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The Stanford Sounding Rocket Program was established to demonstrate, promote and grow a new hybrid fuel technology, namely, liquefying hybrids. To this end, three sounding rockets in various stages of development are described. The motivation for this program is that at laboratory scale and in scale-up ground tests, a paraffin-based hydrocarbon fuel has shown great promise in that it burns three to five times faster than conventional hybrid fuels. The impact of this high regression rate fuel is that it is now possible to design safe, compact, high performance hybrid rockets that are competitive with liquid and solid rockets in a variety of applications.

Introduction

The hybrid rocket has been known for over 50 years, but wasn't given serious attention until the 1960's (Refs. 1-3). The primary motivation for developing the hybrid was the non-explosive character of the fuel, which led to safety in both operation and manufacture. The fuel could be fabricated at any conventional commercial site, even at the launch complex with no danger of explosion. Thus a large cost saving could be realized both in manufacture and launch operation. Additional advantages over the solid rocket are; greatly reduced sensitivity to cracks and de-bonds in the propellant, higher specific impulse, throttle-ability to optimize the trajectory during atmospheric launch and orbit injection and the ability to terminate thrust on demand. The products of combustion are environmentally benign unlike conventional solids that produce acid forming gases such as hydrogen chloride.

Early hybrid rocket development and flight test programs were initiated both in Europe and the U.S. in the 1960's. The European programs in France and Sweden involved small sounding rockets, whereas the American flight programs were target drones, (Sandpiper, HAST, and Firebolt) which required supersonic flight in the upper atmosphere for up to 5 minutes. These latter applications were suitable for the conventional hybrid because its low burning rate was ideal for a long duration (i.e. sustainer) operation.

Despite the low regression rate of the fuel, in the late 1960's Chemical Systems Division of United Technologies (CSD/UTC) investigated motor designs of larger diameters that could produce high thrust suitable for space launch vehicles. They experimented with a 38 inch diameter motor delivering 40,000 lbf. of thrust. In order to achieve a high fuel mass flow rate, a 12-port fuel grain was required. Although the motor was successfully fired several times, it was recognized that the volumetric fuel loading efficiency was compromised, which would lead to deficiencies in vehicle performance.

Interest in the hybrid was revived again in the late 1970's when concern was expressed about the storage and handling of the large solid propellant segments of the Shuttle booster. The storage of potentially explosive grains is costly in terms of requirements for reinforced structures and interline distance separation. The same safety concern arose again after the Challenger disaster, when it was recognized that a thrust termination option might

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have avoided the failure. This concern was heightened when, a few months later, there was a Titan failure, caused by an explosion of one of the solid boosters.

In the last decade or so, two significant hybrid efforts have occurred. One was the formation of the American Rocket Company (AMROC); an entrepreneurial industrial company devoted entirely to the development of large hybrid boosters. The second, with encouragement from NASA, was the formation of the Hybrid Propulsion Industry Action Group (HPIAG) composed of both system and propulsion companies devoted to exploring the possible use of hybrids for the Shuttle booster and other launch booster applications. Both efforts ran into technical stumbling blocks, basically caused by the low regression rate fuels, which resulted in large diameter motors with many ports to satisfy thrust requirements. The resulting configuration not only compromised potential retrofit for the Shuttle and Titan boosters but also raised questions about internal ballistic performance of a thin web multi-port motor, especially toward the end of burning. Although AMROC had many successful tests in 51-inch diameter motors, they ran into difficulties when the motor was scaled to 6 foot diameter and 250,000 lbf. thrust. The low regression rate of the fuel dictated a 15 port grain design and problems of excessive diameter and grain integrity resulted. In 1995 AMROC filed for bankruptcy.

Several hybrid propulsion programs were initiated in the late 80’s and early 90’s. The Joint Government/Industry Research and Development (JIRAD) program involved the testing of 11 and 24 inch diameter hybrid motors at the Marshall Space Flight Center. Another hybrid program initiated during the early 90’s was DARPA’s Hybrid Technology Options Project (HyTOP). The goal of this program was to develop the HyFlyer launch vehicle and demonstrate the feasibility of hybrid boosters for space applications. The members of the HyTOP team were AMROC, Martin Marietta and CSD/UTC.

The Hybrid Propulsion Demonstration Program (HPDP) began in March 1995. The goal of the HPDP was to enhance and demonstrate several critical technologies that are essential for the full-scale development of hybrid rocket boosters for space launch applications. The government and industry participants in the program are NASA, DARPA, Lockheed Martin, CSD/UTC, Thiokol, Rocketdyne, Allied Signal and Environmental Aeroscience Corporation. Even though the tasks of the HPDP program included systems studies and subscale testing, the main objective of the program was the design and fabrication of a 250,000 pound thrust test-bed. The design of the motor was guided by the subscale motor tests performed under the JIRAD program. The wagon wheel 7+1 multi-port fuel grain is made of conventional hydroxyl-terminated-polybutadiene (HTPB)/Escorez fuel. The motor was fired for short times in July 1999. The motor exhibited large pressure oscillations and unequal burning rates in the various ports. Problems related to low regression rate inherent in conventional hybrids fuels were not solved and we are not aware of any further activity planned for the HPDP program.

On Dec 18, 2002, Lockheed Martin launched a 60,000-lbf thrust multi-port HTPB/AL sounding rocket from NASA Wallops Island Flight Facility. The rocket was 57-feet long, 2-feet in diameter and is the largest hybrid to ever fly. The rocket did not reach the target apogee altitude apparently because of a partial structural failure of the multi-port fuel grain. Pieces of unburned fuel were found in the vicinity of the launch pad. Even so, this LOX fed vehicle represents a significant advance in hybrid rocket technology.

Currently, one of the most significant hybrid development effort in industry is the development of a propulsion system for SpaceShipOne, (an X-prize entry being built by Scaled Composites Inc.) A competition has been set up between eAc and SpaceDev that will result in a downselect of a single hybrid propulsion system. The systems being studied are nitrous oxide fed HTPB-based hybrids.

The conclusion from this history is that if a significantly higher burning rate can be realized for the hybrid motor, the difficulties mentioned above can be greatly reduced and a smaller, more efficient motor can be obtained. Although this deficiency was recognized early on, attempts to increase the burning rate by more than 50-100%, without compromising the safety and low cost features, have been largely unsuccessful until recently.

A fast burning, long chain hydrocarbon-based (non-polymeric, i.e. paraffin) hybrid fuel has been developed and successfully tested at Stanford University in a lab-scale motor (Ref. 4). The results indicated regression rates 3-4 times larger than the rates of the conventional polymeric fuels. Scale-up testing (i.e. 10” motor OD and up to 3000 lbf of thrust) of the same fuel, performed at NASA Ames Research Center, confirmed the high regression rates observed in lab-scale testing (Ref. 5).

These newly identified high regression rate fuels burn in a fundamentally different way than the slow burning evaporative-diffusive dominated combustion process of conventional hybrid fuels. During combustion, a thin, hydro-dynamically unstable liquid layer forms on the melting surface of the fuel. Entrainment of droplets from the liquid-gas interface substantially increases the rate of fuel mass transfer leading to much higher surface regression rates than can be achieved with conventional polymeric fuels (hence, the newly coined term "liquefying hybrids"). High regression rate is a natural attribute of the fuel material and the need for oxidizing additives or other regression rate enhancement schemes is avoided. These fuels provide Isp performance comparable to kerosene but are approximately 20% more dense than kerosene.

This permits the design of a high volumetric loading single-port hybrid system with a density impulse comparable to or greater than a hydrocarbon fueled liquid system. The high regression rate hybrid removes the need for a complex multi-port grain and most applications up to large boosters can be designed with a single port. The fuel contains no toxic or oxidizing components and can be shipped by commercial freight as a non-hazardous commodity. Hence, we believe that the liquefying hybrid technology can be the basis for hybrid rockets with simple grain designs, reduced cost, reduced complexity and increased performance; one that can compete favorably with conventional solid or liquid systems in a variety of applications.

With the satisfactory results obtained from ground tests (more than 50 tests have been conducted at moderate scale and approximately 150 at lab-scale of paraffin-based fuels with several oxidizers), we have decided to demonstrate this new fuel technology's feasibility and the improvements that it will introduce in the field, with some small scale sounding rocket flight tests. A three phase approach has been taken to the flight program.

Phase 1: This phase involved the flight of a small sounding rocket (i.e. 2 1/4" OD) based on a commercially available kit hybrid motor, Aerotech RMS/Hybrid 54, (Ref. 6). The Aerotech unit was modified to utilize the new fast burning fuel. The development of this small and low performance prototype system was the first step to prove the fundamental idea and also to guide the development of the more involved Phase 2 and Phase 3 technology demonstrator (TD) motors. Several flights (apogee of approximately 5800 ft.) were made in October 1999.

Phase 2: In this more involved phase we are developing a larger scale sounding rocket based on a hybrid motor specifically designed and fabricated to operate with the new fast burning fuel. The Phase 2 technology demonstrator rocket is optimized for our baseline hydrocarbon-based propellant and is designed to reach a much higher altitude of roughly 88,000 ft. The first flight of this rocket is scheduled for September 2003.

Phase 3: This rocket is currently in the preliminary design phase. It is a sounding rocket that is recoverable and reusable. It has been designed to have a performance level comparable to the Orion (a solid rocket commonly used by NASA's Sounding Rocket Program). Aside from being simply a technology demonstrator, this recoverable rocket will provide several unique capabilities to the sounding rocket community, including throttle-ability and ultimately thrust vectoring. We are currently attempting to secure funding for this ambitious effort.

We have selected nitrous oxide as the oxidizer for all three phases of the program because of its self-pressurizing capability, non-cryogenic nature, safety and commercial availability (Ref. 7). All these properties of N_2O make it a good candidate for simple, inexpensive systems with moderate performance requirements. The major shortcomings compared to the other commonly used oxidizers such as LOX are low performance (low Isp), high optimal O/F, (oxidizer to fuel ratio) low density and temperature sensitivity. Some of these deficiencies can be ameliorated by the addition of energetic materials such as aluminum.

The basic designs and progress to date are described in the following sections. Reported are the results of Phase 1 flights, detailed information concerning the development of Phase 2 and the Phase 3 sounding rocket conceptual design.

Phase 1 Sounding Rocket

Phase 1 motor development

The Phase 1 development spanned a four-month period wherein a small sounding rocket was designed built, tested and flown. This was the first flight of a hybrid rocket using a paraffin-based fuel. The first prototype propulsion system was based on a commercially acquired Aerotech hybrid model kit, RMS/Hybrid 54. The most important modification to this system was the replacement of the original cellulose-based fuel material used by Aerotech with the paraffin-based fuel formulation.

The RMS/Hybrid 54 system consists of a 54mm motor casing and a high pressure aluminum alloy cylindrical oxidizer tank, connected via a forward closure/injector plate assembly. The oxidizer tank is used for storing the liquefied N_2O, a self-pressurizing oxidizer that reaches a pressure of 760 psi, at 68^o F. The injector plate assembly gives the flexibility of using 2 to 4 injector ports via screw on devices. The system is ignited via a pyromatch and a forward solid fuel grain where a Pyrovalve™ (housed in the forward closure) is combusted to allow the flow of nitrous oxide.

To confirm the expected performance of the propulsion system, a small ground test stand was built and instrumented for thrust measurements.

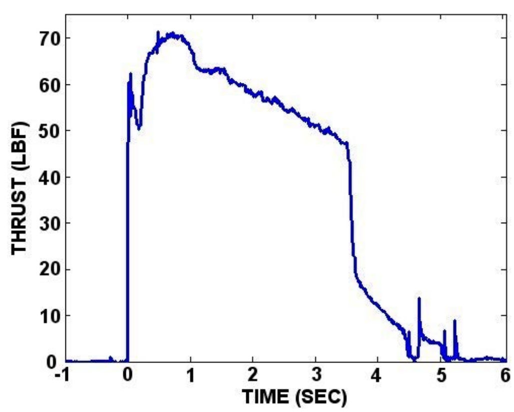

Fig. 1. Thrust – Time data for ground test #2.

Prior to the flight tests, three ground firings of the modified Phase 1 motor were performed. After the flight tests, an additional 11 ground firings were done and a regression rate law was established for the paraffin based fuel and N_2O oxidizer combination. Figure 1 shows the thrust-time curve for ground test number 2. The declining nature of the thrust time history is caused by the drop-off in the N_2O pressure in the oxidizer tank as the oxidizer is expelled and also high erosion rate of the nozzle throat, which was made of a thermoset plastic. The measured thrust-time profiles were in good agreement with the predictions.

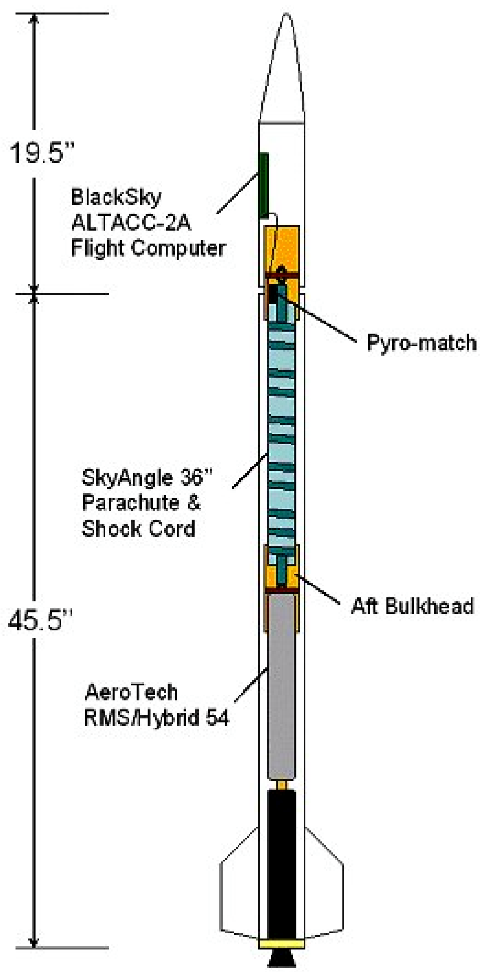

Fig. 2 Phase 1 sounding rocket design (dimensions in inches).

Phase 1 Sounding Rocket

The Phase 1 motor was integrated into a 2-inch OD, 65-inch long sounding rocket with an aluminum fairing. This rocket successfully completed 3 test flights in October 1999 at the Black Rock Desert, Nevada. Two of these flights utilized the new hydrocarbon-based fuel, and one flight test was conducted with the original cellulose–based Aerotech fuel for comparison purposes. The Phase 1 rocket main systems and instrumentation were based on commercial off-the-shelf components. Figure 2 shows the sounding rocket structural design and the main components.

The main rocket structural assembly consists of two independent aluminum fairings friction coupled through a phenolic coupler that houses the front-bulkhead. The front portion is attached to a plastic 9-inch long nose cone that houses the flight computer

(BlackSky AltAcc-2A). The aft portion is divided into two main compartments via the aft-bulkhead. The first compartment houses a 36 inch diameter parachute and shock cord recovery system (SkyAngle) and the second compartment is where the Aerotech hybrid model kit - RMS/Hybrid 54 is inserted

The flight instrumentation system (AltAcc-2A) consists of an altimeter and accelerometer to measure rocket performance and a timer that was used to initiate deployment of the recovery system. The parachute deployment sequence is controlled via a starting-descent/maximum-altitude-reached check flag. With this signal, a pyro-match is used to ignite gun powder to build enough pressure for separation of the friction coupled two main body pieces. This separation enables the deployment of the parachute. During descent, the two main rocket pieces are connected via a shock cord to which the parachute is attached with free rings.

Flight data was written to non-volatile computer memory for download upon recovery of the rocket. This data set enabled the determination of the rocket altitude, thrust and vehicle velocity throughout the entire flight. The performance data for these three flights are listed in Table 1. Shown in figure 3 is the liftoff of flight number 1.

Table 1. Phase 1 sounding rocket flight performance.

	Flight 1	Flight 2	Flight 3
Fuel	Paraffin-based 1	Cellulose-based	Paraffin-based 2
Max Altitude, ft	5,800	5,359	4,455
Max Acceleration, g's	7.97	8.49	9.19
Max Velocity, ft/sec	637	585	597
Rocket Gross Weight, kg	3.294	3.280	3.388

Fig. 3. Flight of the first paraffin fueled hybrid rocket (October 1999).

Table 2. Phase 1 sounding rocket-operational data for first paraffin-based fuel flight

Flight number	1
Flight Date & Time	10/09/99 - 11:45 am
Ambient Temperature	71 ^oF
Ambient Pressure	665.5 mm Hg
Motor Weight Before Firing	1.562 Kg
Rocket Empty Weight	1.732 Kg (No motor)
Rocket Gross Weight	3.294 Kg
Rocket c.g.	52.96" (from the front)
Rocket c.p.	43.48" (from the front)
Stability Margin (at liftoff)	9.48"
Ox tank empty weight	0.532 Kg
Ox tank loaded weight	0.860 Kg
Port Diameter	1.0 inch fore, 1.39 inch aft
Grain Outside Diameter	1.835 inch
Total Grain Length	9.67 inch (7.925 inch aft)
Total Grain Weight	0.169 Kg
Total Grain weight (after burn)	0.080 Kg

The second paraffin-based flight, which was the third flight of the day, showed reduced performance because of increased aerodynamic drag exerted on the rocket. This occurred because the parachute failed to deploy on the second flight causing a slight misalignment of the fuselage on impact. The flight data for this launch, which includes acceleration (g), velocity (ft/sec) and altitude (ft) is shown in Figure 4.

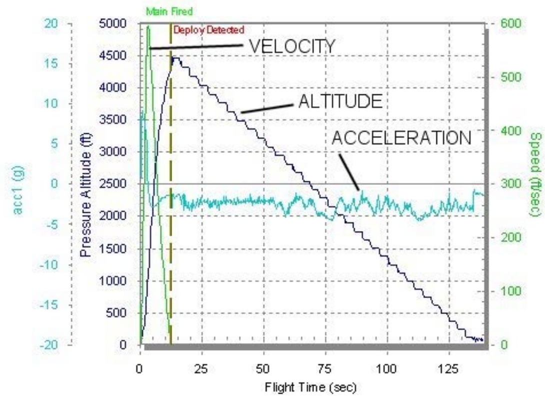

Fig. 4. Flight data from flight 3.

Phase 2 Sounding Rocket Design

Aside from being substantially larger than the Phase 1 rocket, the Phase 2 sounding rocket design is the near optimal configuration required for operation with a fast-burning hydrocarbon-based propellant and N_2O oxidizer. The targeted maximum altitude is approximately 88,000 ft. The diameter of the rocket is 7 inches and the overall length is close to 12 ft. The gross weight of the rocket is 150 lbf. The design and fabrication of the Phase 2 rocket required a much more substantial effort. In fact, several sophisticated design tools were developed to predict the performance of the system and to optimize the configuration within the budget constraints.

One such tool is an integrated simulation program that involves the integration of three major elements of the rocket flight: propulsion, aerodynamics and vehicle dynamics. This approach enabled us to analyze and make necessary changes of system characteristics based on simulations of the powered and unpowered parts of the flight. The propulsion element involves aspects of the combustion process that defines the thrust produced by the system. The main propulsion elements are the oxidizer tank system, feed system and the motor internal ballistics. The aerodynamics element provides the necessary aerodynamic database for the surfaces mounted on the rocket. Finally, vehicle dynamics element ties the dynamics of the rocket flight with the propulsion and aerodynamics.

A schematic of the Phase 2 layout is shown in figure 5 and selected design specifics are listed in table 3. Presented in figures 6-14 are predicted time histories of the motor performance and Phase 2 rocket flight. In these figures, burnout (defined as the point in time when the liquid oxidizer is consumed) occurs at 12 secs and apogee is reached 75 secs. into the flight. The maximum acceleration that the vehicle experiences is 7.5 g and the maximum Mach number is 2.25. The plots, showing the propulsion system parameters, have two distinct characteristics. At the point where burnout occurs, residual N_2O gas remains in the oxidizer tank that is consumed for approximately 12 seconds after burnout producing a reduced but useful thrust level.

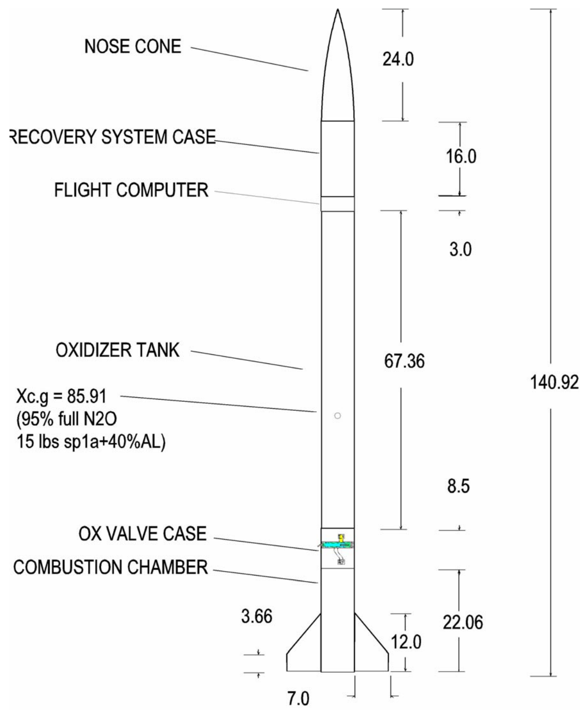

Fig. 5. Layout of the Phase 2 rocket.

Table 3. Phase 2 design specifics.

Parameter	Value
Combustion chamber O.D., in.	7
Rocket overall length, in.	141
Burn time (liquid oxidizer), sec	12
Burn Time (gaseous oxidizer), sec	12
Propellant Mass, lbm	70

Fuel mass, lbm	15
Vehicle Liftoff Weight, lbf	146.4
Apogee Altitude, ft	88,000
Initial oxidizer mass flow rate, kg/sec	1.9
Average O/F	4
Peak Mach number	2.25
Maximum thrust, kN	5
Maximum acceleration, g	7.7

Phase 2 Predicted Flight Performance

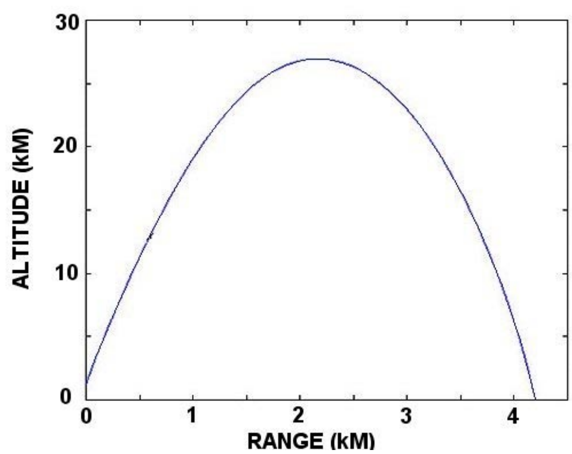

Fig. 6. Altitude versus range (Phase 2).

