of this engine is extremely unlikely. At the same time, the R-40B serves as the Reaction Control System (RCS) engine on the Space Shuttle; as the Space Shuttle nears retirement, it might be possible to inexpensively procure some R-40B engine spares. Aerojet's MR-80B engine is, on the other hand, a current production; a set of eight of these engines was recently installed on the Mars Science Laboratory descent stage as Mars Lander Engines. It is worth noting that the MR-80B engine is a special use engine and requires a two-year procurement time. No MR-80B engines are available off the shelf. ATK engines STAR 13B and 15G have been out of use for an extended time. While it might be possible to procure one of these engines, this will require a lead time on the order of two years and non-recurring costs that will have to be absorbed over the small scale manufacture for FT 1. In light of the above, and of the four choices listed, only the MR-80B looks like an attractive candidate at this time.

Aside from the above engines, there are other options. Experimental engines, including small LOX/CH_4 engines are being developed (for example, for NASA's Morpheus project). The benefits of LOX/CH_4 engines include ease of handling, use of a non-toxic propellant, and synergy with the on-going development effort, which might result in some cost sharing. The risk associated with using experimental engines would, of course, have to be considered.

The cost/availability and performance considerations, touched upon in this section, lead us to conclude that two candidates are likely choices for SRP FT1: the Aerojet R-40B or an appropriately sized LOX/CH_4 engine. Both of these could provide the desired performance at lower cost than the other available candidates.

7. CONCLUSIONS

Five flight test article concepts were considered for a proof of concept flight test of SRP. They were designed such that they could be packaged and delivered to appropriate test conditions on a sounding rocket. Each concept was deemed capable of decelerating through a sufficient range of velocity and generating sufficient thrust for such a test. In addition, a large range of test article trajectories was identified that is capable of satisfying test phase requirements. The performance and packaging assessment of the flight test article concepts indicate that a sounding rocket is a viable platform for a proof of concept test of SRP. The cost and availability of propulsion hardware remain as large discriminators among the concepts and will be critical for future concept refinement and down selection.

REFERENCES

1. Zang, T. A., and Tahmasebi, F., "Entry, Descent and Landing Systems Analysis Study: Phase 1 Report," NASA TM 2010-216720, July 2010.
2. Braun, R. D., and Manning, R. M., "Mars Exploration Entry, Descent and Landing Challenges," *Journal of Spacecraft and Rockets*, Vol. 44, No. 2, March-April 2007, pp. 310-323.
3. Adler, M., Wright, M., Campbell, C., Clark, I., Engelund, W., Rivellini, T., "Entry, Descent, and Landing Roadmap: Technology Area 09," NASA Office of the Chief Technologist, November 2010.
4. Edquist, K. T., Dyakonov, A. A., Korzun, A. M., Shidner, J. D., Studak, J. W., Tigges, M. A., Kipp, D. M., Prakash, R., Trumble, K. A., and Dupzyk, I. C., "Development of Supersonic Retro-Propulsion for Future Mars Entry, Descent, and Landing Systems," AIAA 2010-5046, June 2010.
5. Korzun, A. M., Braun, R. D., and Cruz, J. R., "Survey of Supersonic Retropropulsion Technology for Mars Entry, Descent, and Landing," *Journal of Spacecraft and Rockets*, Vol. 46, No. 5, September-October 2009, pp. 929-937.
6. Goddard Space Flight Center, "NASA Sounding Rocket Program Handbook", June 2005.

8. ACKNOWLEDGEMENTS

The authors would like to acknowledge the support of the Exploration Technology Development and Demonstration (ETDD) Program, managed at NASA-Glenn Research Center. The work documented herein was performed as part of ETDD's Entry, Descent, and Landing (EDL) Technology Development Project, which is managed at NASA-Langley Research Center and supported by NASA-Ames Research Center, NASA-Johnson Space Center, and the Jet Propulsion Laboratory. The authors would also like to acknowledge Art Casillas, Jeremy Shidner and Bill Studak for their analysis support and guidance, as well as the following individuals from Wallops Flight Facility for their support of this work: Justin Babcock, Brian Hall, and Scott Schaire. Part of this research was carried out at the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration.