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Concept and Preliminary Flight Testing of a Fully Reusable Rocket Vehicle

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A fully reusable rocket vehicle is proposed to demonstrate good operability characteristics both on the ground and in flight. For achieving technical readiness for future space transportation systems, design considerations not only for higher-performance-related issues but also those for good operability are needed. The proposed vehicle is to be used as a sounding rocket and has the capabilities of ballistic flight, returning to the launch site, and landing vertically, making use of clustered liquid-hydrogen rocket engines. Before the development of this type of reusable rocket was initiated, a small test vehicle with a liquid-hydrogen rocket engine was built and flight tested. A demonstration of vertical landing and exercise of turnaround operation for repeated flights are the major objectives of the test vehicle. Two flights were performed in succession, and the flight-test operation provided many valuable experiences for designing the fully reusable rocket vehicle.

Introduction

A NUMBER of performance-related innovations are needed for future space transportation systems such as fully reusable launch vehicles. High-performance engines and super-lightweight materials and structures are the technologies needed to be ready to build these future vehicles. In addition to these technical challenges, it is necessary for the flight and the ground operations of these advanced vehicles to be closer to those of aircraft. It is believed to be a key to an order-of-magnitude cost reduction of the transportation between ground and low Earth orbit, which is a major element of the whole space infrastructure of the future. Current expendable launch systems prevent us from achieving easy access to space because of their ammunition-type operations. The discussion here concerns the space transportation architecture that could be achieved by the current technology for the next 20–30 years.

Many proposals and studies have been conducted in the past and are being conducted in the present that involve the selection of propulsion systems, staging policies, and vehicle system configuration. Reference 1 recommended the rocket single stage to orbit (SSTO) as the most realistic and a market-competitive candidate system for covering world demand for the next 30 years, assuming that the current launch demand does not change drastically. The SSTO is a vehicle that conducts its flight by itself like an aircraft without staging and expending any part of the vehicle. From a technical point of view, the rocket-propelled SSTO has become realistic through the current maturation of lightweight materials and structures, but no such vehicle has yet been demonstrated.

Changing the launch culture should be a key to the future of space launch systems. This means that the current launch culture of using expendable launch systems is not that for transportation but is an extension of warhead launch or ammunition. These characteristics result in a few launches per year, thousands of standing armies for each launch preparation, and a long waiting time and delay, each of which automatically costs a lot. An order-of-magnitude or more cost reduction is possible only by a change in the culture. Only a system like an aircraft will achieve this goal of reduction by its simplified operation, orbit on demand, quick turnaround, and continuous intact abort capabilities. These features are so-called aircraft-type operations in comparison with the current existing launch systems. The degree of reusability is categorized into A, B, and C, after Ref. 2. Class A stands for the reusability of aircraft, B is that of the U.S. STS orbiter, and C is that of the solid rocket boosters of the STS. The distinction between class A and class B is that the shuttle orbiter is refurbished, inspected, tested, and certified after each flight as if it were newly built, whereas the aircraft is obviously not. True reusability like this is achieved only with tremendously low-cost operation, as is done with aircraft.

With the above-mentioned necessary characteristics taken into consideration, a conceptual vehicle system for a reusable sounding rocket is proposed, and this preliminary design work will show that these necessary and essential aspects would be common to the types of fully reusable space transportation systems to be built. In addition to these design considerations, a small test vehicle was built and flight tested in order to gain experience on how reusable vehicles should be designed and operated. Because the capability of the vehicle is limited, the present study focuses on the landing flight characteristics and turnaround operation of a vehicle of this kind.

Reusable Sounding Rocket and Design Considerations

Proposed System Overview

With the above considerations in mind, a rocket vehicle was preliminarily designed.[3,4] The first goals of the vehicle were to achieve full reusability and enhanced operability and to demonstrate the benefits of good reusability. At the same time, the rocket vehicle was used as a sounding rocket. Easy access to flight opportunities is quite important for users such as astrophysicists and researchers of the upper atmosphere. The microgravity community is also a potential user of the vehicle. By enhancing flight operability and achieving low-cost operation, the vehicle will give a good opportunity for these researchers and users, which means that frequent use of the vehicle is expected by them. Because the development of an orbital vehicle is a huge business, starting from a small vehicle with good reusability would be one of the effective paths to these goals, and flight opportunities for the vehicle would potentially be beneficial to users. These are the background ideas for the present study. In addition, new technologies necessary for a future vehicle like the SSTO, such as an altitude-compensation nozzle and new materials, require in-flight demonstration, because it is very difficult to qualify these new technologies with ground-based facilities. Thus the test vehicle will be also beneficial for technical studies.

As for rocket performance, the proposed vehicle has a ballistic flight capability to an altitude up to 300 km, safely returning to the

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launch site. Finally, it lands vertically and provides safe return of the payload without any difficulty in recovery, unlike conventional sounding rockets. As presented in the preceding section, a quick-turnaround capability will support repeated use by the user and several flights in one launch campaign would be made possible. A design goal for the turnaround interval should be daily flight.

For a higher-lift-to-drag-ratio vehicle, such as a winged vehicle that makes a relatively shallow entry into atmosphere, it is difficult to return to the launch site after ballistic flight without a propulsion system for the return cruise flight. A deep reentry flight path angle by a winged vehicle will impose major difficulties in both wing loading limitations and significant attitude change maneuvers in the descent. Preparing separate launch and landing sites many hundreds or thousands kilometers apart will not be beneficial for effective reusability. Because one of the objectives of the present system is to show good reusability, a flight operation with a single ground base is a major necessary condition. Thus a vertical-lander-type system was chosen for the present study. A nearly vertical flight makes it possible to perform the return flight to the launch site, and a vehicle with relatively low lift-to-drag ratio is capable of this type of flight. For range safety reasons, the vertical ascent was to be made 10–15 km off from the launch site. Preliminary flight analysis showed that a base-entry vehicle with a 0.2–0.3-lift-to-drag ratio can achieve aerodynamic return flight to the launch site.[3] Then the vehicle restarts the engines for final approach and landing flight without an attitude turnover from its reentry flight to vertical landing. In addition to these flight capabilities, one of the benefits of the vertical lander is potentially to simplify ground operations. Although the horizontal-landing rocket vehicle has to be reoriented into a vertical position for the next launch, the vertical-landing vehicle has the advantage of reducing necessary ground operations and facilities. This advantage will contribute greatly to quick turnaround and is another reason why the vertical lander was chosen.

Figure 1 presents the resulting vehicle with fuselage removed to show its internal structure. Its propulsion system is composed of four liquid-hydrogen/liquid-oxygen engines, and it lifts the vehicle vertically like a conventional rocket; then the vehicle reaches an altitude of 300 km. After falling down into the atmosphere, it is aerodynamically decelerated and lands vertically at the same place from which it was launched, as shown in Fig. 2. The aerodynamic shape of the vehicle is designed so as to perform this aerodynamic return flight. Because the base heating is not significant because of relatively low aerodynamic heating at its entry into the atmosphere, the holes for engines on the bottom surface are shielded by appropriate movable covers during its aerodynamic descent flight. The payload capacity to the 300-km altitude is 100 kg, which is equivalent to the current sounding rocket of the Institute of Space and Astronautical Science (ISAS), Japan. Accelerations during the flight are up to $5g$ during ascent and $15g$ of deceleration during atmospheric reentry. The vehicle's dimensions and specifications of the engines are summarized in Table 1.

Because of the benefit of the deep throttling capabilities of each engine, the vehicle has flexible maneuverability in the flight, e.g., hovering at a constant altitude, low-speed powered flight, and so on. For research in the middle and the upper atmospheres, there has not been a good tool for observation and sampling. This is because balloons never reach an altitude higher than 40 km and conventional sounding rockets pass through at supersonic speed, which degrades measurement and quality of samples of the composition of atmosphere and aerosols. Therefore the flight maneuverability presented will enhance sounding and sampling opportunities greatly. Worldwide Earth environmental awareness will encourage the sounding of the middle and the upper atmospheres and could potentially be a driver for frequent use of the vehicle. In addition, quick turnaround and repeated flight make it possible to conduct qualitatively different observations. One of the difficulties in measurements by rocket is the preparation of the space-qualified instruments onboard, which is far different from doing things in a laboratory. Even for current sounding rocket campaigns in comparison with the use of satellites, flight opportunities for research are far fewer than desired. Easy access to space flight is expected to satisfy these demands, and reusability enables us to easily launch more sophisticated and expensive instruments. The targeted turnaround concept for the vehicle is shown in Fig. 3. To operate the vehicle as shown in the figure, many new design considerations on vehicle subsystems for good operability, as well as considerations for new aspects of the whole system architecture presented in the following sections, are needed. Good operability of the fully reusable rocket vehicle will satisfy these demands for easy access to space flight. Then the benefit of the vehicle with class A reusability, as described, is demonstrated.

Table 1 Dimensions and engine specifications of the reusable sounding rocket

Parameter	Value
Vehicle	
Length, m	4.4
Body diameter, m	2.2
Liftoff mass, kg	3800
Landing mass, kg	1460
Propellant mass, kg	2400
Payload mass, kg	100
Maximum altitude, km	300
Engine	
Propellant	Liquid oxygen/liquid hydrogen
Specific impulse, s	350 (sea level)
Thrust, kN	16.7 (sea level)
Dray mass, kg	70
Throttling, %	100–30

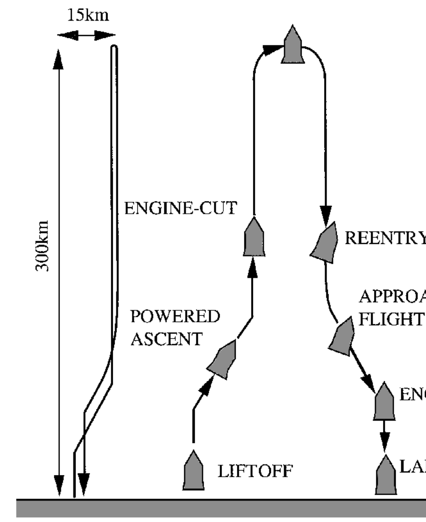


Fig. 2 Flight sequence for the reusable sounding rocket.

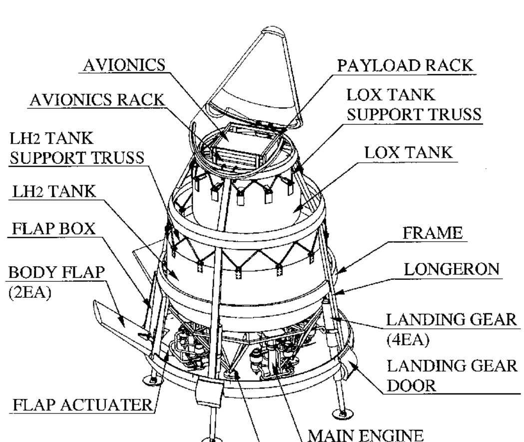


Fig. 1 Proposed reusable sounding rocket: LOX, liquid oxygen.

Considerations for Safe Flight Abort

The U.S. STS has its safe flight-abort procedure. However, the STS does not have a continuous intact abort capability, particularly

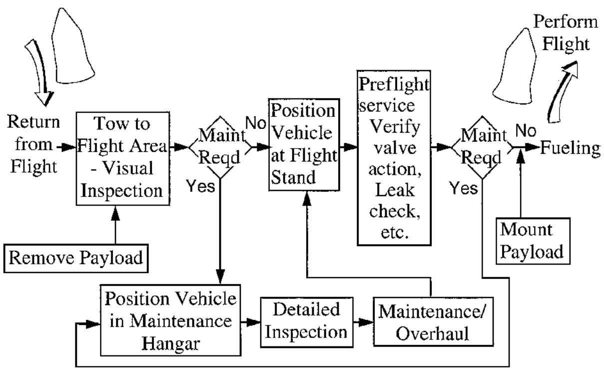


Fig. 3 Turnaround operation concept for the reusable sounding rocket.

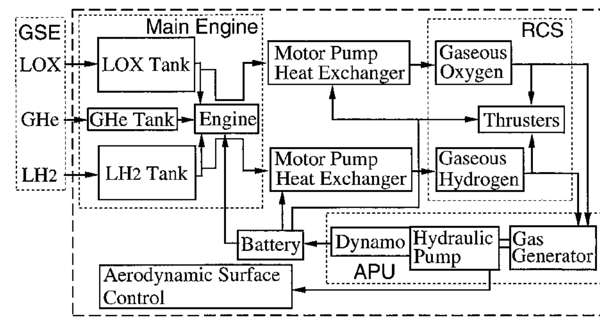


Fig. 4 Ideal propulsion/energy integration. GSE, ground support equipment; LOX, liquid oxygen; GHe, gaseous helium; LH2, liquid hydrogen.

at its initial ascent phase, because of the solid-propellant boosters.[5] For the in-flight operational aspect of the present vehicle, a continuous intact abort capability is stressed. Because of the nature of the rocket-propelled accelerator vehicle, a wide range of flight conditions should be taken into consideration; however, a safe return in a one-engine-out condition at any time during its flight is to be a requirement in this regard. To do this, a safe abort requirement is transformed into requirements on the vehicle hardware, such as the number of rocket engines, their throttling capabilities, attitude controllability, failure detection, and so on. Either one engine's or both engines' thrust and aerodynamic forces and moments are used for maintaining safe flight in a one-engine-out flight condition, and differential throttling of the main engines for producing attitude control moment will enhance this flight capability.

The proposed vehicle uses four engines for ascent, and it uses two in symmetrical positions for landing by throttling down to ~30% of maximum thrust level. The vehicle's landing gears are designed so as to bear the static and dynamic load on final vertical landing. In the case of a one-engine-out condition during powered ascent flight, the vehicle immediately cuts the opposite engine. Then the vehicle continues the ascent flight by using remaining two engines and consumes the fuel. Then it performs the return flight. Trajectory analysis shows that ascent by two engines will result in a lower maximum altitude than that of nominal flight. However, it reaches up to 100 km in altitude and above, even if the one-engine-out condition occurs during its initial ascent. The lower maximum altitude above 100 km does not influence the return down range during atmospheric flight greatly, which means that the vehicle safely returns to the launch site even in a one-engine-out failure mode.

When one engine is out at a powered landing, the vehicle immediately cuts the opposite engine and switches to the sound pair of engines, because a pair of engines in symmetrical position is used for the landing as described above. To be ready for the one-engine-out condition, all four engines must be waiting for start before landing, and besides operating two engines in nominal landing flight, the remaining pair of engines is kept ready to start at any instant during powered landing flight. The vehicle propulsion and attitude control requirements are established according to these failure conditions throughout entire flight regime, including liftoff, ascent, return flight, and final vertical landing.[4] A study of an expander cycle engine with deep throttling capability and dynamic throttling responses was conducted. A preliminary firing test with an existing engine characterized the throttling capabilities,[6] and the result will be applied to the present vehicle.

Integrated Propulsion/Energy System

The operability of a hydrogen-fueled aircraft was extensively studied in the past,[7] and the Delta Clipper Experimental (DC-X) and XA vertical-landing demonstrator showed that repeated flight and turnaround of a hydrogen-fueled rocket vehicle was possible for a test-scale vehicle.[8] Referring to these vehicles' fueling systems, we point out another important aspect for the good operability: to simplify and unify the fuel systems much more than those of current expendable vehicles. The fuel system includes those for primary propulsion, auxiliary propulsion such as a reaction control system (RCS), and power generation. An ideal way is to use liquid hydrogen and liquid oxygen for all of the onboard power and propulsion subsystems. Such an integrated system will make it possible to reduce the number of high-pressure gas bottles and to eliminate toxic fuels for the RCS and power generation. The system is composed of primary propulsion, fuel and oxidizer tanks, the RCS, a power generator such as an auxiliary power unit (APU) or fuel cells, a gassing system with heat exchanger, and their management subsystems. A tradeoff should be made to evaluate the benefit of this integrated system in comparison with independent subsystems such as those of present expendable vehicles. The integrated system will help simplify the post-flight and preflight ground operations. It results in a safe and quick-turnaround capability for the vehicle.

Considering these degrees of integration, many possibilities were chosen and compared with each other.[4] An example of an ideal integrated propulsion/energy system is presented in Fig. 4, in which liquid hydrogen and oxygen are used for primary and auxiliary propulsion and for power generation. Before engine start on the ground, external power will be needed for gassing the liquid hydrogen and oxygen. In addition, the system should have a wide and flexible operational range in terms of energy balance for various in-flight conditions such as engine on/off and thrust levels. Further work is still needed for the activation procedure for the vehicle, the deactivation process after flight, the way to ensure readiness for the next flight, and so on. Preliminary system analysis shows that the integrated system will work for the present mission primarily because the flight time is as short as 500 s from launch through landing.[4] Although the present concept might not be always true for orbital vehicles like the SSTO, because the size of the vehicle and the flight sequences are different from those of the proposed vehicle, these considerations will give good direction for further studies.

Exercise of the Flight-Test Vehicle

Test Vehicle Overview

A small test vehicle was built and flight tested in order to be partly ready for the design of the vehicle proposed above. This campaign is named reusable vehicle testing (RVT) and will give us an opportunity to exercise new design and operational aspects peculiar to the reusable rocket vehicle. The following two primary objectives are addressed for this RVT exercise arising from many important design issues as presented in the preceding sections: 1) operational lessons in rocket turnaround, and 2) vertical landing of a rocket-propelled vehicle. For designing a fully reusable rocket vehicle, these two topics are among the essential characteristics. It is necessary for the design and the construction of the reusable vehicle to experience these in-flight and on-the-ground operations.

For the required high performance of the future vehicle, liquid hydrogen is the only fuel that gives promise to the rocket SSTO. Regarding the ground operation of the vehicle, a cryogenic fuel will impose essential constraints for operational and safety reasons. Therefore a slightly modified engine based on ISAS's existing liquid-oxygen/liquid-hydrogen small rocket engine is used for the vehicle. It is a regeneratively cooled engine with a continuous throttling capability for making vertical landing possible. Although this engine was not originally designed for reusability, it was estimated that at least 30 firings were possible from the viewpoint of the stress level of its combustion chamber material. The vehicle's fueling systems are designed so as to achieve the repeated flight easily and quick turnaround, e.g., daily flight or two flights per day. The vehicle's dimensions and specifications are summarized in Table 2, and an overview of the vehicle is presented in Fig. 5.

The vehicle's flight navigation, guidance, and control subsystems are built based on the conventional inertial measurement unit (IMU) and a laser altimeter. The attitude of the vehicle is controlled by a cold-nitrogen-gasjet RCS placed on the top of the vehicle, as shown in Fig. 5. The test vehicle has four-leg landing gear that makes it possible to eliminate any launch support or launch tower. The design landing speed is less than 1 m/s and the maximum allowable is 3 m/s from structural and damper stroke limitations. The final landing guidance concept is to let the descending speed follow the predetermined line with respect to the altitude, as described in the subsection flight-test results. At the instant of landing, the engine is to be cut off automatically with respect to altitude and sink-rate measurements. Therefore no landing sensors or ground-contact sensors are required. The telemetry subsystem sends the measured status of propulsion subsystems as well as other onboard subsystems, and this status is monitored by the ground operator. Emergency commands are to be sent when an anomalous status is found. Then the vehicle's flight mode is to be transferred to emergency landing or to stop the engine when range safety issues would occur.

Table 2 Specifications of the RVT test vehicle

Parameter	Specification
Vehicle	
Dry mass, kg	320
Height, m	3.0
Engine	Liquid oxygen/liquid hydrogen
Thrust level	2750–3920 N (throttling by liquid-oxygen feed)
Propellant	Liquid hydrogen, 100 liters; liquid oxygen, 42 liters
Flight time, s	20 (maximum)
Attitude control	Gaseous nitrogen RCS
Navigational source	Inertial measurement unit + laser altimeter
Landing gears	Fixed + damper
Turnaround	Two flights/day (target)

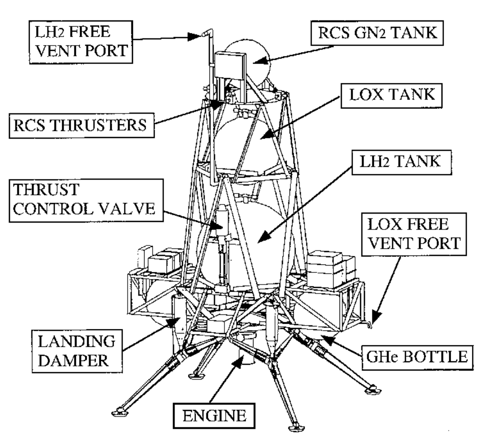


Fig. 5 RVT test vehicle.

Propulsion Subsystem

The engine is a pressure-fed system for the sake of simplicity. The fuel and oxidizer tanks are pressurized from helium bottles on both sides of the vehicle, as shown in Fig. 5. Figure 6 is a schematic diagram of the primary propulsion system in which all the valves and external interfaces such as fuel loading, venting, draining, and pressurizing ports are indicated. These external interfaces are designed so as to give good operability in both preflight and postflight handling requirements, as described later. Fuel and oxidizer tank pressures are kept constant by regulation of the helium-gas supply. The regulated pressures are 2.2 and 2.9 MPa for the liquid-hydrogen tank and the liquid-oxygen tank, respectively. Engine throttling is done by controlling the flow rate of the liquid oxygen alone for simplicity, too. As a result, the oxidizer/fuel mixture ratio varies from 3 to 7 as a function of the level of the thrust. To chill down the feedlines of fuel and oxidizer to the combustion chamber before ignition, the vehicle uses a free venting to the atmosphere. This also helps eliminate the necessary ground venting lines, quick disconnect couplings, and associated equipment. The fuel for chilldown is freely drained into the air horizontally from the top of the vehicle, as shown in Figs. 5 and 6. Liquid oxygen for chilldown is vented downward from the side of the vehicle, and it is drained ~5 m apart from the vehicle through a pipe on the ground in order to keep a distance between the freely dumped fuel and oxidizer.

To achieve a safe and soft landing, both static and dynamic engine throttling characteristics are required. The engine is a part of the guidance and control plant of this flight-test system. Taking into account the results of landing flight analysis, a 1-Hz and higher response thrust control requirement was imposed on the dynamic throttling characteristics of the engine. Because of the simplicity of the pressure-fed system, it was not very difficult to satisfy this requirement by choosing the appropriate proportional valve and electrically driven actuator. Before to flight testing, two series of engine-firing tests were carried out. The engine's ignition transient and static and dynamic throttling responses were characterized. The resulting frequency response of the engine thrust is given in Fig. 7, in which the response of the thrust with respect to the throttling command is summarized. The response is represented in the form of a Borde chart as is done for the normal control element. We took the data by giving oscillation to the throttling command by changing its frequency and amplitude in the ground firing tests. It was found that the response was sufficient to conduct the landing flight, and these characteristics were modeled and used for guidance and control analyses. We also predicted the dynamic behavior by an engine simulation model, as indicated in Fig. 7, by modifying the model developed through the throttling test of the pump-fed system presented earlier.[5] During these firing tests, engine-firing-oriented mechanical environments for the vehicle structure and onboard instruments were also identified. In the last stage of these firing tests, a vehicle stand-alone firing was made. By the test, the overall functions of the onboard subsystems were qualified and the mechanical

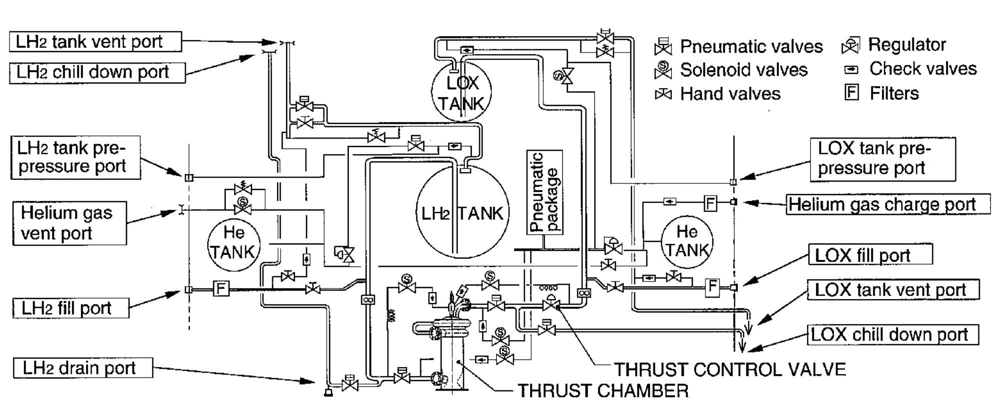


Fig. 6 RVT primary propulsion system.

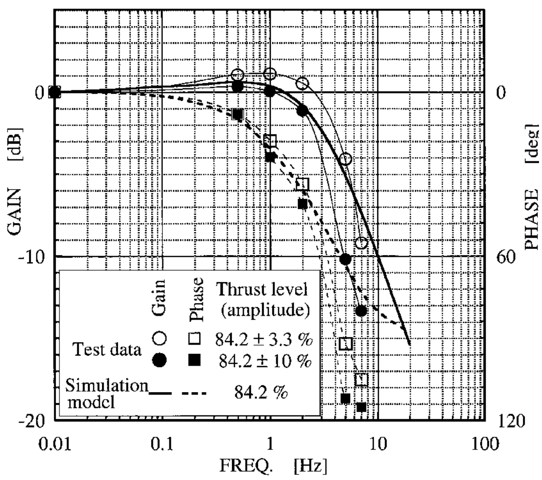


Fig. 7 Dynamic throttling response characteristics.

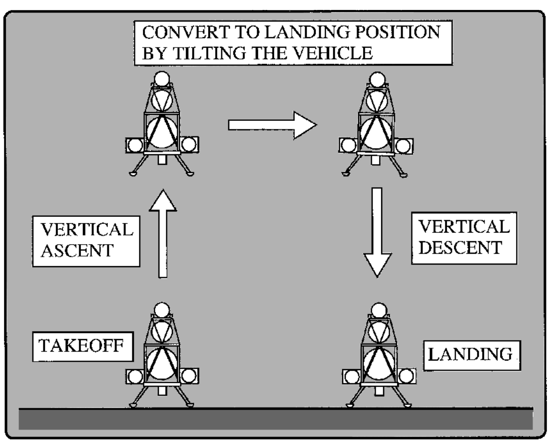


Fig. 8 RVT flight sequence.

and thermal environments were characterized, particularly from the viewpoint of the mechanical and thermal ground effects at vertical takeoff and landing.