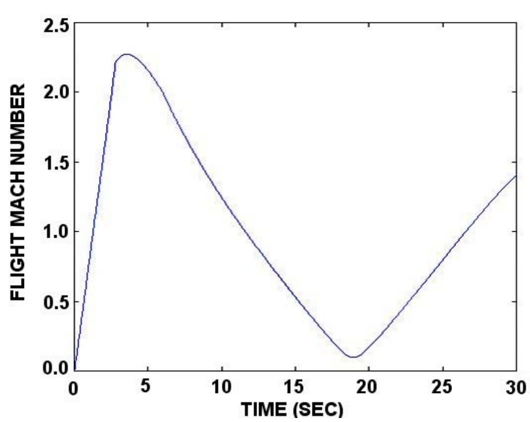

Fig. 7. Predicted Mach number time history (Phase 2).

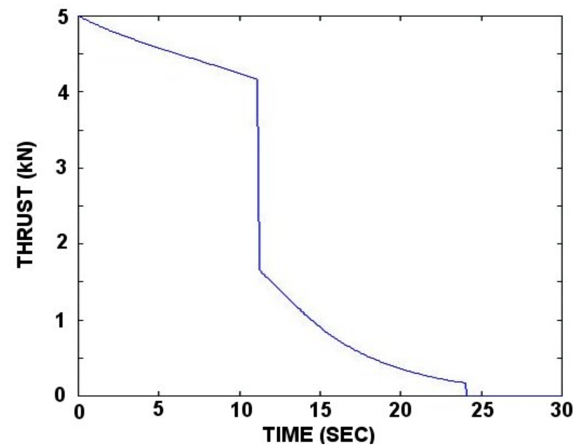

Fig. 8. Predicted thrust time history (Phase 2).

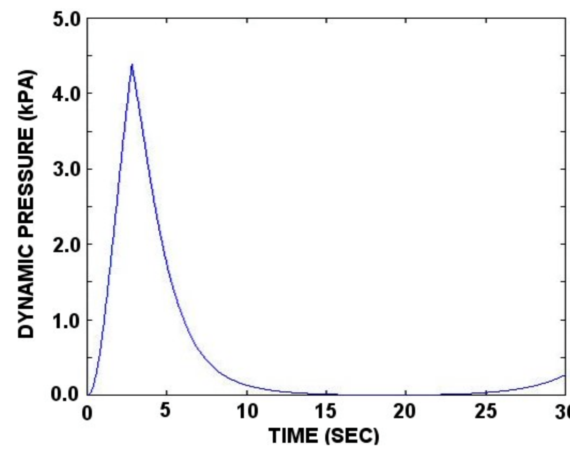

Fig. 9. Predicted dynamic pressure time history (Phase 2).

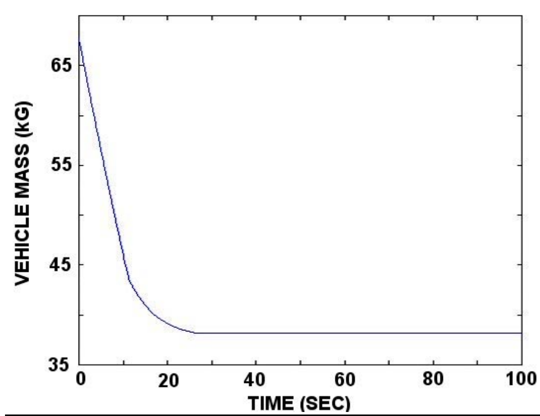

Fig. 10. Predicted vehicle mass time history (Phase 2).

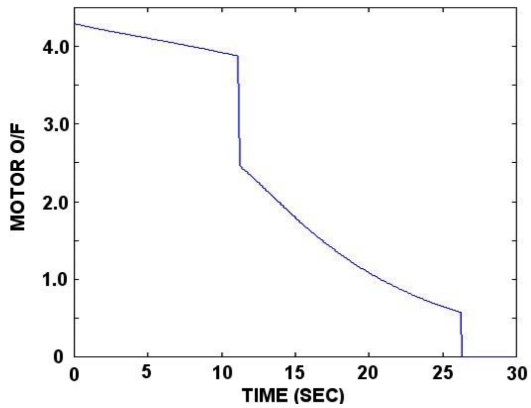

Fig. 11. Predicted motor O/F time history (Phase 2).

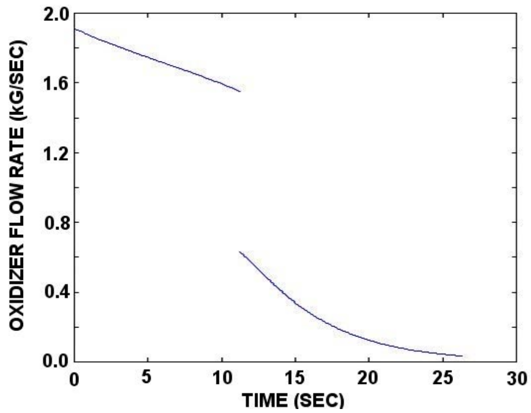

Fig. 12. Predicted oxidizer flow rate time history (Phase 2).

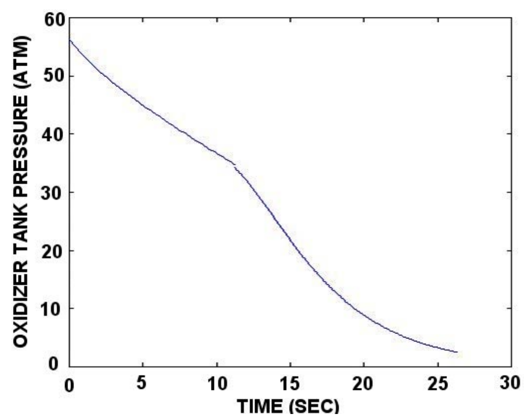

Fig. 13. Predicted oxidizer tank pressure time history (Phase 2).

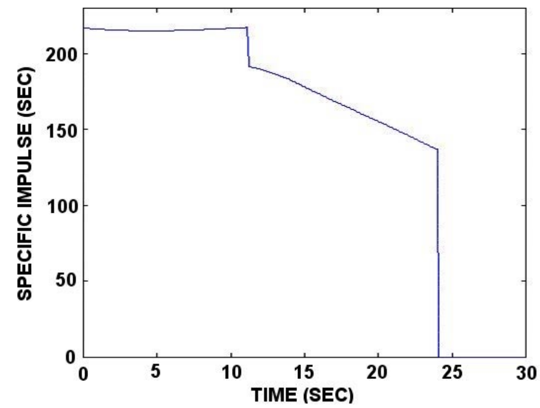

Fig. 14. Predicted specific impulse time history (Phase 2).

Motor Design

The Phase 2 Technology Demonstrator motor has been designed, fabricated and tested. A cut-away of the combustion chamber is shown in Figure 15. This motor operates at a starting chamber pressure of 450 psi, average O/F of 4 and initial oxidizer mass flux of 500 kg/m^2-sec. Note that because of the high regression rates of the fuel, the combustion chamber length to diameter ratio is less than 3, a value quite low for conventional hybrids.

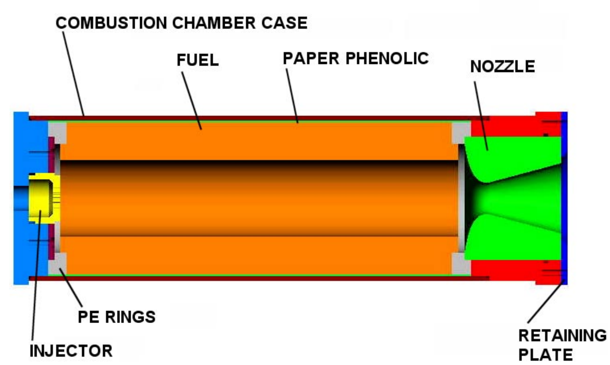

Fig. 15. Cutaway of the Phase 2 combustion chamber.

The combustion chamber casing is manufactured out of 6061-T6 aluminum with several unique features including threaded end plates with corner o-ring seals. A grain cartridge is loaded by simply unscrewing the aft motor plug and sliding the grain into the combustion chamber casing. The combustion chamber wall is 0.160 inch thick and was designed with a safety factor of 2 (threads are the weakest point). The combustion chamber was hydro-tested at a pressure of 900 psi. No leaks were observed.

The nozzle material for Phase 2 is ATJ graphite. This material is not optimal as far as weight or erosion is concerned but cost considerations guided this choice. The nozzle area ratio that maximizes the altitude was determined to be 4.5. The nozzle is sandwiched between the aft Polyethylene fuel grain ring and a retaining plate that is bolted to the aft end of the motor.

Two types of injectors were designed, namely, a simple showerhead and a reverse-flow slash plate. Both kinds of injectors will be tested and the configuration that achieves the most stable and efficient combustion will be selected.

The fuel grain design is similar to that used previously (see ref. 5). It consists of a paper phenolic cylinder with a polyethylene ring bonded at each end. The paraffin is centrifugally cast in the phenolic tube using a small casting facility at Stanford University.

Phase 2 Oxidizer Tank

An extensive search was undertaken to find an off-the-shelf pressure vessel that met the design requirements for the Phase 2 rocket. All of the tanks identified had length-to-diameter ratios that were too low. As a result, it was decided that a custom tank would have to be designed and fabricated.

The design strategy for the oxidizer tank was to design a reasonably light aluminum tank that could be fabricated at minimal expense and would have a minimum safety factor of 2. The requirements were that it contains 25 Kg of N_2O at a maximum expected operating pressure of 850 psi. Since the length-to-diameter ratio is approximately 9.5, the weight advantages of hemi-spherical end caps are minimal and a substantial cost saving could be realized by using flat end caps. Shown in fig. 16 are the dimensions of the flat end cap, the V-grove full penetration weld and a portion of the 0.160-inch thick shell (Note that dashed line is the tank centerline).

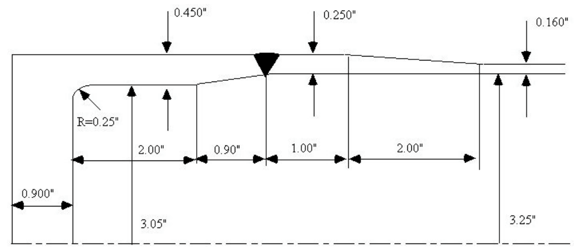

Fig. 16. Phase 2 oxidizer tank (end-cap detail, dimensions in inches)

The wall thickness variation is gradual and the inside corners have a radius to minimize stress concentrations. The outside wall of the tank is turned on a lathe to reduce the wall thickness from 0.25 inch to 0.160 inch resulting in a weight savings of nearly 5 Kg.

The tank is made of 6061-T6 aluminum, with a maximum diameter of 7.0 inches, an overall length of 67.36 inches and has a ¾ inch NPTF thread in the motor end of the tank and a 3/8 inch NPTF thread in the opposite end cap for fill line and relief and purge valve connections. The tank has an internal volume of 2159 $inch^3$, an empty weight of 14.6 Kg and has a capacity to hold 25 kg of N_2O.

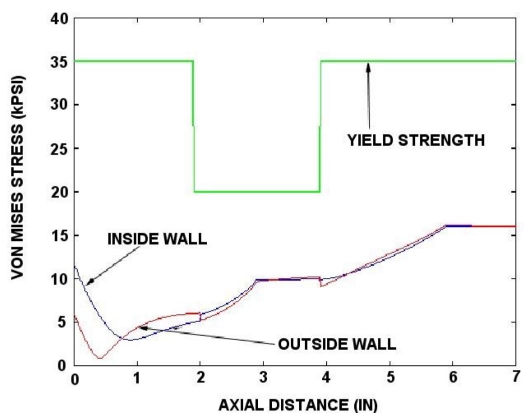

Fig. 17. The Phase 2 oxidizer tank von Mises stress distribution.

Presented in fig. 17 are the von Mises stresses that arise at the operating pressure. The weakest point in the tank is in the vicinity of the weld (a safety factor of 1.96) so weld quality is critical. The reduced yield strength presented in figure is the suggested yield strength for aluminum material in the vicinity of a weld. The tank will be hydro-tested at 950-psi pressure in the near future.

Oxidizer Valve

The oxidizer valve is designed to be reusable and is constructed of materials compatible with nitrous oxide and several other oxidizers. Upon actuation by a pyrotechnic charge, the valve latches in the open position. A schematic of the valve is shown in Fig. 18.

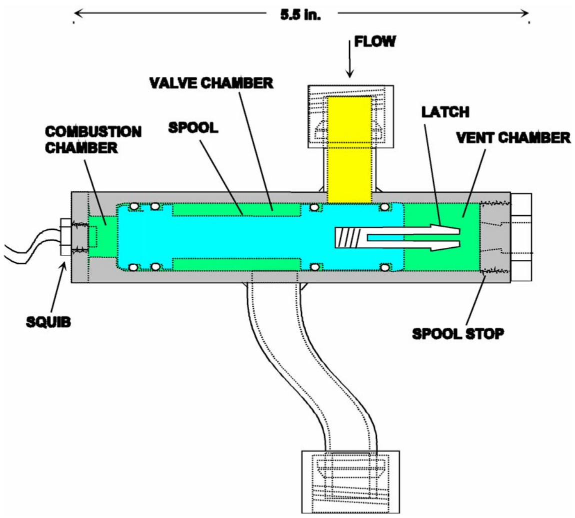

Fig. 18. Pryo valve schematic (valve in closed position).

The main body of the all-steel pyro valve is made out of SA 479 type 304 bar stock. The valve has a flow coefficient (C_v) of 7.2, weighs 1.66 lbf and has 5/8 in. Swagelok tube connections. The spool stop and latch were heat treated for added strength and resistance to deformation. One interesting thing to note about the design is that the valve orifice (minimum cross-sectional area) is located in the 5/8 inch feed tube so the valve should be less susceptible to cavitation than other designs that have substantial pressure drops across the seats of the valve (e.g. a globe valve).

The pyro-valve was hydro-tested to 1200 psi and has been certified with a MAWP of 800 psi. With further testing, (not performed to avoid the possibility of damage to the valve) a significantly higher MAWP could be established. Alternately, this valve could be made substantially lighter by using aluminum but it was decided that safety and reliability were overriding factors.

In comparison to many pyrotechnically actuated devices, the pressure required to actuate the current design is relatively low and an optimal off-the-shelf squib charge could not be located. Rather than using a squib that was too powerful, a custom charge was manufactured in house.

The custom made oxidizer valve squib contains 0.025 gram of FFFFg black-powder (BP) contained in a 3/8-24 hex head bolt that has been drilled out. The BP is sealed in the squib housing by a layer of epoxy applied over a thin layer of protective wadding. The pressure cartridge is ignited by a 30 gauge nichrome wire that is silver soldered to solid 24 gauge copper lead wires and positioned in the charge.

Phase 2 Ignition System

The hybrid rocket ignition system is a steel wool system that minimizes complexity and rocket inert mass while maintaining reliability and robustness. The ignition controller uses three dual one-shot timer chips (NTE74LS123) coupled with three dual Master-Slave J/K flip-flops (NTE7476) to generate ignition activation signals. The one-shot chips are wired with potentiometers to enable different timing configurations. All circuitry is contained in a project box connected to an external power supply. A locking switch and flip-flop arming/disarming ensure several levels of safety. All valves and relays in the system are nominally off to ensure system lockdown in case of power failure. The ignition control system sends signals to four main ignition components: 1) igniter voltage 2) GOx solenoid valve 3) main nitrous valve (a valve used during ground testing for safety) 4) pyro-actuated nitrous valve.

Upon receiving the first control signal, a mechanical relay closes and allows a high level of current to flow through the steel wool. The steel wool is heated in this fashion for a few seconds until the second control signal activates the GOx solenoid. Gaseous Oxygen flows through the flexible tubing, causing the steel wool to burn violently and ignite the paraffin fuel grain. Upon receipt of the third and fourth signals, the main nitrous valve and pyro-actuated valve open, allowing nitrous flow through the main feed line. The entire igniter is ejected from the rocket motor upon nitrous injection, eliminating igniter inert weight on the rocket.

Phase 2 Aerodynamics and Stability

Shown in figure 19 is a component build-up plot of drag at zero angle of attack for the Phase 2 sounding rocket followed by an analysis of the stability of the configuration (center of pressure is shown in figure 20). The aerodynamic predictions were performed using a program called Aerolab. Aerolab is based on the DATCOM method, so it has a fair degree of empiricism but has been shown to give good results in the past.

The parameter space considered includes a Mach range of 0 to 3. Note that the reference area used is the cross-sectional area of the rocket body (i.e. 38.48 in^2). One small launch lug was included in the analysis. The vehicle is designed with only three fins to minimize the fin interference drag. In addition, the fins have straight trailing edges because past experience has shown that fin designs that have surface area extending below the furthest aft portion of the rocket body have been found to be susceptible to damage as the rocket hit the ground during recovery

A good rule of thumb is that for stability, the c.p. should be between 1 and 2 body diameters aft of the c.g. (our body diameter is 7 in.). For static margins less than this, the rocket is unstable, more that this and weathercocking becomes a problem. The stability analysis shows that during the burn, the c.g. shifts from $X_{c.g.}$ = 85.91 in. (X measured from the nose cone tip) at liftoff to $X_{c.g.}$ = 83.11 in. at burnout (a surprisingly small c.g. shift).

The maximum Mach number is expected to be M_∞=2.15 and this occurs at burnout. The c.p. at burnout is at its most forward location. At Mach M_∞ = 1, the c.p. is at it most aft location. Since the c.g. doesn't shift much during the flight, for this rocket, the fin size is a tradeoff between limiting weathercocking at M_∞=1 and maintaining stability at burnout.

The c.p. location varies between $X_{c.p.}$ = 109.0 in. at Mach=1.0 to $X_{c.p.}$ = 92.7 in. at Mach = 2.15. At Mach=0.0, $X_{c.p.}$ = 100.0, So at liftoff, the static margin is about 2.0 body diameters (a good margin). At burnout, (12 secs after liftoff at approximately 20,000 ft) the static margin is 1.37 body diameters. During the ascent as the rocket passes through Mach=1.0 the static margin peaks at about 3.6 body diameters (may weathercock slightly during this phase of the flight). After burnout, the rocket will continue to ascend (to approximately 88,000 ft) and during this period, the c.p. will move from 92.7 through a peak of 110.0 in. to 100 in. just prior to apogee. So the bottom line is that the rocket should be stable throughout the flight and the aerodynamic performance of the fins is expected to be reasonably high.

One failure mode worth considering is the case of non-ignition. Under this scenario, the oxidizer would be expelled creating a thrust to weight ratio slightly greater than 1.0 but no fuel would be burned. The rocket would take off and as the oxidizer is expelled, the c.g. would move from $X_{c.g.}$ = 85.91 in. to $X_{c.g.}$ = 91.02 in. The vehicle would still be stable because the Mach no. would be low (less than 1.0) and $X_{c.p.} \approx$ 100 in.

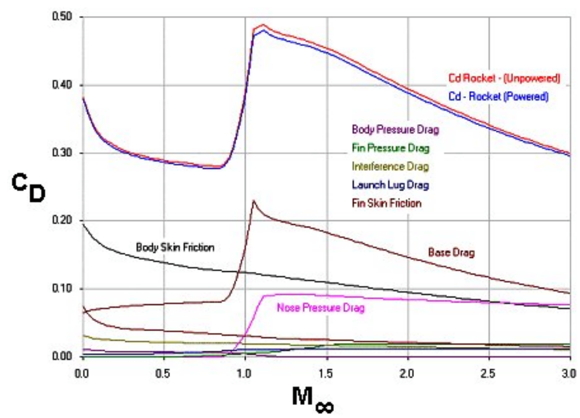

Fig. 19. Phase 2 drag coefficient.

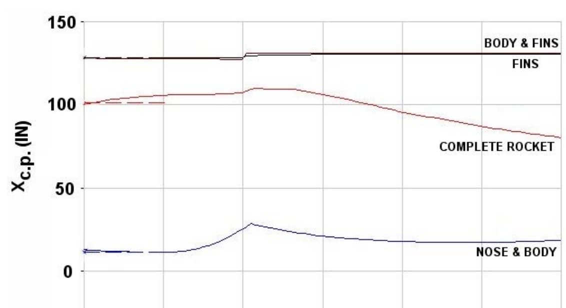

Fig. 20. Phase 2 Center of pressure.

Table 4. Phase 2 weight distribution.

No.	Item	Weight	Xcg	Item Xcg	Overall L
		lbf	in.	in.	in.
1	nose cone	1.21	16	16	24
2	parachute	10	28	8	16
3	recovery system case	1.8	28	9	18
4	flight computer+gps+telemetry+	0.58	41.5	1.5	3
	igniter board+accelerometer board+				
	9vdc radio battery+				
	beacon+2 batteries+				
5	flight computer case	0.5	41.5	2.5	5
6	oxidizer tank	31.87	76.68	33.68	67.36
7	oxidizer (N2O) (full to 95%)	55.12	78.37	32	
8	payload	0			
9	oxidizer valve+squib	1.66	113.76	2.38	6.46
10	ox tank-chamber connect case	1.25	114.61	6.25	12.5
11	fuel (d=2.86" port sp1a+40%AL)	15	128.68	9.79	15.375
12	phenolic tube+ 2 PE rings	1.37	128.68	8.47	16.938
13	combustion chamber fore end +	19.33	128.56	9.7	22.063
	injector+ P.T.+ swage adapter +				
	aft end				
14	Nozzle	3.7	138.47	1.8	3.82
15	fins	3	136.04	13.12	18
	stiffeners				
	bulkheads				
	fasteners				
		146.39			
	Definitions:				
	Item Xcg is distance from most forward point of the item				

Recovery system

A main parachute and a drogue chute will be used to recover the vehicle. The dry weight of the vehicle will be about 76 lbf and so to reduce the impact velocity to less than 15 ft/sec, either a 20-ft diameter conical parachute or a 26-ft cross parachute is required assuming a parachute drag coefficient of 0.9 and a vehicle empty weight of 76 lbf.

The plan is to deploy a 6 ft drogue at apogee (88,000 ft) and let the rocket fall to between 3000 and 5000 ft. above ground level and then deploy the main chute.

A 20 ft conical parachute provides the required 15 ft/sec decent rate at impact, however, this canopy has more weight and packing bulk compared to a cross type parachute. A cross parachute is a more efficient recovery system and is recommended for lower packing bulk and weight, as well as for lower parachute opening loads and deployment at higher speeds. The shock loads on deployment can be 50 to 100 times the static weight of the rocket.

Projected weight for the packed recovery system with main canopy, main deployment bag, drogue, drogue bag, and parachute bridles is 8 lbf to 10 lbf for a 26 to 28 ft cross parachute recovery system. Weight and packing bulk can be minimized by using lightweight kevlar and spectra materials.

Preliminary packed parachute recovery system volume is 375 in^3 to 403 in^3. Packed recovery system length is approximately 12 inches to 14 inches in length for a 4-ft drogue, and 14 inches to 16 inches for a 6-ft drogue in a 7-inch diameter tube.

The parachute system choice (i.e. cross or conical) will be made based on budget considerations closer to the time of the launch.

