Feasibility of Developing a Refrigerant-Based Propulsion System for Small Spacecraft

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ABSTRACT

This paper documents the feasibility of developing a low pressure, low-budget, two-phase refrigerant propulsion system for small spacecraft. The spacecraft design teams at the University of Missouri-Rolla, University of Texas at Austin, and Washington University in St Louis have collaboratively researched and assessed the feasibility of using a refrigerant propellant to provide a safe and practical type of propulsion system for the small spacecraft community. As an alternative to a typical inert cold-gas system, the teams investigated two-phase refrigerant-based systems motivated by the excellent propellant storage advantages and the ease of use and inherent safety. A primary benefit is its ability to be stored as a saturated liquid with inherently lower pressures as the constant volume system maintains self-equilibrium at saturation pressure. The associated laboratory safety of using a refrigerant propellant and ease of constructing cold-gas hardware make the propulsion system an ideal choice for low-budget satellite developers. The safety and performance analysis conducted on a general system indicates that with appropriate precautions and conservative design, test and analysis a refrigerant-based propulsion system can be safely implemented on small spacecraft and is a viable propulsion option. This feasibility study has been used as a guide to design and develop propulsion systems for each of the universities.

INTRODUCTION

In today's aerospace community, small spacecraft (mass range 10-100 kg) are becoming more prevalent due to their cost effective design. Internal spatial restrictions, however, limit the components and systems that can be integrated into the spacecraft, which is particularly problematic with large and robust propulsion systems. This is because most traditional propellants are stored in the gaseous state and must be held under high pressures in large containers or vessels. Of those propellants that can meet the strict volume criteria of a small spacecraft, the majority are either combustible and/or toxic and, thus, pose a greater risk to personnel and equipment. Another concern related to small spacecraft is that they are often flown as a secondary payload so there are significant safety concerns that must be addressed. The intent of this research is to source and justify the feasibility of an easy to use and safe propulsion system that can be utilized by lowbudget spacecraft developers. One potential alternate solution to traditional high pressure, large volumetric systems is storing the propellant as a two-phase saturated liquid. With these propellants, significantly more mass can be stored in the denser liquid state. Volumetric constraints associated with small satellite propulsion can be alleviated while still maintaining a relatively low storage pressure.

The concern with such a system is assuring that the system is inherently safe throughout all mission environments. For a system to be characterized as safe, it must meet or exceed specific guidelines and criteria; namely that the system must pose no substantial risk to the personnel or equipment encountered by the system throughout its operational and physical lifetime.

To evaluate this concern, a generic saturated liquid propulsion system is considered for use on any small spacecraft with mass and volume constraints. The objective is to perform a thorough safety assessment that includes temperature, pressure, and energy considerations to ensure all safety issues are addressed across a conservative environmental envelope. This study is targeted for the use of a refrigerant propellant, however, the approach used can be tailored for any potential saturated liquid propellant option. Refrigerants were selected for study because of their inherent safety, ease of use, and availability. R-134a and R-123 are the primary refrigerants analyzed and recommended for use as a small spacecraft propellant.

The two-phase propulsion system discussed in this paper will provide a small spacecraft with a compact, efficient, and above all, safe method of achieving its mission goals. The application of this study has been demonstrated with the design and development of the propulsion systems of each university for their respective Nanosat-4 satellites.

Propulsion System Design Requirements

Each of the universities was an individual participant in the University Nanosat Program (UNP), Nanosat-4 (NS4) competition that concluded in March 2007. The UNP was developed as a collaboration between the U.S. Air Force Research Laboratory Space Vehicles Directorate (AFRL/VS), the U.S. Air Force Office of Scientific Research (AFOSR), and the American Institute of Aeronautics and Astronautics (AIAA).

The mission objectives of each university drove the need to develop a low-cost propulsion system, while the requirements of the UNP competition motivated the search for a suitable propellant. The design requirements of the propulsion system were initially set to the payload restrictions on board the Space Shuttle, which are the industry's most stringent. NASA Technical Standard, (NASA-STD-5003), *Fracture Control Requirements for Payloads Using the Space Shuttle*, is a reference used in this study and defines the sealed container parameters. An initial specific requirement of this study was that the propulsion system must maintain a sealed container status, defined by two criteria¹:

- Maximum Design Pressure (MDP) < 100 psia (689.5 kPa)
- Stored energy < 14240 ft-lbs (19310 J)

Other practices and restricted hardware requirements that were utilized for this feasibility study included the prohibition of pyrotechnic devices and/or mechanisms, cast metallic or welded joints and the use of parts or assemblies for which safety is highly dependent upon the build or assembly process. Examples include composite materials and certain deployment mechanisms.

For this study, the selection of a propellant compound that met the following criteria was implemented:

- Nontoxic / non-flammable propellant
- Safe and easy laboratory handling procedures
- Environmentally friendly
- Easily obtainable without the need for licensing or permits
- Simple storage requirements
- Easily transportable
- Compatible and chemically inert with common spacecraft materials

Environmental Considerations of Using a Refrigerant

When considering the use of a refrigerant propellant, it is necessary to research the legal implications of purchasing, using and releasing the refrigerant. In the USA, the use and release of all refrigeration compounds is governed by the Environmental Protection Agency (EPA) Clean Air Act (CAA), specifically, Title VI Sections 604, 605, 608, and 612.

The refrigerant, R-123, as defined by the CAA is a Class II hydrochlorofluorocarbon (HCFC) and is in the process of being replaced by other more "environmentally friendly" refrigerants. The production and sale of Class II substances will be illegal no earlier than January 2015.

R-134a is a Hydrofluorocarbon (HFC), and consequently an alternate refrigerant that is friendlier to human health and the environment. A primary advantage of R-134a is that there are minimal restrictions on sales and it can be purchased in small quantities "off the shelf."

The CAA dictates that it is illegal to release a refrigerant into the environment if the application it is used for is a heat transfer fluid (refrigeration cycle). Used as a spacecraft propellant, there are no direct legal restrictions on release and containment. The quantities released are small and the intent is for educational and research purposes which do not directly fall under a category of the CAA². Additionally, the amounts released during testing in the laboratory and once on-orbit are very small in comparison to global release rates.

Saturated-Liquid Propellants in Space

Storing a propellant as a liquid has been practiced for years on a range of spacecraft and propulsion applications. The use of a saturated-liquid propulsion system, where the propellant is stored in two phases and the vapor is extracted and exhausted, is not new technology but has fewer flight applications.

Spacecraft developer Surrey Space Technology Limited (SSTL) launched the satellite SNAP-1 in June 2000. This 6.5 kg satellite successfully demonstrated the use of a cold-gas multi-phase propulsion system. A total of 32.6 g of butane propellant was stored as a liquid and vaporized by a heater before flowing out the valve /nozzle assembly³.

At operating conditions, the system is capable of providing a nominal thrust of 65 mN, with on-orbit results indicating a thrust of 46 mN was achieved. The specific impulse (I_{SP}) was measured to be 43 s that also suffered in comparison to the theoretical value of 70 s. Additional on-orbit data indicates that the propulsion system provided between 1.9-2.1 m/s in total change in velocity (ΔV), raising the orbit altitude between 3.1 and 3.4 km with a total of 98 firings, mostly of three second duration to give 297.1 s of total firing duration³.

Another satellite developed by SSTL and launched in 1999, was the 350 kg UoSat-12. This spacecraft utilized two propulsion systems, a standard cold gas system using nitrogen (N₂) stored at 2900 psi (20 MPa), and a revolutionary resistojet utilizing the storage of liquid nitrous oxide (N₂O). The N₂O was stored at a vapor pressure of 696 psi (4.80 MPa), and if used as a cold gas, would have an I_{SP} of approximately 66 s, however with the use of the 100 W resistojet the I_{SP} was raised to 127 s and produced a thrust of 125 mN. The N₂O resistojet was flown as a technology demonstrator for orbital maneuvers and produced a total of 10.4 m/s ΔV^4 .

The University of Toronto's Institute for Aerospace Studies has developed the CanX-2 spacecraft to establish flight heritage of propulsion technologies to be used on future CanX- spacecraft. The 10x10x30 cm spacecraft will implement a cold gas system storing Sulfur Hexaflouride (SF₆) as a liquid. At 21 °C, the vapor pressure of SF₆ is 315 psi (2.17 MPa) and the MDP is limited to 500 psi (3.45 MPa) with a relief valve. With a 10 ml storage tank, the target performance goals for the system are an I_{SP} of 45 s and 50 mN of thrust⁵.

Refrigerants in Space

Refrigerants in space are currently used primarily in conventional applications such as temperature control fluids in heat management systems. The Space Shuttle Orbiter has an active thermal control system that utilizes the refrigerant dichloromonofluoromethane (Freon-21). The Freon-21 circulates in two independent coolant loops that are used to remove heat from the water coolant loop system, fuel cell power plant and avionics systems and warms the oxygen supply line and hydraulic fluid system⁶.

NASA spacecraft Pioneer 12, which orbited Venus for 14 years, providing numerous maps and environmental data utilized liquid Freon in a partially filled tube for nutation dampening. Pioneer 10 used a bellows filled with liquid Freon that was controlled to thermally expand and contract, moving a piston which was used to time thruster firings aligning the communications antenna with Earth.

FEASIBILITY STUDY PARAMETERS

It is necessary to consider the entire environmental operating envelope and define design constraints when analyzing a refrigerant propulsion system to ensure all safety and performance scenarios are considered. There is heightened safety associated with a propulsion system because of the stored energy of a pressurized fluid and its dynamic and active nature that changes with environmental variations and usage. Detailed here are the design parameters used to conduct the feasibility study.

Temperature Range

The temperature envelope is an important parameter as the maximum value will correlate to the highest pressure and energy of a stored propellant. The temperature range of -50 °C to 100 °C is an extremely conservative range that has been chosen for use in this study to ensure that the safety and integrity of the system remains uncompromised. This temperature range accounts for fluctuations occurring on the ground, launch and in virtually any low Earth orbit (LEO). This range is justified as conservative based on the data taken from numerous satellites in LEO. This temperature range is also beyond the operational limits of numerous onboard space systems. For spacecraft components an acceptance test thermal range of -24 to 60 °C is an industry standard. The bounds are expanded to define the qualification test range, which spans at least -34 to 70 °C⁷.

Pressure Range

A primary objective of the study was to maintain a maximum pressure rating of a sealed container (100 psia). This pressure guideline severely restricts the mass of storable propellant on a small spacecraft and limits the life and objectives of most missions. As a consequence, a study of propulsion systems and their pressure ranges was undertaken to determine the feasibility of increasing this maximum pressure. A maximum of 100 psia can be deemed low pressure with high pressures reaching as high as 4500 psia⁸. In light of this observation it was deemed safe and practical to conduct the study with a maximum pressure of 350 psia (2.41 MPa). This is still a conservative maximum pressure limit and provided all measures are taken the system can be deemed safe for all launch providers.

Internal Energy

In a similar nature to the pressure requirement, the internal energy limitation of a sealed container 14240 ft-lbs (19310 J) has been maintained as a guideline but in reality will be exceeded in order to provide sufficient propellant.

Two-Phase Dimensional Comparison

When selecting a propellant it is important to exploit thermodynamic properties as well as meet performance and design constraints. With a refrigerant-based propellant, it is highly beneficial to study the two-phase characteristics and perform a dimensional-based analysis in order to quantify the envelope of operating conditions. Thermodynamic figures were used to show the properties of pressure, temperature, and state for four example cases of refrigerant propellant masses in a 1 L tank.

By defining the tank volume and using four propellant mass scenarios, the density is fully determined. By varying temperature, the second thermodynamic property, the pressure and propellant state can be determined and plotted. Figure 1 shows the dimensional example of the thermodynamic properties and operational conditions profile of R-134a. Figure 2 shows the application example of R-123.

Four propellant masses (25 g, 50 g, 75 g, and 100 g) representing a suitable range of realistic density implementations were chosen for comparison. The

expected temperature range used was -20 $^{\circ}$ C to 100 $^{\circ}$ C (-4 $^{\circ}$ F to 212 $^{\circ}$ F). A 1 L (61.02 in³) tank was chosen for analysis as it is a suitable size for small spacecraft integration and the results can be linearly scaled to fit other tank sizes.



Figure 1: Design Envelope R-134a in 1 L Tank



Figure 2: Design Envelope R-123 in 1 L Tank

Refrigerant Propellant Performance

It is necessary to analyze the performance of the refrigerants and beneficial to compare them with other cold gas propellants. R-134a and R-123 are analyzed alongside a number of typical cold gas propellants as well as other non-traditional two-phase candidates.

The primary properties used to compare propellants are the I_{SP} and ΔV . Also shown are other thermodynamic properties that should be considered when selecting a two-phase propellant. These include critical temperature, the two-phase saturated vapor pressure and the liquid density. The details of the performance and property comparison are displayed in Table 1.

Propellant	Change in Velocity (ΔV) m/s	Specific Impulse (I _{SP}) s	Critical Temperature °C	Saturated Vapor Pressure @ 20 °C, psia (kPa)	Liquid Density kg/m ³
R-134a	1.15	48.52	100.9	82.98 (572.1)	1150
R-123	1.24	31.60	183.86	10.98 (75.71)	1460
Xe	0.89	30.44	16.5	-	3057
CO ₂	0.63	64.38	31.0	830.92 (5729)	763
Ar	0.48	55.53	-122.3	-	1400
N ₂	0.47	75.58	-146.9	-	809
Ne	0.34	78.32	-228.8	-	1207
Не	0.15	175.93	-267.9	-	125
H ₂ O – Water	131.26	132.77	374	0.338 (2.33)	998
SF ₆ – Sulfur Hexafluoride	1.38	41.79	45.5	304.72 (2101)	1880
C_4H_{10} – Butane	0.91	65.09	134.9	43.8 (302)	556
NH ₃ – Ammonia	0.39	101.43	132.35	124.4 (857.8)	682

Table 1 Propellant Theoretical Performance and Thermodynamic Property Comparison

The data that were utilized in this analysis were generated using Engineering Equation Solver (EES), distributed by McGraw-Hill, 2006. EES calculates the thermodynamic properties using a real fluid, highaccuracy, equation of state. This equation of state includes all two-phase properties and can be used in the proximity of the critical point.

The I_{SP} is calculated with a regulated nozzle inlet operating pressure of 30 psia (206.84 kPa) and temperature of 20 °C. The ΔV was calculated using the sealed container restriction of maximum absolute pressure 100 psia (689.48 kPa) and the conservative maximum temperature of 100 °C to generate the maximum amount of propellant storable in a 2.5 L (152.5 cu. in.) tank. The nozzle parameters used are a throat diameter of 0.5 mm and an area expansion ratio of 100. The ΔV calculated is for a 25 kg spacecraft.

There are many assumptions used to calculate these theoretical performance parameters including the approximation of the propellant being exhausted as a calorically perfect ideal gas at 20 °C (68 °F). Isothermal conditions were also utilized, assuming that the stored propellant, tank and hardware maintained a fixed temperature. These are valid assumptions in this analysis given that it is a tool to compare the relative performance of each propellant as opposed to an absolute analysis of the individual system.

The saturated vapor pressure is calculated at 20 °C. The propellants that have a critical temperature below 20 °C will exist as a single phase (superheated gas) at

the operating conditions and thus do not have a corresponding saturated vapor pressure listed.

A high ΔV is an advantageous parameter for orbital maneuvers. As shown, the heavy molar mass of the refrigerants increases the ΔV in comparison to the other traditional cold gases.

Water (H₂O) is a safe and cheap propellant that offers good performance parameters. The limitation of water is its severely low vapor pressure at the desired operating conditions. The power budget required to obtain sufficient temperature and vapor pressure is beyond the bounds of a small spacecraft. H₂O offers a significantly higher ΔV value as it exists in a two-phase state when the propellant mass is calculated at the maximum environmental conditions.

Sulfur Hexafluoride (SF₆) offers a high ΔV of the analyzed propellants. The disadvantage of SF₆ is its low critical temperature in combination with its high vapor pressure. If the sealed container limitations were to be extended, SF₆ would suffer in performance compared to the refrigerants. Additionally, SF₆ is the most potent greenhouse gas with a global warming potential that is 23,900 times greater than CO₂ as per the U.S. EPA classifications⁹ and consequently was not considered a viable propellant for this study.

Butane and other hydrocarbons were not considered because of their flammability. Ammonia was removed from eligibility due to its corrosive and caustic hazards to humans. It also possesses flammability potential at low percentage air mixtures.

Saturated Liquid Phase Changes

Utilizing a two-phase propellant has many advantages for a small spacecraft, however, it is important to develop a thorough understanding of the phases that will be present in the system. The likely location of a phase change can be predicted and measures can be implemented to prevent this occurrence. To operate a two-phase propellant, a phase change or at least a single phase, gas, is desired for propulsion exhaust. The necessary hardware to achieve this must then be incorporated. This section discusses the areas of a propulsion system where a phase change is likely to occur.

The tank is intended to store the propellant in a twophase saturated liquid state. Depending on the temperature range, the propellant will exist in an equilibrium state of liquid and vapor or may become a single phase of superheated gas. Maintaining self regulated equilibrium in the tank is a highly desired advantage of the two-phase refrigerant propellant. The continual phase changes occurring in the tank are both anticipated and desired. The concerns of reduced structural integrity with any associated temperature reductions as a result of phase changes must be addressed.

The feed lines and other hardware components are another area to be considered. During maneuvers, the propulsion system is required to release only gas for optimum performance and efficiency. If vapor is extracted from the tank, the phase change from liquid to gas will occur in the tank and it is necessary that no further phase changes occur. Alternatively, if liquid is extracted from the tank, it is necessary that heat is transferred to the fluid and a phase change to gas occurs in the system prior to nozzle release.

If vapor is extracted from the tank there is a minor concern that undesired phase changes may occur as the gas flows through hardware components. The reason for this phase change is due to variations in cross sectional areas, surface finishes and other system environment conditions. Temperature and pressure changes, as well as energy losses will be encountered, with the possibility of causing the gaseous propellant to condense back to a liquid state. The locations where this could occur are through valves, nozzles, regulators and general system fittings.

The concern of a phase change from liquid to solid is not considered as the temperature required for both R-134a and R-123 is beyond the bounds of the temperature envelope. At atmospheric pressure, the freezing temperature for R-134a is -96.6 °C, (-142 °F), while for R-123 the freezing temperature is -107.2 °C, $(-161 \text{ °F})^{10}$.

Latent Heat effects

When a substance undergoes a phase change there is associated energy involved with this process which is commonly known as latent heat. Energy transfer to or from a fluid will cause a temperature change in the fluid. During a phase change, however, the energy is used in changing state and the fluid temperature remains constant. The energy (enthalpy change) associated with a change of state from liquid to vapor is known as the latent heat of vaporization, which is the phase change of interest here.

The latent heat of vaporization is a measure of the energy required to convert the fluid from liquid to gas at the given boiling point conditions. Refrigerants, by nature have a high heat of vaporization as this maximizes the cooling that is achievable. When a liquid undergoes vaporization to a gaseous state, the process is endothermic. This results in the refrigerant absorbing energy (heat) from its surroundings (i.e. positive latent heat of vaporization). When a gaseous refrigerant condenses to a liquid state, the process is exothermic. The resulting energy (heat) is being transferred to its surroundings (i.e. negative latent heat of vaporization)¹¹. The strong endothermic nature of the latent heat of vaporization of refrigerants is evident during laboratory testing of R-134a. When R-134a is stored in a saturated liquid state and then exhausted as a gas, the pressure vessel and surrounding apparatus experience a significant temperature drop as the phase change to gas absorbs the surrounding heat energy.

An important consideration is the effect these enthalpy changes have on the propulsion system and propellant during a phase change, whether desired or undesired. When the stored liquid propellant undergoes vaporization to gas, the tank and surrounding hardware (valves, tubing, fittings etc.), will experience a decrease From a safety viewpoint, this in temperature. temperature drop is not a significant problem as the propellant pressure remains constant with constant temperature. If the propellant suffers a temperature loss, the pressure similarly reduces. From a performance perspective, the thrust is proportional to the gas temperature of the propellant, thus if the gaseous propellant temperature drops the performance characteristics will also reduce.

If the energy levels are significantly low, there is potential for the refrigerant to condense back to the liquid form. This is a safer, lower energy state, but will degrade performance characteristics. The thermodynamics of the refrigerant under these scenarios are very important and must be considered when investigating and designing a propulsion system. While the safety of a refrigerant system should not be compromised with latent heat effects during these scenarios, it is important that the propulsion system being designed undergo thorough analysis and laboratory testing. This will ensure the propellant properties and conditions are known and best utilized and the performance levels are maintained.

Phase Change Actions and Control Methods

Whether to induce a phase change or to prevent an undesired change, it is necessary to implement hardware and mission strategies when using a twophase propellant.

In the tank it is necessary to monitor the propellant thermodynamic properties at all times, particularly when a heater is implemented. Monitoring should be implemented for safety reasons to ensure no high pressures are encountered and to assist in propulsion performance characteristics and calculations. Monitoring hardware can include thermal sensors and pressure transducers.

With a two-phase refrigerant propellant it is necessary to also implement an active thermal control system with tank heaters. It may also be necessary to utilize heaters downstream of the tank, such as on the lines or nozzles to ensure only vapor is propelled. This active monitoring is advantageous for safety reasons as well as performance characteristics.

Multi layer insulation (MLI) should also be implemented when a heater is used. This will increase the efficiency of the system by reducing any radiated heat transfer lost to the surroundings.

Hardware Selection

The most important aspect when selecting hardware components for a spacecraft propulsion system is safety as identified in this feasibility study. It is necessary to mitigate hazards that arise as a result of hardware failure. There is heightened fear with a propulsion system due to the propellant's pressurized stored energy.

As well as NASA-STD-5003, another primary source of safety requirements information for hardware was obtained from the NASA document NSTS 1700.7B, *Safety Policy and Requirements for Payloads using the Space Transportation System.* The military standard MIL-STD-1522A - *Standard General Requirements for Safe Design and Operation of Pressurized Missile and Space Systems,* was also used as a safety reference. The tank is a crucial component of the propulsion system. It must safely store the active propellant throughout the entire mission from ground operations to orbit. The primary tank requirement corresponds to a pressure factor of safety relating burst pressure to MDP. The requirements utilized in this study that initially originated from the UNP competition were:

- Factor of safety greater than five (Burst : MDP)
- Structural fatigue test diagnostics
- Leak before burst failure
- Metal construction (No composites or overwrapped tanks)
- Constructed/welded by certified manufacturer

As per the guidelines of NSTS 1700.7B all lines, fitting and components shall have a pressure ultimate factor of safety of at least four (if diameter is less than 1.5 inches). In a similar manner other components such as valves and regulators require a pressure factor of safety of at least 2.5.

The valves will serve two primary functions in a propulsion system design. The first is for control, where the valves are used to hold and release the propellant with the required timing. The second use will be as an isolation safety feature providing a physical interruption of propellant flow between tank and nozzles. The valves, as defined by NASA NSTS 1700.7B, will be the inhibitors of the propulsion system. It is required that there are three mechanically independent flow control devices (inhibitors) in series to prevent catastrophic hazard in the case of premature valve opening.

It is also required to have one of the three inhibitors as a "fail-safe" valve where it will close by default in the absence of an open electrical signal. While any series of three valves will be mechanically independent, it is also required that the electrical inhibits that operate the valves be arranged such that they operate individual valves. This ensures that if there is a failure of one electrical inhibit there will only be a maximum of one flow control device opening. Further specific hardware safety requirements have been addressed in the application section.

A general requirement of the study is that all hardware acquired includes a manufacturer's Certificate of Compliance (CoC) and full materials list. The CoC ensures that the hardware is designed, manufactured and tested to the specifications quoted. The materials list is used to confirm that materials meet outgassing limitations, and corrosion and flammability resistance. The materials list must be provided for components purchased from vendors as well as university manufactured items.

Outgassing Requirements

Outgassing is a measure of the release of gas by a nonmetallic material leading to a loss in mass. This loss of material is heightened by the vacuum of space and can cause contamination of surrounding hardware, or lead to component malfunction or failure. NASA has developed a list of low outgassing materials based on two properties that quantify the outgassing potential of a material in a vacuum. These two properties are the collectable volatile condensable material (CVCM) and a total mass loss (TML). The material properties that NASA defines as low outgassing, and are a requirement of this study, are a maximum CVCM of 0.1% and a TML of 1.0% or less.

Material Compatibility

One significant concern with cold gas propulsion systems is their tendency to leak, which is a significant concern when propellant mass is already at a minimum. In order to minimize leaks, plastic or elastomer seals are often used in fittings, and component connections. It is highly recommended that the use of non-metal seals be avoided wherever possible in a spacecraft propulsion system due to compatibility and outgassing concerns. As an alternative all metal components are recommended.

Some hardware components cannot function without a non-metal seal. When selecting a seal compound for these components it is necessary to account for both compatibility and outgassing. If a seal is not compatible with the fluid, then it may suffer degradation, strength loss and other physical changes that may lead to malfunction or even failure of the seal.

Limited information on chemical compatibility can be found in the Material Safety Data Sheet (MSDS), *Stability and Reactivity* section. The MSDS should be consulted for both the refrigerant propellant as well as the seal compound under consideration. The manufacturers of these materials also publish technical documents and compatibility data that can be referenced.

The MSDS specifies that R-134a should not be used with Viton rubber, magnesium alloys, alkali metals or zinc. Other compatibility information is available from automobile refrigerant systems. Common gasket materials compatible with R-134a include polyacrylate, high-grade neoprene and hydrogenated nitrile butadiene rubber.

Chemical compatibility data of certain compounds are often not available. In addition, the compatibility of compounds such as refrigerants is often not available. Compatibility of plastic or elastomer compounds is also heavily dependent on the application environment and the thermodynamic conditions requiring care to be taken when researching the compatibility of substances.

SAFETY ASSESSMENT

While the current studies of a refrigerant propellant identify advantageous performance and thermodynamic properties the disadvantages and safety concerns must be thoroughly reviewed and assessed. This will ensure the propellant will operate within the guidelines set by the university developers and the satellite community. In that regard, the safety assessment will identify possible hazards and failures in the system. From this system, each identifiable hazard associated with the propellant can be classified, the risk assessed, and mitigation solutions found. A detailed report is presented for each hazard and is summarized by the safety program efforts to highlight the acceptable risk for flight. This is applicable not only for a refrigerant propellant, but any cold gas two-phase propulsion system.

Hazard Classification and Identification Methodology

To begin the safety assessment, a hazard classification system was tailored for this study and used the following definitions:

- **Catastrophic** A catastrophic hazard is defined as any single or multiple system failure which has the potential to cause damage/harm not only to the spacecraft, but to surrounding equipment/personnel as well.
- **Critical** A critical hazard is defined as any system failure which results in damage/harm to the spacecraft and/or has the potential to negatively impact mission objectives to the point of failure.
- **Tolerable** A tolerable hazard is defined as any system failure which results in minimal damage to the spacecraft/mission without failure.
- Acceptable Risk for Flight Acceptable risk for flight is defined as operating the system with known hazards classified as tolerable or with hazards which can be mitigated by use of the appropriate safety devices and measures.

Based on these definitions, hazards are classified not by the likelihood of their occurrence but rather by the ramifications of said occurrence. In this way, identified hazards can be ranked on a relative scale, and the impact of each identified; thus enabling proper design choices to be made.

Safety Analysis and Design

Hazards present serious risks to personnel and equipment and are possible in all engineered systems. Identification of all such hazards within a system is the only possible way to ensure that proper mitigation efforts are in place. In a two-phase propulsion system such hazards may be caused by natural thermodynamic events (i.e. temperature changes due to ambient conditions or system usage) or component failures.

The general design of any propulsion system contains many possible hazards of each classification. In most cases, propellant is initially stored in a small, pressurized vessel and from there distributed to the thrusters by means of tubing. By taking into account mission objectives, a prototype design can be implemented; however, before the design can be further refined, the safety assessment must be completed to ensure selected components meet the mitigation criteria.

A. Catastrophic Hazards

The greatest risk inherent to the system comes from uncontrolled and unexpected changes in the state of the propellant. The catastrophic hazard is directly caused by an increase in system temperature, but may have many indirect causes. As a result of this increase, the pressure of the propellant could rise to levels above the maximum design pressure mandated for the system components, which in turn could lead to increased leak rates and/or system rupture. The use of storage tanks defined as pressure vessels greatly amplifies the effects of burst since they contain enough internal energy to seriously impact the surrounding area. Both passive and active methods of mitigation are available to combat the adverse effects of pressure increase. The first passive measure is simply designing the storage vessel with a large enough factor of safety to withstand any fluctuations within the system. Also, the system should be designed to be leak-before-burst; thus alleviating dangerous over-pressurization through low energy fluid discharge rather than an explosive release of energy. The active method uses sensors to monitor system conditions and discharges the system once dangerous levels have been reached.

Another consequence of a rise in temperature is felt within the system materials. Many materials, metallic in particular, expand and contract with changes in temperature causing excess stress at connection points. If these stresses are not accounted for in the design of the system, increased leak rates and/or rupture could occur. Additionally, if materials with dissimilar thermal expansion rates are used at connection points, the possibility of mission damaging leaks increases many fold. Two possible sources of differing thermal properties are the use of multiple materials (e.g. aluminum connected to steel) and the existence of thermal gradients between connected components. To guard against the possible consequences of thermal expansion, proper material selection must be performed with particular attention to obtaining sufficient yield and fracture stress properties, and if possible, avoiding the use of dissimilar materials.

Finally, under drastic conditions and extreme temperatures, the selected refrigerants have the added hazard of decomposition and even the possibility of auto-ignition. Decomposition of R-134a and R-123 occurs at temperatures above 250°C and auto ignition at or above 743°C and 770°C respectively¹². All values are well above the expected temperature range; however, the seriousness of the consequences produced by this hazard merits mention. Both refrigerants decompose into highly volatile and caustic chemicals, such as hydrofluoric acid, which can cause serious burns and weaken equipment. Care should be taken during construction and storage of the satellite so propellant does not come into contact with excessive heat such as open flames.

When dealing with pressure vessels, structural strength of the selected material is of the utmost importance. However, merely designing to worst case scenarios is no guarantee of successfully avoiding structural failure since thermal cycling has, in addition to those risks associated with the corresponding maximum and minimum temperatures, the potential to cause structural failures due to thermal fatigue. Temperature fluctuations for a two-phase propellant system can occur due to both system and environmental influences. During propulsive maneuvers the endothermic phase change lowers the overall system temperature. Environmental factors, such as leaving and entering eclipse, can also cycle system temperatures. To avoid thermal fatigue, it is first necessary to thermally insulate the system through use of MLI which will greatly reduce the effects of the spacecraft's environment. To reduce the effect of system processes, system monitoring and some method of energy addition to the system (i.e. heaters) are required. The heaters should be turned on during propulsive maneuvers to account for endothermic phase change and minimize thermal gradients. Finally, system materials should be chosen in such a way as to limit the effects of thermal cycling where possible.

B. Critical Hazards

Catastrophic hazards may pose the greater threat to surrounding equipment and personnel; however, critical hazards are no less destructive to mission success. As with hazards classified as catastrophic, critical hazards are often products of the propellant state whereas mitigation methods normally center on proper component selection and procedures.

The effects of a temperature decrease within the system represent a critical hazard rather than catastrophic as the internal energy contained within the system is far less than that for the case of temperature increase. As such, the overall magnitude of possible consequence for any resulting failure is less. This does not mean, however, that thermal decrease can be ignored. Any substantial decrease in the temperature of the fluid will result in a phase change. If the temperature falls to the freezing point of the propellant, the fluid will solidify. The effectiveness of the propulsion system's internal mechanisms will be reduced with a potential of damage to internal mechanics of the tank if any of the solid propellant freely moved. However, the system need not reach the propellant freezing point in order for a hazard to be present since there exists the potential for system materials to experience reduced structural integrity (brittle) due to the low temperatures generated by the fluid. Also, as with thermal expansion, thermal contraction can lead to propellant leakage and eventual mission failure if different contraction rates exist between components. Mitigation efforts should include system heaters and insulation to lessen the probability of significant temperature decrease. Also, system materials should be selected to avoid mismatched thermal contraction rates and materials which can become brittle within the expected temperature range.

Temperature and pressure are not the only propellant properties to consider during a hazard analysis; the material compatibility and potential for chemical reactivity are also a concern. While refrigerants are generally chemically inert, as previously mentioned there are certain substances with which a negative reaction can occur. Any system material should be thoroughly researched for its compatibility with the chosen propellant. System materials which have direct contact with the propellant must have a zero to very low reactivity rating to ensure continued system When determining an acceptable functionality. degradation rate, mission length should be accounted for with appropriate margins. For shorter missions, a somewhat faster reaction rate might be acceptable so long as mission goals are not negatively impacted; however, longer missions require much lower reactivity. Materials with no or limited exposure to the fluid under normal operating conditions must also be considered since any leaks could bring said material in contact with the propellant. To prevent harm to equipment and personnel, any material reactions determined to be explosive or combustible require the selection of a different material. Where material reselection is not possible, such as on board the launch vehicle, it is important to make sure the system has minimum leakage to lessen the chance of reaction with an unknown material.

C. Tolerable Hazards

Throughout ground operations, there is the possibility of exposure to the propellant which is a tolerable hazard that can be avoided. Direct skin contact can have two results: skin irritation and/or frostbite. Skin irritation is a symptom of chemical exposure to the refrigerants, while frostbite results from the low temperature nature of the refrigerant. Asphyxiation is possible if proper venting is not present during the discharge of any propellant. Personnel should be required to wear suitable protective clothing and eyewear. In addition approved ventilation and warnings should be instituted in the work environments where potential exposure to the propellant can occur.

DESIGN APPLICATION

With the promising results of the feasibility study it is beneficial to apply the assessment to an actual hardware system. The three universities that developed this study are designing and constructing propulsion systems based on this guide.

University of Missouri – Rolla (UMR)

The UMR satellite design team (UMR SAT) developed two satellites, Missouri–Rolla satellite (MR SAT) and Missouri–Rolla secondary satellite (MRS SAT), to advance studies and knowledge of distributed space systems missions. The specific emphasis is to study autonomous close formation flight, in which the larger satellite (MR SAT) will utilize an onboard propulsion system to maintain a 50 m \pm 5 m orbital separation. The mission objective is to perform a minimum of one orbit of formation flight after separation as shown in Figure 3.



Figure 3: MR SAT and MRS SAT On-Orbit Shortly After Separation

The propulsion system will be used primarily for orbital maneuvers but can also be used as a supplementary three-axis attitude control device with eight thrusters geometrically placed around the hexagonal structure of MR SAT. The propellant selected for use is R-134a.

The UMR SAT team utilized this design guide on two separate levels. The first was with the application of the thermodynamic and performance analyses through engineering modeling. The second was with the design and procurement of hardware.

Analysis

Analyses were conducted to model the properties and performance of the two-phase refrigerant propellant system. An important feature of the study that was utilized in these analyses was the implementation of the conservative temperature range (-50 °C to 100 °C). Utilizing the sealed container limitations completely defines the thermodynamic operating constraints.

Very early in the design phase, propellant comparisons and trade studies were conducted using these sealed container and thermal envelope parameters. R-134a refrigerant was compared to more traditional cold gas propellant options. Under these constraints and UNP requirements the refrigerant propellant demonstrated the most advantageous performance properties as required for the UMR SAT mission.

Of critical importance to the UMR SAT mission is the duration of formation flight, which is a direct function of the amount of propellant loaded in the tank and its rate of consumption. The initial mass of propellant is proportional to the maximum pressure for the given environmental conditions. The propellant rate is a function of the hardware configuration and nozzle properties.

Engineering models were developed to analyze the expected performance of the R-134a propellant. This was used to assist in the design of hardware configurations such as the use of a regulator and its output pressure. Nozzle performance was conducted and ultimately its design was finalized with this modeling based on the requirements of the UMR SAT mission. Figure 4 shows the nozzle performance characteristics (ΔV and I_{SP}) of R-134a as a function of area ratio that was used for nozzle design.

This preliminary analysis demonstrated that the UMR SAT mission objective could not be achieved under the sealed container constraints. There was insufficient mass for R-134a as well as the other cold gas propellants that were analyzed to provide adequate ΔV .

It was therefore required to model the system beyond the limits of the sealed container.



Figure 4: R-134a Nozzle Performance Characteristics

The analysis was conducted with the equivalent temperature range, however, the maximum pressure limit was extended to 310 psia. The internal energy requirement of the propellant was also exceeded. This refined engineering model indicated that the UMR SAT mission objective (of conducting a minimum of one orbit of formation flight) could be achieved by increasing the maximum pressure to this still conservative value. This value was chosen as the saturated vapor pressure will never exceed 307.2 psia at a maximum temperature of 70 °C, regardless of the mass of propellant.

An outcome of the feasibility study was that it would be necessary to implement a thermal control system to account for the endothermic nature of the liquid to vapor phase change. An analysis of the R-134a propellant was conducted to quantify the required energy input and a suitable heater power budget to account for the high latent heat of vaporization.

Hardware Selection

In addition to the structural requirements of a storage vessel there are additional MR SAT requirements that were considered when selecting a propellant tank:

- Space certified and tested highly preferred
- Flight history preferred
- Propellant management device integrated
- Stainless steel preferred
- Dimensions within design envelope
- Mass less than 2 kg
- Volume range 2 L to 3 L
- All wetted materials compatible with R-134a

A tank was selected and purchased "off the shelf" that was designed for small spacecraft propulsion systems intended to store saturated liquids. The tank is space qualified and is currently operating in LEO on four satellites. It meets the required pressure factor of safety (five) for the UMR SAT mission. The tank also features a propellant management device system that utilizes mesh baffles to reduce liquid sloshing during maneuvers and promote vapor extraction with increased internal surface tension attracting liquid propellant.

The entire MR SAT propulsion system is constructed of stainless steel (SS) to reduce effects of thermal loading between components and fittings as identified in this study. SS tubing of 1/8 inch outer diameter has been integrated into the satellite. Standard "off the shelf" SS compression fittings have been used to distribute the propellant to the thrusters. The tubing and fittings meet the required pressure factor of safety. The propulsion system integrated into a disassembled MR SAT structure is shown in Figure 5.



Figure 5: Propulsion System Integrated in MR SAT

The MR SAT propulsion system features ten valves located with two in series downstream of the tank and one at each of the eight thrusters. The valves function as a control device as well as inhibits in the system. For simplicity and ease of manufacture, it was decided that each of the ten valves would be identical and all are consequently "fail-safe" as required.

The valves selected are micro-solenoid devices for quick response, high precision fluid control. They meet pressure factor of safety requirements and have been tested to required proof levels. They were manufactured by aerospace professionals in custom configurations and with compression fittings for direct MR SAT integration. The valve with nozzle and tube fitting (thruster assembly) is shown in Figure 6.



Figure 6: Thruster Integrated on MR SAT Side Panel

Heaters were implemented on the tank and on the feed line to actively regulate the thermodynamics of the system and improve propulsive performance. The primary function of the tank heater is to prevent and mitigate temperature losses during maneuvers, improve system response, and ensure the propellant maintains its optimum thermodynamic properties. As a passive thermal control device to efficiently utilize the heaters and conserve any heat loss due to radiation, the tank will be wrapped in multi layer insulation (MLI).

Material compatibility and outgassing have been a key concern of the UMR SAT propulsion team. The standard valves feature a Viton rubber seal which is incompatible with R-134a. Other options were available, however their outgassing rates were not known. Compatibility testing was conducted in the laboratory with two alternate seal compounds. A grade of silastic® silicone and a ethylene propylene diene monomer (EPDM) compound were tested in the laboratory for compatibility with R-134a. After prolonged exposure the samples showed no changes in geometry, mass, texture or properties and were deemed compatible. Outgassing testing remains to be completed for both compounds.

Similar procedures were carried out for other necessary sealing compounds. Custom all metal fittings were also designed and manufactured to avoid the issues of compatibility and outgassing.

A fundamental step in the design process of a spacecraft propulsion system is to perform laboratory testing. Primary advantages of using refrigerant

propellants, particularly R-134a, are their easy and user-friendly testing with inherent safety as a working fluid, and purchase availability. Testing has been conducted in the UMR laboratory using low cost "off the shelf" R-134a. The R-134a refrigerant is easy to handle, transport and store in the laboratory. As per the recommendations of this study, the lab is well ventilated and has the necessary safety gear for personnel. All tests were conducted with at least two personnel supervising for added safety.

The preliminary testing was conducted to quantify the fluid dynamic losses that can be expected with a refrigerant vapor flowing through system hardware. Test data were obtained and used to generate friction factor correlations with the stainless steel tubing over a range of operating conditions. These data have since been integrated into the engineering models to refine the accuracy of the performance predictions.

The laboratory testing was further developed to include losses associated with fittings and bends that are to be encountered in the MR SAT system. Preliminary functional and performance testing of the valve and nozzle is also underway. It has been necessary to implement additional filters upstream of the valve to ensure contaminants do not obstruct functionality.

University of Texas at Austin

The University of Texas at Austin is currently in the developmental stages for the ARTEMIS (Autonomous Rendezvous Thermally stabilizing Enabling Modular Inspection Satellite) propulsion system. Similar to UMR SAT, ARTEMIS consists of a chaser and target satellite that will separate on-orbit and autonomously rendezvous within a target distance of 50 m. ARTEMIS will investigate the use of integrated sensors (sensor fusion) to combine and simplify attitude and orbit sensors. Additionally, through the use of the sensor fusion technology, ARTEMIS will demonstrate satellite on-orbit inspection. Finally, the last goal of ARTEMIS is to design a common satellite bus which will allow for varying subsystem designs to be easily and rapidly integrated.

The propulsion system aboard ARTEMIS will be used to maintain attitude and orbit control. The entire system consists of seven thrusters in which six are devoted to attitude control and one for orbital control. Figure 7 shows the three attitude nozzles that are placed on the corners of the Chaser and are directed inward. The exact angles at which the thruster will be placed are still in design process.

Figure 8 shows an isometric view of the bottom of the Chaser. The orbit thruster is positioned directly along

the center axis. The remaining three attitude thrusters are positioned on the opposing corners of the Chaser.



Figure 7: Top View of ARTEMIS Chaser



Figure 8: Bottom View of ARTEMIS Chaser Analysis

Analysis performed by UT initially focused on selecting a propellant that was capable of meeting mission and UNP requirements. The investigation centered on selecting either R-123 or R134a as the propellant. Aided by the feasibility report, it was determined that R134a was the best candidate propellant. In addition to quantifying R134a and R-123, other propellants used in previous missions were investigated. However, due to safety restrictions and mission requirements such as combustibility and delta V all other candidate propellants were ruled out.

Additional analyses have focused on thermodynamic modeling using a simplified turbulent 1-Dimensional (1D) flow in a duct. Currently, the 1-D model is

capable of performing simple estimations of thrust for various nozzle configurations. In order to adequately model the propulsion system, the loss coefficients for individual components must be adequately known. Actual loss coefficients will be applied to the model once final hardware is selected.

A trade analysis was conducted on various materials to quantify material compatibility with R134a. Material compatibility was investigated through literature published by the Air-conditioning and Refrigeration Technology Institute and material compatibility guides published by various companies. Several materials have been identified thus far but will require laboratory testing to verify compatibility with R134a.

As a result of the analysis conducted thus far, it was determined that the cold-gas two-phase system for ARTEMIS would exceed the sealed pressure vessel and internal energy limitations. The propulsion system is currently being designed to a maximum operating pressure of approximately 250 psia. The selection of the new maximum operating pressure was a result of the thermodynamic properties of R134a. In order to adequately control the phase of the propellant a simple model of the phase change system is currently being developed.

Hardware

Hardware selection has focused around meeting certain requirements such as material compatibility, pressure/thermal ranges, and budgetary concerns (mass, volume, cost). Test hardware was chosen based on commercial off the shelf (COTS) availability. The use of COTS hardware allows for low system costs but has the limitations of not having flight heritage or allowing customization.

The first test system for ARTEMIS consisted of a proof of concept (POC) test bed. The test bed consisted of a propellant tank, pressure regulator, plenum, solenoids, manifold and nozzles. The POC test bed is shown in Figure 9.

Critical lessons learned was the need for adequate component seals and need for further nozzle analysis. Other design issues learned through the POC test bed focused on the lack of adequate thruster valves (solenoids). The solenoid manufacturer was unable to replace the internal seals. The Fluorocarbon (FKM) seals were found to be incompatible with R134a which could lead to complete valve failure. The propulsion team is currently researching replacement solenoid valves with compatible seals. The seal trade analysis identified several seals, with Ethylene Propylene (EP) as a leading candidate. It was found that EP seals exhibits low swelling and low outgassing properties.



Figure 9: ARTEMIS Proof of Concept Test Bed

Additional design issues with the POC are the lack of adequate inhibits. As required by the feasibility study regulations, the propulsion system is required to have a total of three inhibits. The first and second inhibits are lacking from the test bed. The absence of inhibits was due the high maximum operating pressure (~300 psi), power consumption, and need for low leak rates. Throughout the hardware selection process it was found that low leak rate COTS valves tend not to operate above 100 psi and/or require large amounts of energy (~10W typically). However, a valve configuration known as a latching valve is under investigation. While latching valves require high power (~10 W) they are advantageous in that they require only a small impulse (~2-5 ms) to open /close and require no power to remain open/closed.

System components such as tubing and various fittings were selected based on thermal, pressure ranges, and size. Due to mass and volume concerns it was determined that 1/8 inch stainless steel tubing and fittings would offer the optimal balance between mass, stress resistance, and volume concerns. The use of 1/8 inch tubing presents its own difficulties such as difficult assembly process and the ease of damage.

The design of the phase change system is currently underway. The test bed depicted in Figure 9 did not have an active thermal control system. However, the future iteration of the test bed calls for the inclusion of an active system stationed in two locations: filter-trap and plenum. As was found in the feasibility study, refrigerants require high amounts of energy to undergo a phase change and maintain temperature. Thus, in order to minimize losses in supplied energy, the phase change will occur primarily in the filter-trap (located immediately after the propellant tank). The plenum will be supplied energy (through resistive heat wraps similar to Figure 5) to maintain a relatively constant propellant temperature.

Planned Testing

At this point in the design process the propulsion system for ARTEMIS is ready for preliminary testing. Test plans involve the qualification of the individual system components such as fittings and valves. Additionally, system concepts such as the propellant management and thermal control systems will be validated under Earth and micro gravity conditions.

The prime concern for ARTEMIS is the propellant management system. The system involves the use of a filter-trap and possibly baffles within the propellant tank. The function of the system is to separate vapor propellant from any liquid or liquid/vapor mixtures downstream of the tank. In order to quantify the system a simple test using the filter-trap and propellant tank will be conducted. The propellant will be stored as a liquid/vapor mixture and forced through the filter-trap. Mass flow rate will be measured along with pressure and temperature to quantify the quality of the propellant along with the effectiveness of the filter-trap system. Baffles may be included to increase the effectiveness of the system by retaining more amounts of liquid propellant inside the tank.

The thermal control system will revolve around the filter-trap system. Testing planned for the thermal system is to quantify the necessary energy needed to induce a phase change. Testing will be conducted under normal Earth gravitational conditions. However as was found in feasibility study, a fluid will behave significantly different under micro-gravity conditions. In order to test R134a in micro-gravity drop tower tests and C-9 micro-gravity flight tests will be conducted to measure the amount of energy and effectiveness of the thermal control system.

Washington University - St Louis

The Washington University mission is composed of two spacecraft, a six degree-of-freedom (6DOF) microscale (3 kg) free-flyer called Bandit and its host, Akoya (29 kg). Bandit's mission is to flight-test proximity operations technologies, including repeatable docking, navigation within 5 m of a target, on-orbit charging, and image-based navigation. Bandit's mass and power requirements are reduced by distributing long-term power, ground communications, complex processing, and docking to Akoya. Distributed subsystems allow Bandit to be very small, while allowing space for mission-specific components, like the propulsion system. The engineering units of each respective satellite are shown in Figure 10.