Flight Sequence and Onboard Status Monitoring

A planned flight sequence is shown in Fig. 8, in which the sequence is divided into four modes: takeoff and vertical ascent, change of the vehicle's position, vertical descent, and final landing. The change of the vehicle's horizontal position is accomplished by tilting the vehicle's attitude. For each mode of flight and on the ground, an onboard status monitoring system is readied and activated. Unlike other conventional expendable vehicles, the test vehicle has an intact abort capability by which safe abort at any instant of the flight and transition to a safe landing are executable. This abort capability is also one of the new characteristics of the reusable vehicle. The onboard monitoring system checks the status of the engine, pressurized tanks, RCS, and onboard computer itself. The system automatically calculates the predicted values and allowable ranges of several important state variables, such as engine combustion pressure and temperatures of critical parts of the engine. These in-flight estimates were done with respect to the level of the engine throttling in which the transient responses are also taken into account, because the dynamic throttling range is wide. Once one of these monitored variables travels out of the allowable tolerance, the vehicle automatically transitions into the safe landing mode.

Readiness for Flight Testing

An extensive flight analysis was carried out from the landing dynamics and safety points of view. At the instant of the landing ground contact, the vehicle might fall down or crash to the ground when there is unexpectedly large horizontal motion and/or angular motion. Taking into account many sources of uncertainty such as wind effect, structural alignment and center-of-gravity offset, navigation sensor errors, and so on, a series of landing motion analyses was performed. It was found that uncertainty in the friction force between the ground and the landing leg's pad is important according to the analysis. The vehicle's navigation/guidance system and its performance were qualified by a preflight vehicle-in-motion simulation, in which the vehicle was lifted up and down to the ground, imparting the same vertical speed as that of flight. All the subsystems of the vehicle except the engine were activated in the test, which was the final verification before the flight test. After completion of these preflight tests and analyses, the flight test was conducted in March 1999 at ISAS's Noshiro Testing Center (NTC). Following the functional tests and the engine's static firing tests in a tied-down configuration on the ground, two flights were carried out. In the second flight, as shown in Fig. 9, full vehicle motion as presented in Fig. 8 was performed. The flight time was 11.5 s and the maximum altitude was 4 m. These two flights were conducted within 2 days and the interval between flights was 23 h.

Flight-Test Results

The vehicle performed the flights as planned, and the vehicle's propulsion and navigation/guidance/control systems worked as expected. Figure 10 shows the history of flight altitude and the engine's

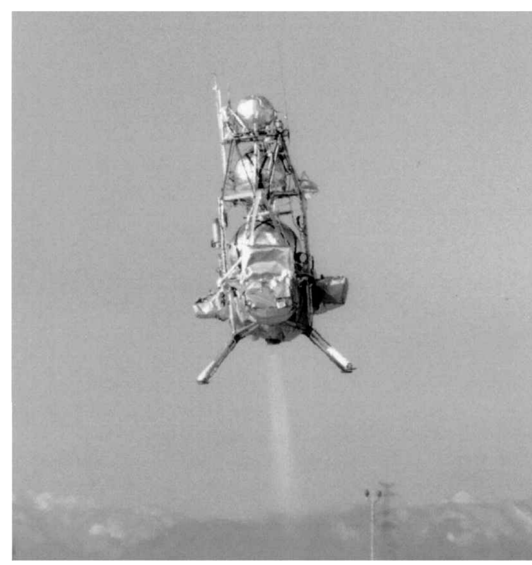


Fig. 9 RVT in flight.

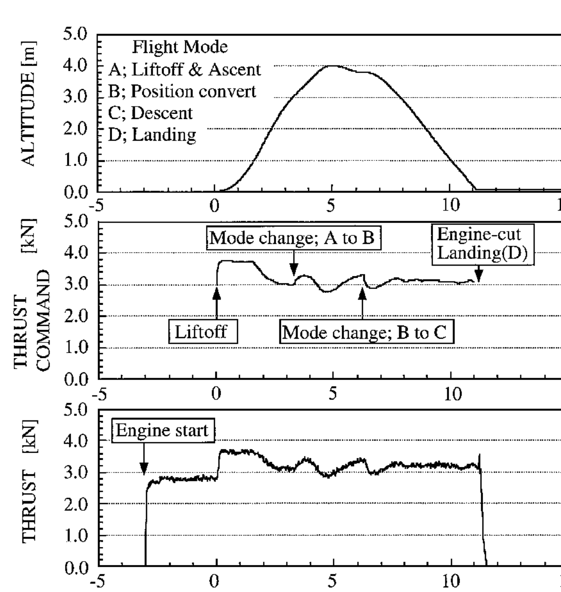


Fig. 10 Flight altitude, engine throttling command, and resulting thrust in the second flight.

thrust for the second flight test. The value of thrust is estimated from the measured combustion chamber pressure. The horizontal distance from takeoff to landing was 4 m. Because of the short flight time and the vehicle's fuel limitations, transition to each flight mode occured before the vehicle's position and attitude were settled very well, but the vehicle did perform a planned flight and landing. The engine started 3 s before liftoff, the flight mode transition from liftoff to landing was done as presented in Fig. 10, and engine throttling was well controlled with respect to the command from the navigation/guidance systems. Throttling response with respect to the command was also as expected from the prescribed dynamic response presented in Fig. 7. For the landing, the vehicle maintained an almost constant sinking rate, and engine throttling was properly controlled during the vertical descent mode, as shown in the altitude history of Fig. 10. Figure 11 shows the resulting landing speed with respect to the altitude before landing. It shows that the landing guidance made the vehicle follow the predetermined lines as described earlier. In the present flight, the landing guidance target was set to keep almost constant landing speed, as shown in Fig. 11. A slightly larger engine thrust than expected resulted in the lower landing speed, as shown in the figure. It was found that the deviation of engine thrust was caused by the slightly higher regulation of fuel tank pressure, but it was within tolerance. As a result, the landing impact was smaller than anticipated, and the dampers of four legs almost did not stroke. The engine was cut off automatically with respect to the measured altitude almost at the instant of ground impact, as planned.