Flight computer and Data Acquisition

The Rocket Data Acquisition System (RDAS) was chosen as the flight computer for Phase 2. This 8051 CPU based computer was developed by AED in the Netherlands and has been used in many demanding applications including being shot out of a cannon. The complete system includes an altimeter, accelerometers, 6 analogue inputs, 4 digital inputs, and real time telemetry and GPS via a plug-in modules. Data is logged at 200 Hz per channel. The original pressure transducer, used for altitude determination by RDAS, was replaced by a transducer that has a greater range (atmospheric pressure at 88,000 ft is approximately 3 percent of standard sea level conditions).

A radio beacon will be attached to the parachute shroud lines to enable recovery of the rocket in case the unpredictable occurs. The radio beacon chosen is the Rocket Hunter system. The transmitter (with battery) weighs a mere 0.7 ounces. The receiver is hand held and coupled with a Yagi antenna, the system has a minimum range of 5 miles.

Phase 2 Ground Tests

Ground tests of the propulsion system were carried out at a local site using a thrust stand that was specifically fabricated for the Phase 2 motor. The flight combustion chamber was held in a vertical orientation and a fairly long oxidizer feed line, with several in-line components, ran between the combustion chamber and a pair of heavy oxidizer tanks that could hold up to 100 lbf of N_2O.

The test apparatus was instrumented with three identical Omega model PX303-1KG10V pressure transducers. The first, or tank, transducer was located at the union of the oxidizer tank and main feed line, in parallel to a redundant mechanical gauge. The second, or feed, transducer was mounted downstream of the main oxidizer feed and check valves just upstream of the injector. The final transducer was mounted to a pressure tap in the face of the combustion chamber. In addition, two load cells were used to measure motor thrust and weight of the oxidizer tank. All data were acquired using a National Instruments A to D board sampling at 500Hz and LabView software.

Prior to the hot-fire ground tests, several system tests were performed including ignition system and oxidizer feed system tests.

Ignition system tests

A series of tests were run on the igniter system to determine the most effective configuration and timing. A steel wool mass of eight grams was found to provide enough fuel for ignition and reasonably short heat times. Igniters were made by folding the steel wool several times, wrapping it around the end of flexible PVC tubing, and securing with electrical tape. The tape also served to attach 12 Vdc lead wires to the steel wool.

During static motor testing, voltage was applied for approximately five seconds to allow sufficient heating before GOx injection. The GOx flow was sustained for roughly two to three seconds. Nitrous injection was initiated one-half second after the GOx flow was started. This configuration and timing was successfully fired twice during the April (2003) static tests.

Oxidizer Feed System Tests

Several oxidizer cold flow tests were performed in an attempt to precisely quantify the oxidizer mass flow rate. Additional cold flow tests were inadvertently performed after ignition system failures (a bad relay). During the initial cold flow tests, it was found that the oxidizer mass flow rate was less than desired. It was believed that the oxidizer mass flow was reduced because of nitrous oxide cavitation in the main oxidizer feed line at a solenoid valve. Subsequently, this valve was replaced with a pneumatically actuated ball valve in an attempt to solve the problem.

Prior to hot-fire ground tests in April 2003, three cold flow tests of the oxidizer system were performed. The initial cold flow test indicated that the previously observed reduction in mass flow had not been rectified. It was believed that this reduction might have been caused by an excessive check valve pressure drop. A second cold flow test, without the check valve in the feed line, showed no noticeable change in mass flow. The third and final oxidizer system test was performed with an increased injector area. This was done in the hope of eliminating any cavitation that may have occurred at the injector face. Unfortunately, the mass flow rate remained unchanged.

It is believed that ultimately the cavitation problem in the ground test motor configuration will be solved by using nitrogen to pressurize the oxidizer tank slightly above the vapor pressure of nitrous oxide (an augmentation of approximately 50 psi) after filling. This excess pressure will insure that the pressure never drops below the vapor pressure of nitrous in the feed line between the oxidizer tank and the combustion chamber. In the flight-motor configuration, the close coupling between that oxidizer tank and combustion chamber should eliminate cavitation. Further testing is required.

Phase 2 motor tests

It was decided that the observed mass flow, although reduced, would be sufficient to obtain ignition and acquire useful regression rate data. Approximately 50 lbf of N_20 were pumped into the main oxidizer tank prior to the first hot-fire test. At the commencement of the test, the oxidizer tank pressure had risen to 725 psi caused by external heating by the ambient air. The motor ignited successfully and burned steadily until the main oxidizer valve was closed 7 seconds later. A steady thrust of approximately 225 lbf was obtained. The results of the first test are shown in Figure 21.

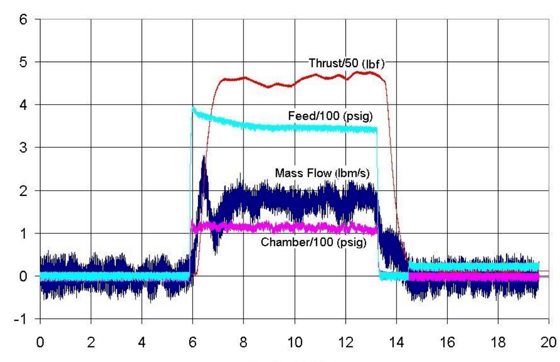

Fig. 21. Phase 2 ground test results (1st test).

Prior to the second test, the oxidizer feed system line was shortened to decrease heat transfer to the oxidizer prior to injection. It was hoped that the decrease would lessen the likelihood of oxidizer cavitation. The initial oxidizer tank pressure was a slightly lower 650 psi. No increase in mass flow was observed, however, a discontinuity in feed pressure was evident 4 seconds after ignition. The thrust profile was less steady, fluctuating about a level of 200 lbf for the first 4 seconds of the 10 second run.

A additional round of ground testing will be undertaken using the final flight configuration of the motor (oxidizer tank and combustion chamber closely coupled with no extra components in between).

Phase 3 Design

Expendable solid fueled motors, comprised of one or more military surplus stages (e.g. Orion and Terrier), propel the majority of sounding rockets launched today. These military surplus motors are often obtained at no cost and thousands are available, hence attempting to compete in the expendable sounding rocket market is futile. What is not readily available is a reusable sounding rocket that can be throttled and may even have a thrust vectoring capability. A rocket with these capabilities could carry expensive avionics systems, experiments and other payloads that could be reused. In addition, a thrust vectoring capability would permit certain upper atmospheric and re-entry experiments to be carried out that are not currently viable within the context of NASA's sounding rocket program. Phase 3 is a first step towards this goal.

Phase 3 is an Orion class N2O/paraffin-based hybrid with a parachute recovery system (see figure 22). The propulsion system length is approximately twice as long as the Orion but following list summarizes the benefits of the paraffin-based hybrid propulsion for the types of missions typically flown by the sounding rocket community.

- Recoverable and reusable
- Storable oxidizer/single circular grain design features of the N_2O/Paraffin-based system significantly improves the reliability and the operational simplicity. Note that these virtues are lost in a complex multi-port LOX/HTPB-based hybrid system.
- N_2O hybrid performs better than the Orion propulsion system.
- Hybrid adds throttling capability.
- Paraffin-based fuel grain is inexpensive and easy to handle/ship.
- Very low sliver fraction (< 1%)

The methodology behind the Phase 3 conceptual design was to utilize an existing sounding rocket design program to identify the optimum hybrid propulsion/vehicle configuration. A 3 degree-of-freedom flight trajectory integration scheme coupled to a hybrid rocket design code was the main design tool. Various parameters utilized in the optimization process are motor O/F, nozzle area ratio, chamber pressure and burn time. As a design constraint the OD of the hybrid vehicle was matched to the OD of the Orion system. Presented is figure 23 is the effects of adding aluminum to the fuel. Aluminum not only increases the specific impulse but also substantially reduces the O/F ratio required to achieve the best performance. In essence, the addition of aluminum exchanges light fluid oxidizer for a dense solid fuel. The results of the conceptual design study are summarized in table 5 and in figures 23-27. The trends in the predicted motor and rocket performance are very similar to that of the Phase 2 sounding rocket. Of course, Phase 3 is a much larger vehicle with a significant payload capability as shown in Orion comparison presented figure 24.