Figure 10: Bandit and Akoya Engineering Units

Bandit contains an active propulsion system for proximity operations; Akoya, the docking station, uses only magnetorquers for attitude control. Bandit's propulsion system is used in free-flying to and from Akoya.

Bandit's propulsion system consists of ten valves and eight thrusters that give Bandit 6DOF maneuvering capabilities. Two sets of four thrusters are located around the center of mass at compound 45° angles on opposite parallel sides of Bandit. Each thruster is managed by a single solenoid control valve. The control valves are connected to a manifold distribution block, which is connected to the propellant tank with two isolation valves, as seen in Figure 11. This design provides three inhibits between tank and nozzle as per requirements.



Figure 11: Propulsion System Schematic Showing Propellant Flow

The positioning of Bandit's thrusters are shown in Figure 12. The thrusters are positioned such that they stay near the spacecraft center of mass as propellant mass is decreasing during the mission. Due to Bandit's small size, the propellant contributes a significant fraction of the total mass. The mission requires sufficient propellant to perform 12 m/s of ΔV .



Figure 12: Bandit Side Panel Showing Thrusters

Analysis

Analysis performed by Washington University focused on selecting a propellant that could provide the required ΔV and I_{SP}. The trade study included cold gas propellants, which did not offer adequate energy to complete the mission. R-134a was selected based on its pressure/temperature curve, lower C_p/C_v, molecular weight, and the ability to impart significant changes in internal energy with less on-board power.

Further analysis examined the thermodynamic properties of R-134a. This analysis focused on identifying temperature and pressure operating ranges, and determined how much heating would be required to maintain desired pressures. Thermodynamic analysis helped determine the desired quality upon fill, or the ratio of liquid to saturated vapor of R-134a.

Due to volume constraints, a majority of analysis was devoted to designing a custom square tank to contain the two-phase propellant, as seen in Figure 13. Due to Bandit's very small size, purchasing an off the shelf tank was not an option. The tank will operate at a nominal pressure of 90 psi, with a MDP of 250 psi. The tank was hydrostatically verified to 3,150 psi, a factor of safety of 12.6, at which point it leaked only. Attempts to burst the tank failed; the leak prevented catastrophic failure.

Analysis to determine the ideal operating pressure was inconclusive. A simplified representation of a twophase system was used to determine the approximate thrust force values, but the exact configuration of the valve and tube system was not settled, so the results are only approximate. It was decided that vacuum chamber testing would provide more useful results.

Hardware

The constrained volume envelope and the amount of necessary propellant dictated that Bandit needed a custom square tank. The tank is an R-134a-compatible aluminum, 7075-T73. An o-ring was required to seal the tank. The traditional Viton rubber o-rings are not compatible with R-134a; the two best alternatives are Buna-A and EPDM (Ethylene Propylene). Buna-A costs significantly more than EPDM, therefore EPDM was chosen as the o-ring material of choice. The choice was also confirmed by engineers from the various manufactures of the connection fittings used in Bandit.



Figure 13: Bandit Propulsion Tank

Bandit's tank is uniquely designed such that it feeds directly into the manifold without a mechanical connection. All connections between tubing, valves, and the manifold are made via miniature compression fittings shown in Figure 14. The compression fittings are stainless steel and have an EPDM o-ring to make a tight seal. Additionally, the o-rings provide a medium to accommodate differences in thermal expansion of the different materials.

Bandit uses 1/16 inch OD stainless steel tubing between the thruster blocks and the manifold. Stainless steel tubing was chosen to match the thermal properties of the stainless steel compression fittings.

Bandit uses the same solenoid valve for the isolation valves and the control valves. The valves are Lee Co. extended performance solenoid valves. They are some of the smallest solenoid valves commercially available "off the shelf" that can withstand Bandit's operating pressure. The wetted surfaces of the valves are compatible with R-134a, based on extensive research involving the manufacturer and R-134a compatibility resources.



Figure 14: Bandit Propulsion System Plumbing

To fill Bandit's tank, a fill valve is located in the center of the bottom of the tank. The location allows for filling while Bandit is inside Akoya, such that Bandit may remain unfilled until last possible moment, reducing handling time with a pressurized device. The fill valve is a stainless steel check valve with an EPDM o-ring seal that is activated by an inlet pressure 1 psi larger than the internal pressure of the tank. This method was developed to eliminate the need for traditional large and heavy fill valves.

Testing

Bandit's tank and tank lid have been subjected to many tests demonstrating the sealing capability of the design, as well as the compatibility of EPDM and other materials with R-134a. Test tanks filled with R-134a have been set aside for over eight months, during which the tank's mass was regularly checked to verify that propellant was still present. The tanks were also exposed to heat to test performance under pressure increase. Inspection of the EPDM o-ring following the tests showed no degradation. Additionally, EPDM oring samples have been directly exposed to liquid and saturated vapor R-134a in the lab with no compatibility issues. EPDM meets the outgassing requirements of the feasibility study.

Bandit was test-flown on the NASA C-9 "Vomit Comet" in May, 2007. The goal of the test was to demonstrate the ability of Bandit's propulsion system to maneuver Bandit. Bandit reacted as expected to directional and rotational thrusts. Quantitative results were impossible because miniature accelerations of the C-9 during microgravity over-powered the small force from the thrusters. Those aircraft forces will not be present in true microgravity.

Bandit's propulsion tank underwent hydrostatic testing and has been proven to leak before burst at a pressure of 3,150 psi. Once leak occurred it was impossible to generate enough pressure inside the tank to cause catastrophic failure; despite attempts to drive the pressure to 10,000 psi, 2,800 psi could not be exceeded after leak.

A test apparatus is presently being designed to measure the thrust produced by the complete propulsion system in a vacuum environment. This test will measure both axial force and rotational torque produced by the different thruster combinations that make up the 6DOF maneuvering capability. The results, combined with analysis, will provide a more accurate understanding of the performances characteristics of the R-134a thruster system. This test will help determine more accurate methods of modeling losses due to small tubing diameter, tube bends, and expansion points inside valves.

Thermal testing is in design to observe pressure changes with temperature. These values will be compared to the theoretical values from analysis and tables. This test will help identify the temperature and pressure interaction of an entire propulsion system, not just of R-134a. This will allow fine-tuning of individual components.

Another concern is the on-orbit distribution of liquid and saturated vapor inside the tank. To better understand this, pressure testing will be conducted to determine the variability between thrusts where vapor, saturated vapor, and liquid are expelled from the thrusters. This test will identify the difference in performance based on the quality of the R-134a in the tank.

Along with the pressure testing, tests on energy required to induce phase change will be conducted. These tests will determine the amount of energy required to thermally control the R-134a. These tests will be conducted in a vacuum chamber with normal gravity.

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