The onboard status monitoring system worked properly, but the monitor did not detect any anomalous values, so the vehicle never experienced the transition to the safe landing mode during the flights. Figure 12 shows an example of in-flight status monitoring of hydrogen-gas temperature at the fuel injector, which is a good indicator of the soundness of the combustion chamber and regenerative cooling status. Among many on-board monitoring items, the combustion chamber pressure and the fuel temperature at the injector were activated to be ready to make the transition to safe landing mode in the present flight test. The estimated values with respect to the throttling command were based on real-time calculations with a simplified engine model dynamics stored in the onboard computer. The allowable range of tolerance was defined before the

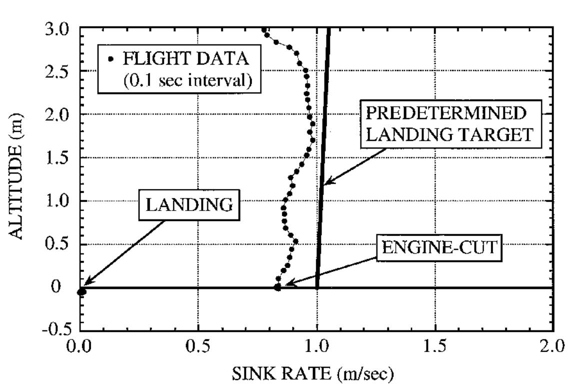


Fig. 11 Landing sink-rate history with respect to flight altitude.

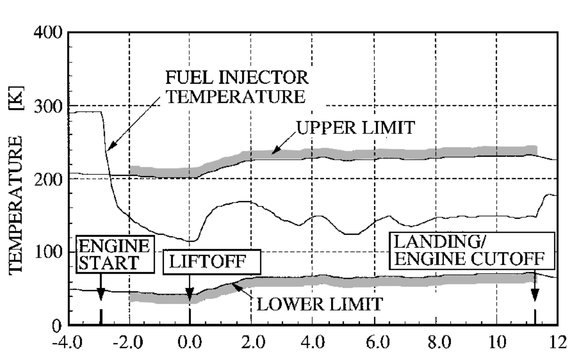


Fig. 12 Example of status monitoring: hydrogen-gas temperature at fuel injector.

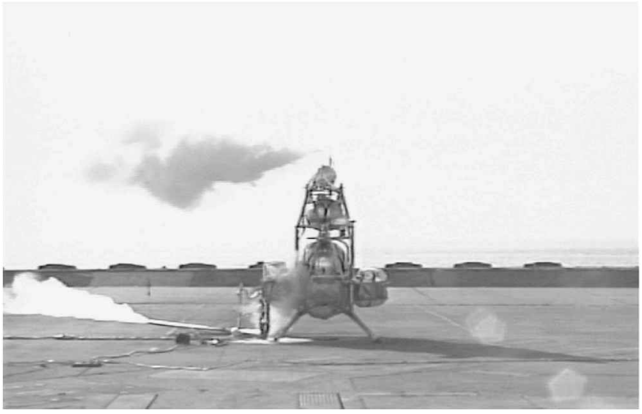
flight. Transient response that was due to engine throttling was taken into account in the estimates, as described. As is seen in Fig. 12, it did not travel out of the tolerance, which shows that the system was sound during the flight. These real-time estimation models and parameters in the models were carefully tuned during the ground firing tests.

The free venting of the fuel and the oxidizer before the flight for chilling down the feedlines for ignition was shown to be in good shape. The final automatic chill down sequence lasted 17 and 11 s for liquid oxygen and liquid hydrogen, respectively. Then they were cut off at 0.5 s after the engine's ignition. The flow rates for chilldown were 1.4 liters/s for liquid hydrogen and 0.2 liters/s for liquid oxygen. Depending on the ground wind direction and speed, the vented hydrogen gas stays around or flows out of the vehicle. Counterventing against the ground wind direction was avoided by a change in the direction of the free venting port of hydrogen. Figure 13 shows a series of views of the free venting, engine ignition, and liftoff in the second flight test. The cloud on the ground on the lower-left-hand side of each figure is that of drained oxygen, and the one on the upper left-hand side exhausted from the top of the vehicle is vented hydrogen. Figure 13a shows free venting before the start of the engine, and Fig. 13b shows the instant of engine start. In Fig. 13c the free venting ends and the lower part of the hydrogen cloud starts to burn, but the rest of the hydrogen cloud dissipates before the time of liftoff. Then the vehicle lifted off, as shown in Fig. 13d. In the present flight, the ground wind direction was from right to left in the figure and the wind speed was ~2 m/s on average. In some other engine firings, the hydrogen cloud was burned up by the engine's exhaust at its ignition, as observed in the pictures, but not in other cases. As a result, there was no serious impact to the vehicle or its flight at all in any engine ignition and firing, including the ground firing tests throughout the campaign.