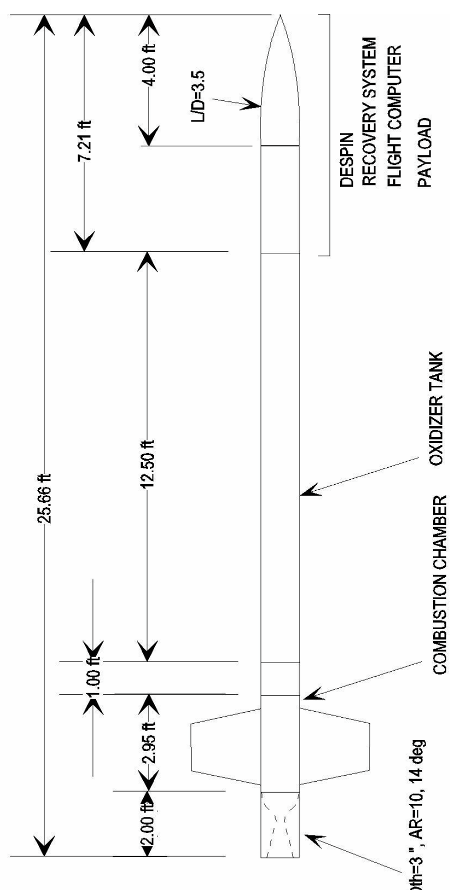

Fig. 22. Phase 3 conceptual layout.

Table 5. Overall System Parameters (Phase 3).

Fuel/Oxidizer	40% Alumunized Paraffin-based/N2O
Rocket Diameter, in	14
Overall Rocket Length, in	307.9
Gross Lift off Mass, kg	471.2
Launch Angle, degrees	85
Launch Altitude	Sea level
Baseline Payload, kg	30 kg
Apogee, km	145.4
Maximum Acceleration, g's	6.80
Maximum Mach Number	4.65
Maximum Dynamic Pressure, Pa	1.52×10^4
Time to Apogee, sec	204

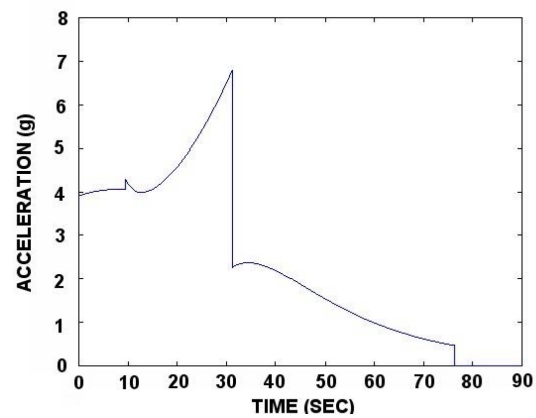

Fig. 25. Predicted acceleration time history (Phase 3).

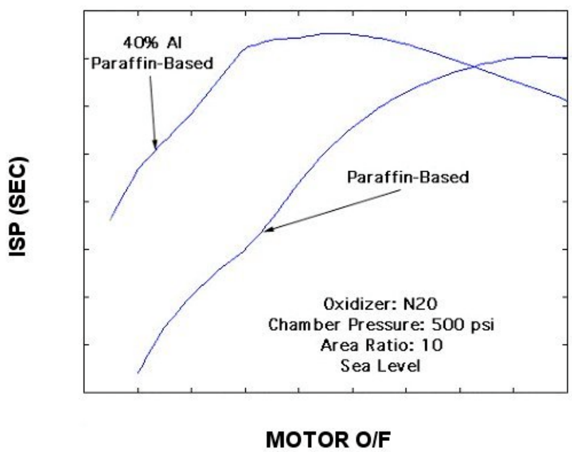

Fig. 23. Predicted specific impulse vs. O/F ratio (Phase 3).

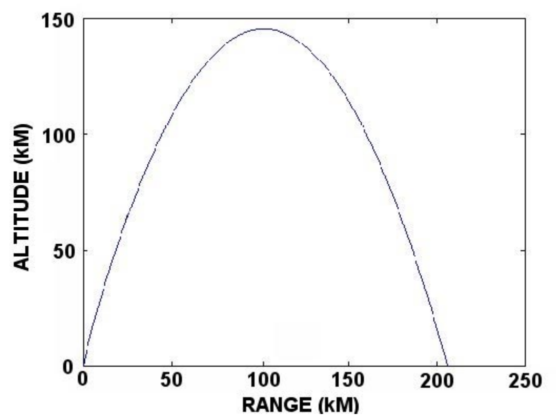

Fig. 26. Predicted altitude-range time history (Phase 3).

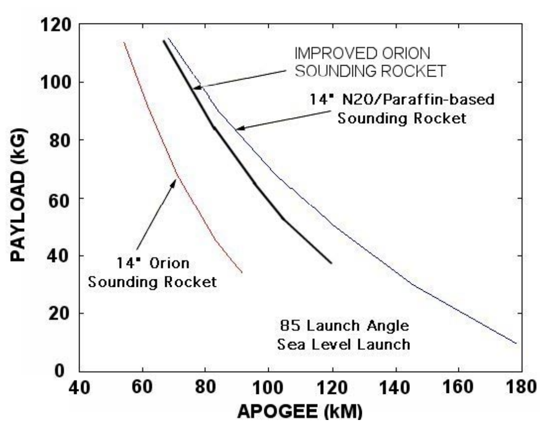

Fig. 24. Predicted payload comparison with Orion (Phase 3).

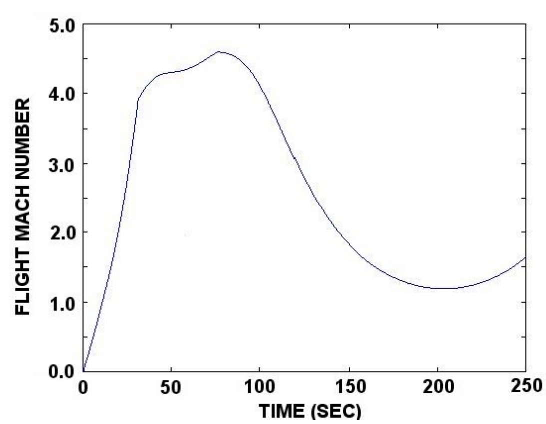

Fig. 27. Predicted Mach number time history (Phase 3).

Conclusions

Three liquefying hybrid technology demonstration rockets have been described. The Phase 1 rocket has flown. The Phase 2 rocket is expected to fly in 2003 and the Phase 3 rocket is in the preliminary design stages.

1.) The Phase 1 rocket was the first ever flight of a liquefying hybrid (i.e. paraffin fuel). Its groundbreaking flight showed that the high regression rate measured in laboratory-scale tests can also be realized in flight. It is difficult draw conclusions concerning the performance of the paraffin fuel in comparison to the original cellulose fuel in a motor that is not optimized for high-regression-rate fuels. Even so the flight data showed that the paraffin-based fuelled rocket reached a significantly higher apogee than the cellulose fuelled rocket (note that the N_2O tank was exhausted with approximately half the paraffin fuel remaining unburned).

2.) The Phase 2-rocket design is optimized for high regression rate fuels. The predicted apogee altitude is 88,000 ft. During ground tests of the motor, oxidizer cavitation was identified as a problem. It is expected that cavitation will be less of an issue in the closely coupled flight configuration of the motor but further testing is required.

3.) The Phase 3 rocket fits a niche that is currently not filled in the sounding rocket world. It is a reusable, throttleable rocket that has performance that promises to exceed that of an Orion. While it is realized that it will be difficult to compete on a cost basis with the limitless supply of military surplus solids, it is hoped that the unique capabilities of this vehicle will be attractive to an organization with the resources to fund its development.

Acknowledgement

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