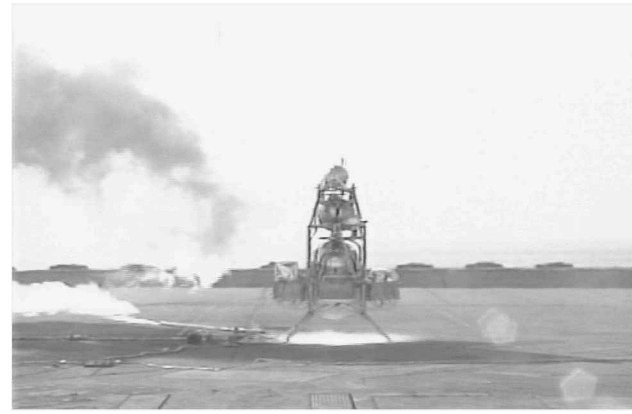
Operational Lessons and Turnaround

Two flight operations and the ground firing tests in the campaign were made on a daily basis, as presented earlier. The turnaround of the vehicle was conducted as shown in Fig. 14. The flight operation was initiated when the vehicle was transported to the liftoff point from the hangar. After completion of electrical functional tests, the accumulators of the primary propulsion system and the RCS were pressurized. Following the liquid oxygen, the liquid hydrogen was loaded. From the beginning, it took 2 h for a crew of three working next to the vehicle to ready the propulsion subsystems. After the evacuation of the engine crew to a safe distance, the fuel and oxidizer tanks were pressurized remotely. Initializing the navigation system, the countdown operation started the engine chilldown. Thirty seconds before liftoff, the valve control authority of propulsion system was handed over from the ground operator to the onboard computer. Three seconds before takeoff, the engine was started at its minimum thrust level. Just before liftoff, an umbilical cable was removed; then the vehicle lifted off. These operations are quite similar to those of the normal rocket launch except for the free venting.

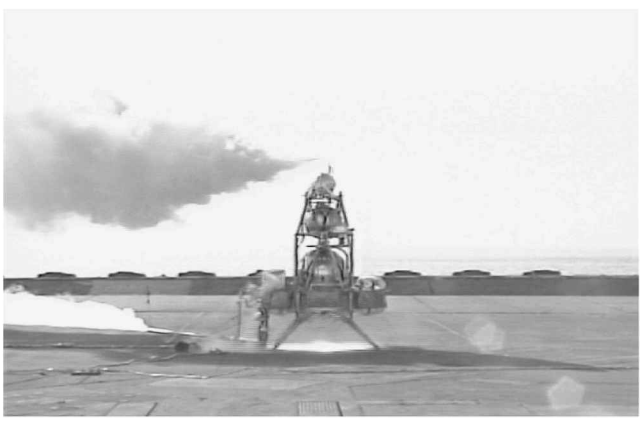
Immediately after landing and engine cutoff, a purging to prevent the engine from freezing was started automatically from the onboard helium accumulators. After the flight, the status of each of the subsystems was inspected remotely by means of telemetry and visual monitors, and the fuel and oxidizer tanks were depressurized to safety levels by commands that were sent. Then the engine crew came to access the vehicle and reconnect the ground support lines for fuel dumping and external power and signal supply. Then the valve control authority returned to the ground. It took ~10 min for these postlanding activities to make the vehicle safe. In case of emergency such as telemetry loss of signal or other safety-related anomalous status, the vehicle is equipped with pressure gauges for tanks and high-pressure bottles to help operators deactivate the system in such a case. The residual fuel dumping was done by connecting the ground vent lines and a vent stuck away from the vehicle. These operations were also done by the three-member engine crew. After 30 min of fuel dumping was completed, the vehicle became totally safe, and the deactivation of the vehicle and a series of postflight inspections were started. It was timelined within 4 h to conduct all the activity from the beginning. These operations were all done



a) Free venting before engine start (X-00:06:02)



c) End of free venting (X-00:00:99)



b) Engine start (X-00:02:98)

d) Liftoff (X+00:02:07)

Fig. 13 Free venting at engine ignition and liftoff.

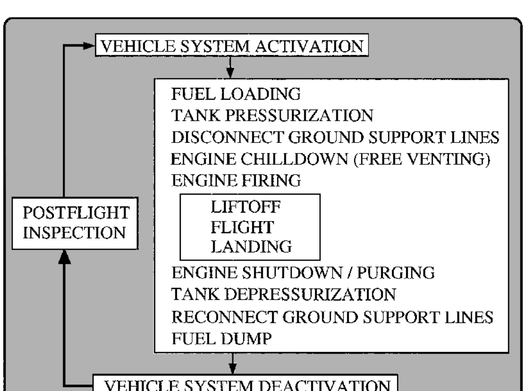


Fig. 14 Actual turnaround operation.

safely. As for turnaround and readiness for the next flight, a visual inspection of the engine and a leak detection test of the propulsion subsystem were done by sealing the nozzle exit after each flight. As a result, the test system, including the vehicle itself and the ground support, functioned as planned and expected, and daily turnaround was achieved.

Conclusions

Taking into account how to contribute to the future rocket vehicle, a fully reusable sounding rocket was conceptually proposed and preliminarily designed. Several essential technical issues such as continuous intact abort and integrated propulsion/energy systems were technically assessed, and these new characteristics will be important for achieving future vehicles with good operability. Performance-related issues could also be incorporated and flight demonstration of technical items such as lightweight materials and structures and propulsion systems will be made possible by the proposed vehicle. Aiming at such a fully reusable vehicle, a small test vehicle was built and flight tested. The test vehicle is small, and its performance is far less than that of the targeted system. However, the present study gave us a precious opportunity to design, build, and operate a new type of the rocket vehicle, without which we can never experience repeated use of the rocket. Many lessons were learned about vertical-landing dynamics, onboard architecture that supports safe flight abort and landing, and turnaround operation for repeated flight of a vehicle of this kind.

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